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ARMY PRELIMINARY EVALUATION I.
MODEL 200 CEFLY LANCER

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Army Aviation Engineering Flight Activity
Edwards Air Force Base, California

June 1974

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The United States Army Aviation Engineering Flight Activity conducted an Army Preliminary Evaluation of the Model 200 CEFLY LANCER aircraft, manufactured by Beech Aircraft Corporation, from 20 February to 6 March 1974 at the Beech facility in Wichita, Kansas. During the test program, 24.8 productive hours were flown. Performance, stability and control characteristics, and miscellaneous engineering tests were conducted. During these tests, three enhancing characteristics,

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20. Abstract

four deficiencies, and eighteen shortcomings were noted. The enhancing characteristics included the stall warning horn automatic readjustment with flap setting to provide a computed warning at approximately 10 knots indicated airspeed above stall, the rudder boost feature during asymmetric power conditions, and the excellent braking characteristics. The deficiencies determined were pitch attitude instability during loading and ground operations of the aircraft in normal mission configuration (aft center of gravity), the requirement to maintain a continuous 2000-rpm propeller speed during ground and taxi operations, loss of power to mission equipment and primary attitude and heading gyros when propeller speed was less than 2000 rpm, and smoke in cockpit and cabin areas at altitudes above 15,000 feet pressure altitude. The most significant shortcoming was the inadequate single-engine performance under heavy gross weight or high temperature conditions in the single-engine takeoff configuration. Consideration should be given to the installation of an autopilot and weather radar system in production aircraft to reduce pilot workload and enhance the all-weather capability of the aircraft.

ERRATA

USAAEFA PROJECT NO. 74-21

ARMY PRELIMINARY EVALUATION I

MODEL 200 CEFLY LANCER

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

1. Pages 46 and 53. Photos captioned B-3 and B-10 were transposed during the printing process. Although these pages are not reprinted, readers are advised of this error.

2. Appendix D, page 74. Add Figure 1, Handling Qualities Rating Scale.

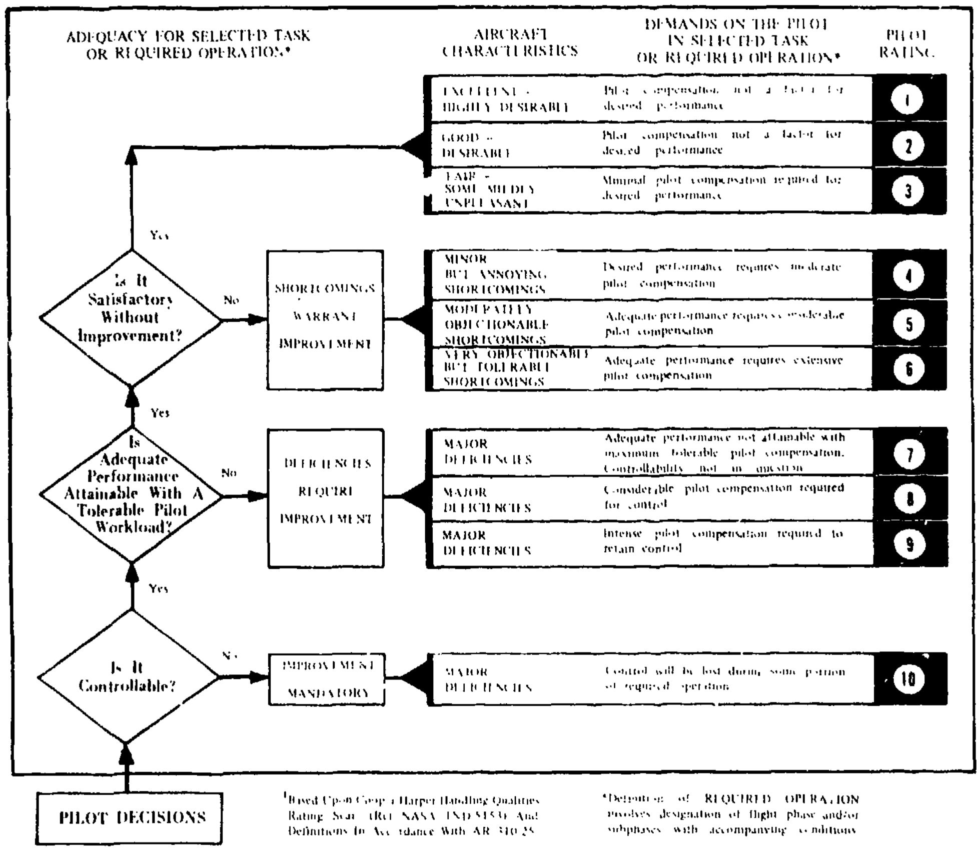

Figure 1. Handling Qualities Rating Scale.

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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 A100 (U-21F) aircraft. These aircraft were to be utilized by the United States Army Security Agency (USASA) 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 speed 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. Phase I was to be conducted with the basic airplane without the external mission equipment installed. Phase II tests were to be conducted with a mission configured airplane.

TEST OBJECTIVES

2. The objectives of APE I 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.

DESCRIPTION

3. The test aircraft was a BAC Model 200, serial number 71-21058, powered by two United Aircraft of Canada, Limited (UACL), PT6A-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 is retractable with dual wheels on each main gear. The 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; and 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. A detailed description of the Model 200 aircraft is contained in BAC Prime Item Development Specification BS22296A and Procurement Specification PS3102 (refs 3 and 4, app A). Appendix B contains a further description and photographs of the test aircraft.

TEST SCOPE

4. The APE Phase I tests were conducted at the BAC facility in Wichita, Kansas, from 20 February to 6 March 1974. During the test program, 16 flights were conducted for a total of 32.3 hours, of which 24.8 hours were productive. The Model 200 aircraft was evaluated to determine performance and handling qualities, and to verify contract compliance. Average test conditions are shown in tables 1 and 2 and test configurations are shown in table 3. The contractor takeoff and landing performance tests were witnessed by test team personnel. Representative takeoffs and landings were performed to spot-check contractor test results. 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. Performance Test Conditions.[1,2]

Test Description	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)	Configuration
Takeoff	2,000	17	15,000	197.4 (aft)	Not applicable	Takeoff
Landing	2,000	17	13,500	197.0 (aft)	Not applicable	Landing
Sawtooth climbs	13,260	-29	14,180	187.3 (fwd)	110, 120, 130, 140, 150	Cruise
	2,810	-12	14,360	187.3 (fwd)	130, 140, 150, 160, 170, 180	Cruise
	3,350	-8	14,340	187.3 (fwd)	100, 110, 120, 130, 140, 150	Takeoff
	3,210	-9	14,800	187.7 (fwd)	120, 130, 140, 150	Single-engine cruise
	2,600	-5	15,000	197.4 (aft)	110, 115, 120	Single-engine takeoff
Continuous climb	Surface to 28,400	-40	14,700	197.2 (aft)	140	Cruise
	12,000 to 12,910	-17	14,300	197.1 (aft)	130	Single-engine cruise
Zero thrust glide	9,560	-8	11,370	183.2 (fwd)	120, 160, 200, 240	Glide
	8,880	-14	11,060	183.1 (fwd)	110, 130, 140, 150	
	8,460	-20	14,000	187.5 (fwd)	110	
	9,440	-10	14,820	187.4 (fwd)	120	
	11,360	7	14,760	187.9 (fwd)	140	
Level flight	15,330	-12	11,900	183.4 (fwd)	[3]100 to 210	Cruise
	13,260	-29	14,600	187.6 (fwd)	[3]105 to 215	
	10,220	-3	14,750	193.3 (mid)	[3]120 to 160	Single-engine cruise and power approach

[1] All tests performed at propeller speed of 2000 rpm.
[2] Stall tests listed in table 6.
[3] At approximately 10-KCAS (knots calibrated airspeed) increments.

Table 2. Handling Qualities Test Conditions [1,2,3]

Test Description	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)	Configuration
Static longitudinal stability	2,580	-13	14,670	187.6 (fwd)	140	Cruise
	2,660	-13	14,780	187.8 (fwd)	170	
	2,620	-14	14,950	188.0 (fwd)	200	
	3,900	-4	14,560	187.5 (fwd)	140	Power approach
	5,300	8	14,780	197.2 (aft)	140	Cruise
	5,300	8	14,880	197.3 (aft)	170	
	5,120	7	14,990	197.4 (aft)	200	
	5,260	7	14,630	197.2 (aft)	140	Power approach
Static lateral-directional stability	9,840	-6	14,330	197.2 (aft)	140	Cruise
	9,840	-6	14,780	197.2 (aft)	170	
	10,400	-6	14,900	197.3 (aft)	200	
	9,840	-6	13,970	197.1 (aft)	140	Power approach
	10,600	Zero	14,190	197.2 (aft)	170	
Dynamic longitudinal stability	10,860	2	14,130	197.0 (aft)	140	Cruise
	10,420	-1	13,960	197.1 (aft)	170	
	10,630	Zero	14,440	197.1 (aft)	200	
Dynamic lateral-directional stability	11,040	4	14,220	197.0 (aft)	140	Cruise
	10,900	4	14,460	197.1 (aft)	170	
	11,000	5	14,800	197.2 (aft)	200	
Maneuvering stability	10,340	-2	14,000	197.1 (aft)	140	Cruise
	10,340	-2	14,100	197.1 (aft)	170	
	10,340	-2	14,600	197.2 (aft)	200	
Roll performance	10,680	1	13,820	197.2 (aft)	120	Power approach
	11,020	5	14,960	197.4 (aft)	200	Cruise

[1]All tests performed at propeller speed of 2000 rpm.
[2]Stall tests listed in table 6.
[3]Control system and trim change characteristics were evaluated during the conduct of other handling qualities tests.

Table 3. Airplane Configurations.

Configuration	Landing Gear Position	Flap Setting (%)	Power Setting
Takeoff (TO)	Down	40	Takeoff
Cruise (CR)	Up	Zero	Power for level flight
Landing (L)	Down	100	Flight idle
Power approach (PA)	Down	40	Power for level flight
Glide (G)	Up	Zero	Power off, propellers feathered
Single-engine cruise (SE CR)	Up	Zero	Power for level flight on left engine
Single-engine takeoff (SE TO)	Down	40	Maximum continuous power on left 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. Additional data were recorded on a 21-channel oscillograph and by a motion picture camera located on the ground and in an F-34 Bonanza chase aircraft. 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 variety of operating conditions with emphasis on operation in the normal mission configuration 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), and to military specification MIL-F-8785B(ASG) (ref 15) to assist in determining future military applications. Four handling qualities deficiencies were identified. These included pitch attitude instability during loading and ground operations of the aircraft in normal mission configuration (aft center of gravity) (cg), the requirement to maintain a continuous 2000-rpm propeller speed to perform ground and taxi operations, loss of power to mission equipment and primary attitude and heading gyros when propeller speed was less than 2000 rpm, and smoke in the cockpit and cabin areas at altitudes above 15,000 feet pressure altitude (Hp) Eighteen shortcomings were noted. The single-engine performance capability under heavy gross weight or high temperature conditions was inadequate in the single-engine takeoff configuration. Other shortcomings included three stability and control shortcomings, five cockpit evaluation shortcomings, and nine reliability and maintainability shortcomings. Three enhancing characteristics were (1) the stall warning horn, which automatically readjusted with flap settings to produce a computed warning at approximately 10 knots above stall, (2) the rudder boost, which greatly reduced pilot workload during asymmetric power conditions, and (3) excellent braking characteristics.

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 single-engine takeoff 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.

Takeoff and Landing Performance

8. Prior to the conduct of APE I, the takeoff and landing performance tests were performed by the contractor at Fort Bliss, Texas (field elevation 3947 feet) and monitored by USAAEFA personnel. These results were spot-checked at the BAC facility at Wichita, Kansas (field elevation 1387 feet) at the conditions shown in table 1. Contractor-recommended takeoff and landing airspeeds were used during the conduct of these tests.

9. A maximum performance takeoff was conducted to verify minimum ground run distance, rotation speed, and lift-off speed. Acceleration was rapid upon brake release. Rotation was initiated at 85 knots indicated airspeed (KIAS) and lift-off occurred at 95 KIAS. The ground run distance was 2600 feet, which was 250 feet less than that determined by the contractor for the test day condition. Within the scope of the tests, the takeoff performance data provided by the contractor were satisfactory.

10. A maximum performance landing from a power approach was conducted to verify approach airspeed, touchdown airspeed, and minimum ground roll distance. Approach airspeed was 120 KIAS. Touchdown airspeed was approximately 92 KIAS with 100 percent flaps. Full reverse was utilized immediately after the nose wheel contacted the ground. The ground roll distance was 1200 feet, which was 5 feet less than that determined by the contractor for the test day condition. Within the scope of these tests, the landing performance data provided by the contractor were satisfactory.

Climb Performance

Sawtooth Climb:

11. Dual and single-engine climb performance 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 maximum continuous power (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. Test results are presented in figures 1 through 13, appendix E. The climb drag polar equations for the Model 200 aircraft without the antenna array are presented in table 4.

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
CR	0	0.0325	0.04684	0	0	0
	1	0.0325	0.04684	0.22917	0.0893	-0.00080
	2	0.0325	0.04684	0	0.0893	-0.0018
TO	0	0.06518	0.0539	0	0	0
	1	0.06518	0.0539	0.7500	0.0558	-0.01398
	2	0.06518	0.0539	0	0.0558	-0.01398

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

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

12. At a maximum gross weight of 15,000 pounds, the aircraft has a positive dual-engine rate of climb of 1800 feet per minute (ft/min) at the recommended best-rate-of-climb airspeed of 138 KCAS in the cruise configuration at sea level on a standard day. At the same conditions in the takeoff configuration, the rate of climb was 1430 ft/min. At a maximum gross weight of 15,000 pounds in the cruise configuration, the aircraft has a positive single-engine rate of climb of

370 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, the rate of climb was -35 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 (aft cg) is a shortcoming. An Equipment Performance Report (EPR) concerning this shortcoming was submitted (ref 16, app A).

Continuous Climb:

13. Dual and single-engine continuous climb performance were evaluated at the conditions shown in table 1. A dual-engine continuous climb was conducted using MCP until full forward power lever settings were obtained at the engines' critical altitude. An abbreviated single-engine continuous climb was performed with the left engine shut down and the propeller feathered. The best-rate-of-climb airspeed schedule determined from the sawtooth-climb data was used.

14. The dual-engine climb airspeed schedule was easy to establish and maintain at all altitudes until passing through 25,000 feet H_P. Above this altitude, aircraft response to control inputs was sluggish and airspeed control demanded frequent attention. The dual-engine service ceiling was determined to be 28,400 feet density altitude (H_D) at a gross weight of 14,500 pounds. This exceeded the 27,750-foot H_D service ceiling predicted by the contractor by 2 percent. A single-engine climb to service ceiling was initiated by intercepting the single-engine airspeed climb schedule at 11,500 feet. The single-engine service ceiling was determined to be 12,910 feet H_D at a gross weight of 14,300 pounds. This exceeded the 11,850-foot H_D service ceiling predicted by the contractor by 9 percent.

Level Flight Performance

15. Level flight performance was evaluated at the conditions shown in table 1 to determine the maximum level flight airspeed, recommended cruise airspeed, range, and endurance capabilities. The zero thrust glide test method was used to obtain the base-line drag polar for the aircraft. The aircraft was stabilized and trimmed at incremental airspeeds in a descent with both engines inoperative and the propellers feathered. The constant pressure altitude technique for single-engine (propeller feathered) and dual-engine drag polar was used. The aircraft was stabilized and trimmed at incremental airspeeds from the maximum airspeed for level flight (V_H) to the stall airspeed (V_S). Performance at conditions not specifically tested was calculated from the drag polar and power-available data, which included installation and accessories losses. The results of these tests are presented in figures 14 through 19, appendix E. Aircraft specific range, recommended endurance, cruise airspeed, and V_H in level flight for the cruise configuration are summarized in figures 20 through 23. The level flight drag polar equations for the Model 200 aircraft without the antenna array are presented in table 5.

Table 5. 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
CR	0	0.0325	0.04684	0	0	0
	1	0.0325	0.04684	0.857	0.070	-0.004
	2	0.0325	0.04684	0	0.140	-0.0055

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

16. At a representative gross weight of 15,000 pounds at 15,000 feet, standard day, the maximum dual-engine airspeed in level flight using MCP was 256 knots true airspeed (KTAS). The recommended maximum range airspeed was 225 KTAS and the recommended endurance airspeed was 175 KTAS. The maximum single-engine airspeed in level flight (left engine shut down and propeller feathered), using MCP on the right engine at 5000 feet, standard day, was 180 KTAS. The recommended single-engine airspeeds for maximum range and endurance were 173 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.

Stall Performance

17. Stall performance was evaluated at the conditions shown in table 6. These tests were conducted by establishing trim configuration at the desired airspeed, then making a slight pitch attitude increase and decelerating at a rate of approximately 1 knot per second until achieving a stall. Test results are presented in table 6.

18. At a representative gross weight of 15,000 pounds, the Model 200 aircraft stall airspeed was 78 KIAS in the landing configuration, 79 KIAS in the power-approach configuration, and 104 KIAS in the cruise configuration. Initial stall warning was provided by the stall warning horn at approximately 10 KIAS above the stall airspeed. Additional warning was provided by a moderate buffet at approximately 2 KIAS in advance of the stall. A detailed discussion of stall characteristics is presented in paragraphs 36 through 38.

Table 6. Stall Test Conditions and Characteristics.

Configuration	Normal Load Factor	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Indicated Airspeed (kt)	Outside Air Temperature (°C)	Density Altitude (ft)	Indicated Airspeed (kt) Horn	Buffet	Stall	Angle of Attack (deg)	Roll Attitude (deg)	Power Setting
Cruise	1.0	14,800	197.3 (aft)	140	6.0	10,640	117	107	104	17.8	Zero	Off
Cruise	1.15	14,770	197.3 (aft)	140	7.0	10,760	123	112	110	18.8	30L[1]	Off
Cruise	1.19	14,690	197.2 (aft)	140	5.0	9980	124	113	110	17.8	30R[1]	Off
Landing	1.0	14,640	197.2 (aft)	120	3.5	10,600	91	81	78	21.5	Zero	Off
Landing	1.15	14,540	197.2 (aft)	120	4.5	10,360	96	88	88	17.5	30L	Off
Landing	1.19	14,620	197.2 (aft)	120	6.5	9320	94	85	85	17.0	30R	Off
Power approach	1.0	14,490	197.1 (aft)	120	3.5	11,800	95	79	79	17.8	Zero	PLF[2]
Power approach	1.13	14,470	197.1 (aft)	120	3.0	11,480	96	92	92	17.8	30L	PLF
Power approach	1.19	14,450	197.1 (aft)	120	4.0	11,520	96	90	90	15.5	30R	PLF
Cruise	1.0	13,890	197.2 (aft)	170	8.0	13,560	106	102	99	9.2	Zero	PLF
Single-engine cruise[3]	1.0	14,350	197.1 (aft)	120	5.0	11,580	111	106	105	15.0	Zero	PLF
Single-engine cruise[3]	1.0	14,240	197.0 (aft)	120	3.0	11,500	110	105	105	12.0	Zero	MCP
Takeoff	1.0	14,400	197.1 (aft)	120	3.0	12,620	89	83	83	19.5	Zero	MCP
Single-engine takeoff[3]	1.0	14,210	197.0 (aft)	120	4.0	11,260	95	87	84	15.0	Zero	MCP

[1]L = left, R = right.
[2]PLF: Power for level flight.
[3]Minimum-control airspeed tests conducted at these conditions.

HANDLING QUALITIES

General

19. 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 aircraft was compared to MIL-F-83785B(ASG) to assist in determining future military applications. Three deficiencies were identified: pitch attitude instability during loading and ground operations of the aircraft in normal mission configuration (aft cg), the requirement to maintain a continuous 2000-rpm propeller speed and perform ground and taxi operations, and loss of electrical power to mission equipment and primary attitude and heading gyros when propeller speed was less than 2000 rpm. Three shortcomings identified were poor long-term trimmability; elevator control force gradient reversal in the power-approach configuration; and lightly damped, easily excited, Dutch-roll oscillations. Three enhancing characteristics were noted: the stall warning horn, which automatically readjusted with flap settings to produce a computed warning at approximately 10 knots above stall; the rudder boost, which greatly reduced pilot workload during asymmetric power conditions, particularly during single-engine tests; and the excellent braking characteristics.

Control System Characteristics

20. Control system characteristics were evaluated on the ground (longitudinally only) and in flight at the conditions shown in table 2. 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. Control system positions in trimmed forward flight are presented in figures 24 through 27, appendix E. The results of the control system evaluation are summarized in table 7. There was no detectable lag in aircraft response for either small or large control input amplitudes along any control axis. Elevator and aileron force and displacement sensitivities were compatible and intentional inputs to one control axis did not cause inadvertent inputs to another axis. Control harmony was good and there was no tendency for the pilot to induce undesirable motion. However, 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, and required moderate pilot compensation to maintain adequate airspeed control (HQRS 4). The poor long-term trimmability was objectionable and constitutes a shortcoming. An EPR concerning this shortcoming was submitted (ref 17, app A).

Table 7. Control System Characteristics.[1]

Test Description		Test Results
Longitudinal (elevator)	Control travel	6.6 in. from full forward to full aft
	Breakout force (including friction)	4 lb (MIL-F-8785(B)ASG min: 0.5 lb; max: 4 lb)
	Gradient	65 lb/in. Steep positive force gradient; however, mild departures from trim (±6 KIAS) did occur with the controls free
	Free play	None noticeable. Does not result in objectionable flight characteristics
	Centering	Positive at any normal trim setting; friction prevents absolute centering
	Control dynamics	Well damped
Lateral (aileron)	Control travel	140° from full right to full left (12.85-in. arc from full right to full left)
	Breakout force (including friction)	2 lb (MIL-F-8785(B)ASG min: 0.5 lb; max: 3 lb)
	Gradient	9 lb/in.
	Free play	0.125 in. left and right from trim; does not result in objectionable flight characteristics
	Centering	Positive; friction prevents absolute centering
	Control dynamics	Well damped
Directional (rudder)	Control travel	7.14 in. from full left to full right
	Breakout force (including friction)	13 lb (MIL-F-8785(B)ASG min: 1 lb; max: 14 lb)
	Gradient	90 lb/in.
	Free play	None detected. Does not result in objectionable flight characteristics
	Centering	Positive
	Control dynamics	Well damped

[1]Evaluation conducted at 170 KCAS.

Static Longitudinal Stability

21. The static longitudinal stability characteristics were evaluated at the conditions shown in table 2. 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 28 through 35, appendix E.

22. The elevator control force gradient, as indicated by the variation in elevator control force with airspeeds, was positive for airspeeds above and below trim in the cruise configuration, indicating stable static longitudinal stability. In the power approach configuration, at 140-KCAS trim conditions, the elevator control force gradient became negative at airspeeds below 120 KCAS in the normal mission configuration (aft cg). This lightening of control force gradient became negative at airspeeds below 120 KCAS. This lightening of control force was objectionable and required moderate pilot compensation for adequate airspeed control (HQRS 4). 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 on the pilot. The static longitudinal characteristics failed to meet the requirements of paragraph 23.175d of FAR Part 23, in that the elevator control force gradient reverses in the power approach configuration below 120 KCAS, and the elevator control position gradient in the normal mission configuration (aft cg) is essentially neutral. These characteristics also did not meet the requirements of paragraph 3.2.1.1 of MIL-F-8785B(ASG). The elevator control force gradient reversal in the power approach configuration is a shortcoming. An EPR concerning this shortcoming was submitted (ref 18, app A). The aft cg limit of the normal mission configuration should be moved forward.

Static Lateral-Directional Stability

23. Static lateral-directional stability characteristics were evaluated at the conditions shown in table 2. 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 36 through 40, appendix E.

24. 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 resulted 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) or

until achieving a 180-pound rudder pedal force (200 KCAS). 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 acceptable.

25. 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

26. The dynamic longitudinal stability characteristics were evaluated at the conditions shown in table ... 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. Time histories of representative dynamic response characteristics are presented in figures 41 and 42, appendix E. Test results are summarized in table 8.

Table 8. Dynamic Longitudinal Stability Characteristics.

Test	Trim Calibrated Airspeed (kt)	Damping Ratio (ζ)	Undamped Natural Frequency (ω_n - rad/sec)	Damped Natural Frequency (ω_d - rad/sec)
Short-period	140	0.70	2.51	1.79
	168	0.75	2.74	1.81
	198	0.93	2.45	0.90
Long-period	140	0.025	0.114	0.114
	168	0.029	0.113	0.113
	198	0.036	0.106	0.106

27. The long-term aircraft response was oscillatory and was very lightly damped. The period was approximately 60 seconds. This weak damping, combined with the large friction band, contributes to the poor trimmability of the aircraft. There is no FAA requirement for long-term dynamics, but the long-term characteristics did fail to meet the requirements of paragraph 3.2.1.2 of MIL-F-8785B(ASG), in that the damping ratio of the phugoid oscillation was less than 0.04. Within the scope of this test, the long-term longitudinal dynamic characteristics are acceptable.

28. 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:

29. The dynamic lateral-directional characteristics (lateral-directional damping and Dutch-roll characteristics) were evaluated at the conditions shown in table 2. The yaw damper installed in this aircraft was inoperative for the duration of the APE. 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. Time histories of representative dynamic lateral-directional characteristics are presented in figures 43 through 46, appendix E. Test results are summarized in table 9.

Table 9. Dutch-Roll Characteristics.

Trim Calibrated Airspeed (kt)	Damping Ratio (ζ)	Roll/Yaw Ratio (ϕ/β)	Damped Natural Frequency (ω_d - rad/sec)
118	0.065	0.82	1.26
140	0.068	0.90	1.51
168	0.071	0.98	1.80
198	0.075	1.22	2.21
223	0.081	1.12	2.29

30. The Dutch-roll oscillations were lightly damped and easily excited. At the recommended maneuver airspeed of 170 KIAS, the period was 3.5 seconds and the damping ratio was 0.071. The aircraft's lateral-directional response and controllability characteristics were poor in atmospheric disturbances. Considerable pilot compensation was required to overcome the sensitive gust response during turbulent flight conditions at all cg locations (HQRS 5). The Dutch-roll characteristics failed to meet the requirements of paragraph 23.177(a)(d) of FAR Part 23 and of paragraphs 3.3.1.1 and 3.3.2.1 of MIL-F-8785B(ASG), in that any short-period oscillation must be heavily damped with the primary controls fixed and free. The lightly damped, easily excited Dutch-roll oscillations are objectionable and are considered to be a shortcoming. An EPR concerning this shortcoming was submitted (ref 20, app A). A reliable yaw damper or an autopilot system which would reduce lateral-directional pilot workload should be installed.

Spiral Stability Characteristics:

31. The spiral stability characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 2. 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 the bank angle achieved after 20 seconds elapsed time. Test results are shown in table 10. Spiral stability, as indicated by change in bank angle with elapsed time, was essentially neutral for both left and right turns. 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.

Table 10. Spiral Stability Characteristics.

Trim Calibrated Airspeed (kt)	Roll Attitude at Release (deg)	Roll Attitude at 20 Seconds (deg)
168	19.6, left	24.9, left
	21.6, right	23.0, right
198	18.2, left	13.7, left
	17.9, right	20.5, right

Maneuvering Stability

32. Maneuvering stability characteristics were evaluated at the conditions shown in table 2. The variation of elevator control force and control position with normal acceleration was determined by trimming the aircraft in coordinated level flight at the desired trim airspeed and then stabilizing at incremental bank angles in steady turns, both left and right. Airspeed was held constant and the aircraft was allowed to descend during the maneuver. Data were obtained at each stabilized bank angle. Symmetrical pull-up maneuvers were required to obtain load factors in excess of 2; symmetrical pushover maneuvers were required to obtain data below +1g. The load factor was varied incrementally to the maximum allowable during these maneuvers. The results of the maneuvering stability evaluation are presented in figures 47, 48, and 49, appendix E.

33. The stick-free maneuvering stability, as indicated by the variation of elevator control force with normal acceleration, was positive (increased aft elevator control force with increased load factor) and was essentially linear for all conditions tested. The elevator control force gradient (stick force per g) was approximately 32 pounds per g. The stick-fixed maneuvering stability, as indicated by the variation of elevator control position with normal acceleration, was positive (increased aft elevator control motion with increased load factor) and essentially linear. The control position gradient was approximately 0.4 inch per g.

34. Buffeting was encountered while attempting to achieve load factors in excess of 2 at 140 KIAS. The maneuvering control forces were high enough to prevent control inputs which might give abrupt aircraft responses and there was no tendency for the pilot to overcontrol. The maneuvering stability characteristics of the Model 200 aircraft met the requirements of MIL-F-8785B(ASG) and are satisfactory.

Roll Performance Characteristics

35. Roll performance characteristics were evaluated at the conditions shown in table 2. These tests were initiated from a trimmed unaccelerated flight condition by applying a rapid maximum deflection aileron control input without changing either elevator or rudder pedal control position. Representative test results are summarized in figures 50 and 51, appendix E. The adverse yaw was small (less than 2 degrees of sideslip) and was not objectionable. No roll cross-coupling characteristics were noted. The roll performance characteristics met the requirements of MIL-F-8785B(ASG). Within the scope of these tests, the roll performance characteristics are satisfactory.

Stall Characteristics

36. Dual and single-engine stall characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 6. 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 are presented in table 6.

37. Dual-engine stalls were evaluated with power OFF and with power for level flight. The initial stall warning was provided by the stall warning horn. A lift computer incorporated in the stall warning system provided a programmed constant stall warning margin with various flap settings. Initial stall warning was provided by the stall warning horn at approximately 10 knots above the stall airspeed. Additional warning was provided by a moderate buffet at approximately 2 knots in advance of the stall. Lateral control effectiveness remained good throughout the approach to the stall and no discernible nonlinear increase in elevator control force occurred prior to the stall. In general, all stalls were characterized by a slight pitch-up, followed immediately by a nose-down pitch attitude. Stalls conducted at maximum continuous power settings were characterized by an uncontrollable left roll to approximately a 35-degree roll attitude. Prompt recovery from all stalls was readily accomplished by relaxing elevator control force pressure and establishing a 15 to 20-degree nose-down pitch attitude (HQRS 3). During recovery, rates of descent in excess of 2000 ft/min were frequently achieved. Lateral control effectiveness returned immediately, using this recovery technique. Applying aft elevator control force prior to firmly establishing full recovery airspeed (approximately 10 to 15 knots above the stall airspeed) resulted in a secondary stall and additional altitude loss. The deep stall characteristics normally associated with T-tailed aircraft were never encountered during these tests. The dual-engine stall characteristics of this aircraft were generally mild.

38. Single-engine stall characteristics were evaluated with the critical engine (left engine) inoperative and MCP on the remaining engine. The stall warning, stall, and stall recovery were essentially the same as the dual-engine stall characteristics. A slight roll to the left accompanied the stall. Recovery was accomplished utilizing the technique described in paragraph 37, and could be accomplished more quickly by reducing power on the remaining engine immediately after the stall. The stall characteristics of the Model 200 aircraft met the requirements of FAR Part 23 and MIL-F-8785B(ASG). The programmed stall warning system provided an excellent stall warning margin for each of the various test configurations and is an enhancing design feature. Within the scope of these tests, the stall characteristics are satisfactory.

Single-Engine Characteristics

39. The single-engine characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 6. These tests were conducted by establishing trim configuration at the desired airspeed and simulating sudden engine failure by

selecting 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 are shown in table 6. Sudden engine failure (rudder boost OFF) resulted in rudder pedal forces of 145 pounds (propeller windmilling) and 130 pounds (propeller feathered) with MCP on the remaining engine. With the rudder boost ON, rudder pedal forces were reduced to 85 pounds (propeller windmilling) and 35 pounds (propeller feathered). With the rudder boost ON, the average pilot could easily maintain directional control following unexpected engine failure immediately after takeoff, due to the reduction of high rudder pedal forces. The rudder boost greatly reduced pilot workload during asymmetric power conditions (HQRS 2) and is an enhancing feature. Transient forces resulting from a sudden engine failure were not excessive. The pedal forces encountered could be trimmed out completely at all power settings with the rudder boost ON, greatly reducing pilot workload during single-engine operation (HQRS 3). With the rudder boost OFF, pedal forces could not be trimmed out for power settings within 10 percent of MCP; however, the remaining pedal force required minimal pilot compensation (HQRS 3).

40. The single-engine V_{MC} was evaluated during single-engine stall tests and during separate V_{MC} testing. The test aircraft always stalled prior to reaching single-engine V_{MC}. Lateral and directional control effectiveness remained throughout the approach to stall and returned immediately after performing the stall recovery techniques described in paragraph 37. The single-engine stall speeds shown in table 6 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

41. The ground handling characteristics of the Model 200 aircraft were evaluated throughout the conduct of these tests. In the normal mission configuration (aft cg), two people standing inside the aircraft in the vicinity of the swing-down door entrance caused the nose gear to lift off of the ground. The fin on the lower aft portion of the fuselage had been bent and cracked by the contractor during normal ground handling in the normal mission configuration (photo A). The swing-down door ground clearance was less than 1 inch in the normal mission configuration (photo B). In this configuration, the nose gear strut was fully extended (photo C), in contrast to a forward cg (photo D). The pitch attitude instability evident during loading and ground operations of the aircraft in the normal mission configuration is a deficiency. An EPR concerning this deficiency was submitted (ref 20, app A).

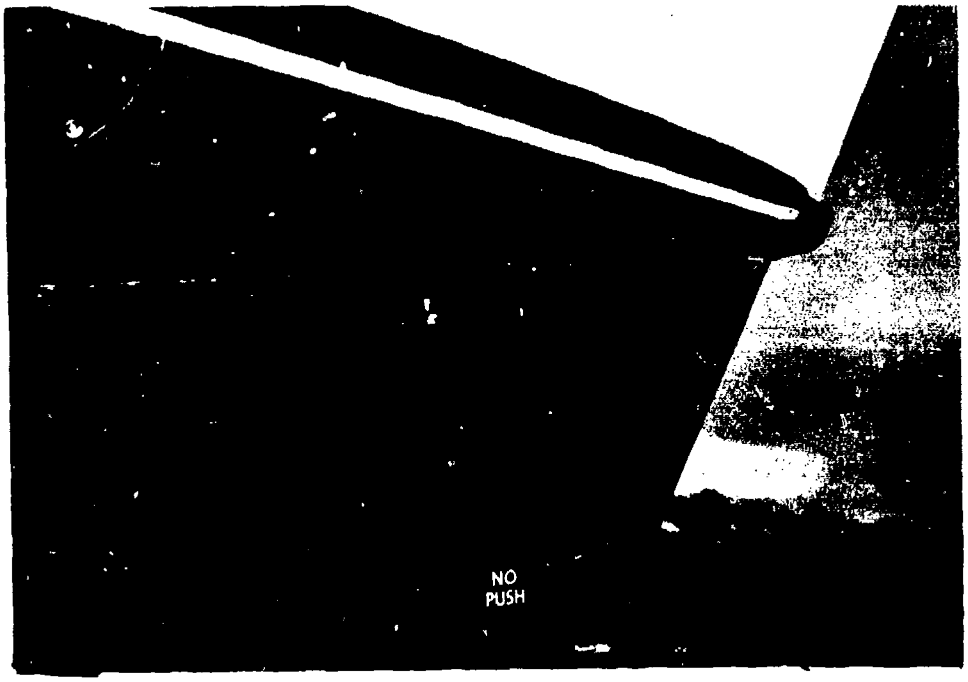

Photo A. Lower Aft Portion of Fuselage.

Photo B. Swing-Down Door.

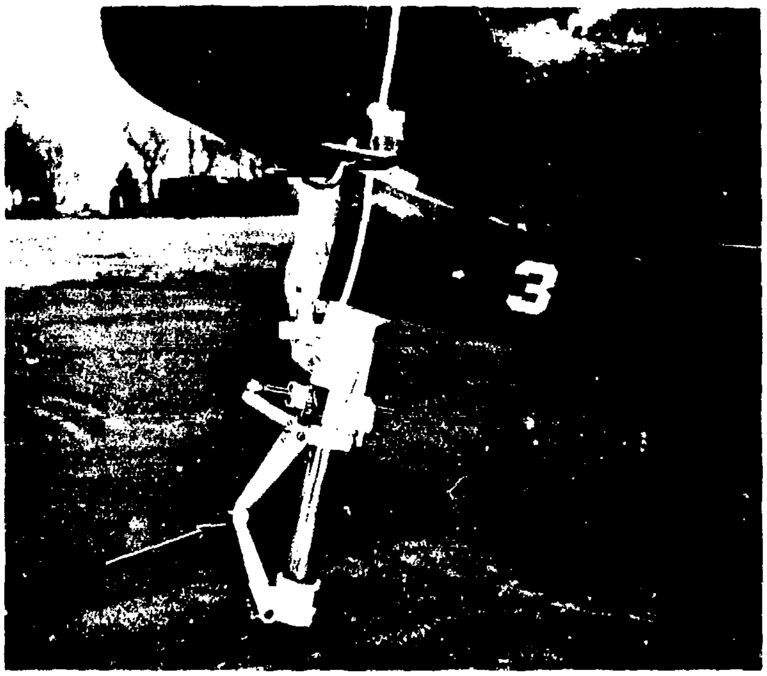

Photo C. Nose Gear Strut at Normal Mission Configuration.

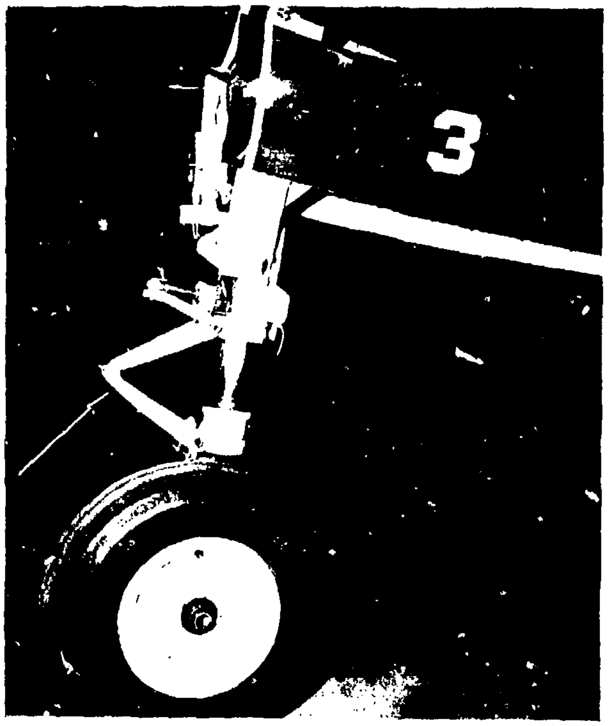

Photo D. Nose Gear Strut at Forward Center of Gravity.

42. An indicated 150 foot-pounds (ft-lb) of torque and 900 rpm propeller speed were required to initiate taxi during no-wind conditions. Nose wheel steering characteristics were good. Minimal pilot effort was required to maintain directional control during ground operations (HQRS 2). Use of the Beta range (propeller pitch setting) on the power lever control console allowed low taxi speeds and reduced braking requirements. Braking characteristics were excellent, with no fading or overheating. No difficulty was encountered when using reverse thrust to back up for short distances while in the aft cg configuration (this technique would not be used under normal circumstances, due to excessive propeller blade erosion). Field of view from the cockpit was good during all ground and taxi operations. Within the scope of these tests, the normal ground handling characteristics are acceptable.

43. The operation of CEFLY LANCER mission equipment requires power from an external auxiliary power unit (APU) prior to starting engines. After starting the engines, it is necessary to maintain 2000-rpm propeller speed to obtain the 400-hertz requirements from the 8.5-KVA AC generators, which provide power for the mission gear when the APU is disconnected. The mission gear cannot accept any intermittent power outages after being warmed up and power losses would be experienced at propeller speeds less than 2000 rpm. It is impossible to meet this requirement and still perform complete mandatory engine run-up checks. Attempts were made to perform normal ground and taxi operations and still maintain a continuous 2000 rpm propeller speed. Adequate performance was not attainable, using a variety of techniques, and maximum tolerable pilot compensation was required (HQRS 7). The requirement to maintain a continuous 2000 rpm propeller speed during ground operations is a deficiency. An EPR concerning this deficiency was submitted (ref 21, app A). The requirement to operate at these conditions should be reevaluated.

Takeoff and Landing Characteristics

44. The takeoff and landing characteristics of the Model 200 aircraft were evaluated on the 5000-foot hard-surface runway of the BAC facility at Wichita, Kansas at the conditions shown in table 1. Runway conditions were varied, and included wet, dry, ice, snow, and slush conditions. Takeoff characteristics were evaluated using normal techniques (gradual application of power until achieving takeoff power) and maximum performance techniques (set power at takeoff power and release brakes). The time required for engine acceleration from flight-idle to takeoff power was approximately 4 seconds. Due to the torque generated during maximum performance takeoff, the aircraft longitudinal axis was lined up approximately 5 degrees right of the runway center line as a technique to permit a smooth takeoff roll. The brakes held well during application of takeoff power and simultaneous brake release was accomplished easily. Aircraft torque brought the aircraft onto the center line and directional control was satisfactory through the rudder effective speed of 45 KIAS. Normal takeoff techniques were utilized for normal takeoff power applications. Rotation was accomplished without excessive aft elevator control pressure at approximately 85 KIAS and lift-off occurred at 95 KIAS. The gear was raised at 105 KIAS. Travel time was

5.3 seconds. The flaps were raised at 120 KIAS. Travel time was 3.0 seconds. Minor longitudinal trim changes (1 or 2 degrees, nose down from takeoff trim settings) were required during transition to initial climb conditions. Field of view during takeoff and initial climb was good. Placard limits for gear and flaps would be difficult to exceed during normal operating conditions. Takeoffs during wet and icy conditions were conducted with caution, but required minimal pilot compensation to achieve satisfactory performance (HQRS 3). Within the scope of these tests, the takeoff characteristics were satisfactory.

45. Landing characteristics were evaluated using normal landing techniques at the conditions shown in table 1. Power approaches to landing were performed, primarily due to the high gross weights used during these tests. Downwind airspeed was 140 KIAS, with transition to 120 KIAS while turning on the final approach. Touchdown airspeeds were approximately 80 KIAS with 100 percent flaps, 90 KIAS with 40 percent flaps, and 100 KIAS with no flaps. Braking characteristics were excellent, with no fading or overheating. Full reverse propeller pitch was utilized on several landings. Roll-out distances were reduced and no controllability problems were encountered. A single-engine go-around to a closed traffic pattern was executed on short final approach with no difficulty, after retracting the landing gear and flaps. Failure to retract the landing gear immediately will result in a negative rate of climb. A single-engine landing was accomplished and maximum reverse thrust on the operating engine was utilized during roll-out. There was no tendency to overcontrol, although considerable pilot effort was required to maintain proper brake application in response to the asymmetric reverse thrust (HQRS 5). The main landing gear braking characteristics were excellent and are an enhancing feature of this aircraft. Within the scope of these tests, the landing characteristics are satisfactory.

Trim Change Characteristics

46. Trim change characteristics were evaluated at the conditions shown in table 2. The aircraft was trimmed in steady-heading, ball-centered level flight at the desired initial trim conditions, then a configuration change was made while holding one or more of the initial trim parameters constant. Variations in power, flap position, and gear position utilized during the conduct of this test are specified in paragraph 23.145 of FAR Part 23 and in paragraph 3.8.6.1 of BAC Specification BS 22301 (ref 14, app A). The test results are presented in table 11. All control force variations with trim changes were light, ranging from 4 to 28 pounds. Only one force exceeded the specification limitations. In descending flight at 140 KCAS with gear down, 100 percent flaps, and power off, the rapid addition of maximum continuous power resulted in a nose-up pitch attitude which required a forward stick force of 13 pounds to maintain 140 KCAS. Although this control force exceeded the requirement of paragraph 3.8.6.1(5) of the BAC specification limit (10 pounds) by 3 pounds, it was not an excessive increase and required minimal pilot compensation to maintain the desired flight condition (HQRS 3). Within the scope of these tests, the trim change characteristics of the Model 200 aircraft are satisfactory.

Table 11. Trim Change Characteristics.

Flight Number	Flight Phase	Initial Trim Condition					Configuration Change	Parameter Held Constant	Control Force (lb)			
									Requirements		Test Results	
		Pressure Altitude (ft)	Calibrated Airspeed (kt)	Landing Gear	Flaps (%)	Power Setting			11,280 lb at 13.88% MAC	15,000 lb at 37.15% MAC	11,280 lb at 13.88% MAC	15,000 lb at 37.15% MAC
1	Approach	5000	140	Down	Zero	Off	Flaps down	Airspeed	25	10	17	9
2	Approach	5000	140	Down	100	Off	Flaps up	Airspeed	20	10	20	4
3	Approach	5000	140	Down	100	MCP	Flaps up	Airspeed	40	20	18	9
4	Approach	5000	140	Down	Zero	Off	Takeoff power	Airspeed	25	25	13	6
5	Approach	5000	140	Down	100	Off	Takeoff power	Airspeed	10	10	13	14
6	Approach	5000	140	Down	100	Off	Power	N/A	40	20	28	7
7	Approach	5000	140	Down	100	PLF	Flaps up	Altitude	25	20	17	17

Flight Number	Flight Phase	Pressure Altitude (ft)	Calibrated Airspeed (kt)	Landing Gear	Flaps (%)	Power Setting	Remarks
8	Cruise	15,000	214	Up	Zero	MCP	Maintains lateral and directional trim satisfactorily in level flight
9	Climb	5000	140	Up	40	MCP	Maintains longitudinal trim satisfactorily
10	Climb	5000	140	Up	Zero	MCP	Maintains longitudinal trim satisfactorily
11	Cruise	5000	170	Up	Zero	PLF	Maintains longitudinal trim within ±6 KIAS
12	Single-engine cruise	5000	140	Up	Zero	MCP	Maintains satisfactory longitudinal and directional trim at an angle of bank of less than 5°

Night Operations

47. The night operational capability of the Model 200 aircraft was evaluated briefly during night operations at the BAC facility. Cockpit lighting features were good. The pilot's fresh air ventilation window reflected the instrument panel and center console lights, but visibility was not reduced and the slight glare was not objectionable. Other lighting features included wing tip and tail strobe lights, wing ice lights, navigation lights, rotating beacons, landing lights, and taxi light. All switches were easily accessible and the systems functioned properly. The taxi light provided adequate illumination for ground operations and the landing lights provided good illumination for takeoff and landing operations. The landing lights are located adjacent to the taxi light on the nose gear (photo E) and no operating time limit is specified for ground operation. Within the scope of these tests, the night operational capability of the aircraft is satisfactory.

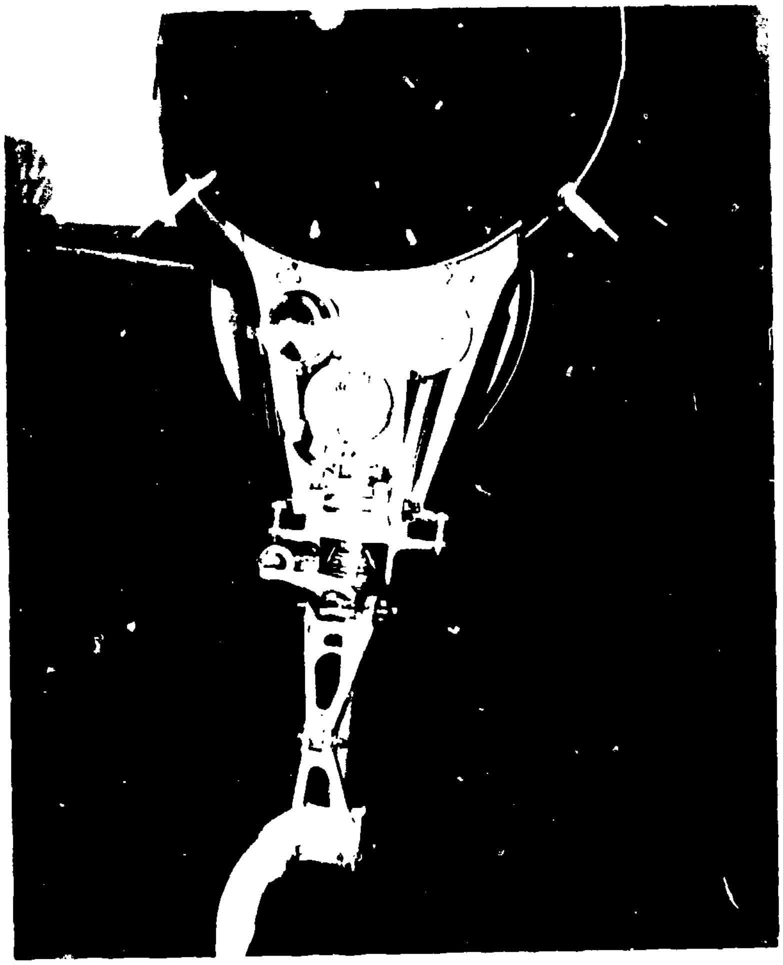

Photo E. Landing Lights.

Instrument Flight Capability

48. A limited evaluation of the instrument flight capability of the Model 200 aircraft was conducted at the end of several engineering flight test periods. A simulated instrument takeoff was conducted and minimal pilot effort was required to overcome takeoff power torque and maintain runway heading (HQRS 3). Holding patterns were flown at 140 and 170 KIAS under simulated instrument conditions. Standard rate turns were easy to perform. The aircraft would hold trim in smooth atmospheric conditions for periods of time long enough to allow the pilot to copy clearance changes or perform other cockpit duties. However, under conditions of light-to-moderate atmospheric turbulence, it was necessary to monitor trim closely (para 20). A practice precision radar approach was flown at McConnell Air Force Base, Kansas, using the contractor-recommended transition airspeeds (140 to 160 KIAS) and the contractor-recommended approach airspeed (120 KIAS). Flight path stability appeared to be good, and adjustments to heading and glide path were accomplished with minimal pilot effort (HQRS 3). Missed-approach procedures were accomplished satisfactorily.

49. During a normal approach and landing at the BAC facility, the power levers were retarded to the flight-idle position to make an altitude correction and the propeller speed dropped to 1900 rpm, resulting in the loss of the 8.5-KVA AC generators and the corresponding loss of the primary attitude and heading indicator gyros. This phenomenon was investigated at altitude and it was determined that at airspeeds of 125 KIAS or less (5 knots above the recommended approach airspeed of 120 KIAS), the AC generators would be lost if the power levers were retarded to flight-idle. In actual instrument conditions, the loss of the primary attitude and heading gyros could result in loss of aircraft control (HQRS 10). The loss of the primary attitude and heading gyros when the propeller speed is less than 2000 rpm is a deficiency. An EPR concerning the deficiency was submitted (ref 22, app A). The absence of a weather radar capability restricts this aircraft's ability to perform its all-weather mission in an environment where ground radar control advisory facilities are not available. Consideration should be given to installing a weather radar system in production aircraft to enhance the all-weather capability of this aircraft.

Aircraft Systems Failures

Yaw Damper:

50. Repeated malfunctioning of the yaw damper system occurred during the conduct of these tests. Divergent lateral-directional oscillations were encountered on several occasions with the yaw damper system ON. These objectionable oscillations were stopped by switching the yaw damper system OFF. On other occasions, steady 5-degree bank angle coordinated turns to the left resulted with the system ON. On the one flight when the yaw damper appeared to be operating properly, lateral-directional oscillations excited by 1-inch rudder control pulse inputs were damped after four overshoots. With the yaw damper system OFF, these same inputs resulted in 8 to 10 overshoots and long-term residual

lateral-directional oscillations. The lightly damped, easily excited residual lateral-directional oscillations which can result with the yaw damper OFF have already been described as a shortcoming in paragraph 30. 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. A reliable yaw damper should be installed.

Rudder Boost:

51. Rudder boost failures were evaluated during the conduct of single-engine tests. Sudden engine failures with the rudder boost OFF resulted in a rudder pedal force of 145 pounds (propeller windmilling) and 130 pounds (propeller feathered). If an engine were lost during takeoff with the rudder boost OFF, it would be necessary to overcome sudden high rudder pedal forces to maintain aircraft control. This sudden increase in rudder pedal force would require moderate pilot effort to maintain adequate control (HQRS 4). With the rudder boost OFF, pedal forces could not be trimmed out for power settings within 10 percent of MCP; however, the remaining pedal force required minimal pilot compensation for desired performance (HQRS 3).

Alternating Current Generator:

52. The failure of one AC generator was simulated in flight by switching off the left AC generator. 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 positions at airspeeds below 125 KIAS. This resulted in the immediate loss of electrical power to the primary attitude and heading gyros. The results of this loss have been stated in paragraph 49. This would also result in the loss of all power for the CEFLY LANCER mission gear. The loss of power for this mission gear, when the propeller speed is less than 2000 rpm, is a deficiency. An electrical power system should be installed on this aircraft to permit use of lower propeller speeds for taxiing, cruise economy, and reduced noise levels.

HUMAN FACTORS

Cockpit Evaluation

53. The cockpit area (flight deck) and ingress/egress areas were evaluated throughout the test program. Entrance to the aircraft is accomplished through a swing-down airstair door. A hydraulic damper allows the door to swing down slowly when opened. In the normal mission configuration (aft cg), the door-ground clearance was 1 inch. With one person on the step, this clearance was further reduced to 1/2 inch. An emergency exit door, placarded EXIT PULL, is located on the right cabin side wall just aft of the copilot seat (photo F). Approximately 10 inches of clearance between the pilot and copilot seats made entrance into the cockpit area awkward. The seat and flight control adjustment were adequate,

but 3 to 4 inches more aft seat travel would greatly facilitate ingress/egress. There is no capability to adjust each lap belt and keep the shoulder strap attachment point to the lap belts centered in the pilot's lap. The single lap belt adjustment provided is inadequate for use with shoulder straps and is a shortcoming. An EPR concerning this shortcoming was submitted (ref 23, app A). Seat comfort is adequate and the fresh air ventilation ducts were conveniently located and easily adjusted.

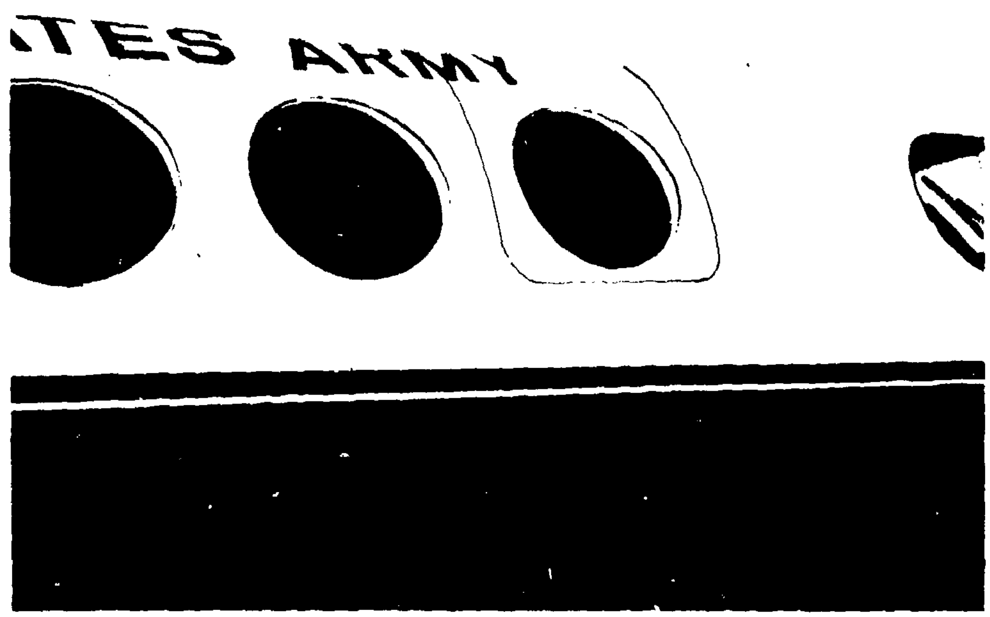

Photo F. Emergency Exit Door.

54. The test aircraft cockpit area is shown in photo G. The grouping and readability of the flight and engine instruments were adequate. The caution, advisory, and warning annunciator panels are well placed and easy to read. The location of the red WARNING and yellow CAUTION flasher lights at eye level, directly in front of the pilot and copilot on the glare shield, is a good design characteristic. The fuel panel arrangement is accessible and easy to read. The circuit breaker side panel is also accessible and easy to monitor. The overhead panel system controls, switches, and placards are easily accessible and their systematic arrangement allows utilization with little or no confusion. The pedestal-mounted propulsion system controls and associated controls are adequate, with good arrangement and accessibility. A feel identification feature is provided on the major

controls (power levers, propeller levers, condition levers, flaps, trim wheels, and friction adjustment). The propeller feather detent is 3/4 inch above the present markings. The improper marking of the propeller feather range on the control console is a shortcoming. An EPR concerning this shortcoming was submitted (ref 24, app A).

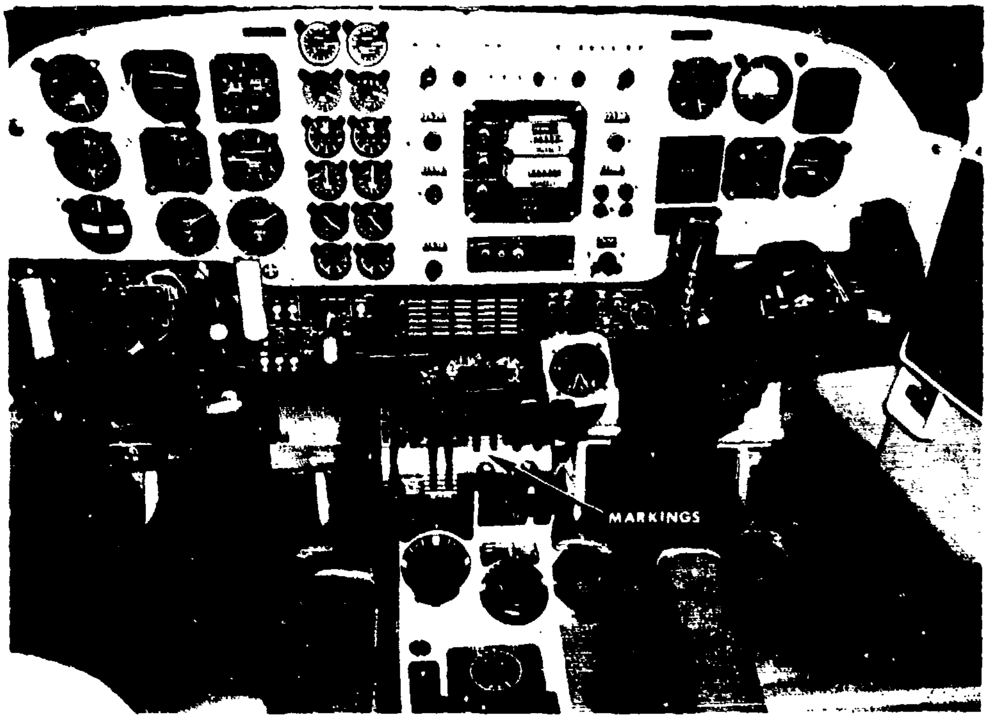

Photo G. Control Console.

55. The communications panel console was inadequate in several areas. There was no provision for a three-position push-to-talk communications switch for either the pilot or copilot in the test aircraft. The absence of a three-position push-to-talk switch is a shortcoming, and fails to meet the requirements of paragraph 3.12.4.3.4.1 of the Aircraft Procurement Specification (ref 4, app A). The communications panel and transmission selection procedures were nonstandard and confusing. On several occasions the copilot inadvertently transmitted to the pilot on intercom and blocked out tower transmissions to the pilot. The copilot was unable to monitor very-high-frequency (VHF) communications radios selected on his signal distribution panel while his transmit-select was in the intercom position, which is a shortcoming. During electrical extension of the ice vanes, the communications cord to the pilot control wheel prevented the ice vane handles from extending fully and is a shortcoming (photo H). Three EPR's concerning

these communications panel shortcomings were submitted (refs 25, 26, and 27, app A). An ultra-high-frequency (UHF) radio in lieu of one of the three installed VHF radios would increase flight communications flexibility and provide an alternate radio for use with military communications networks.

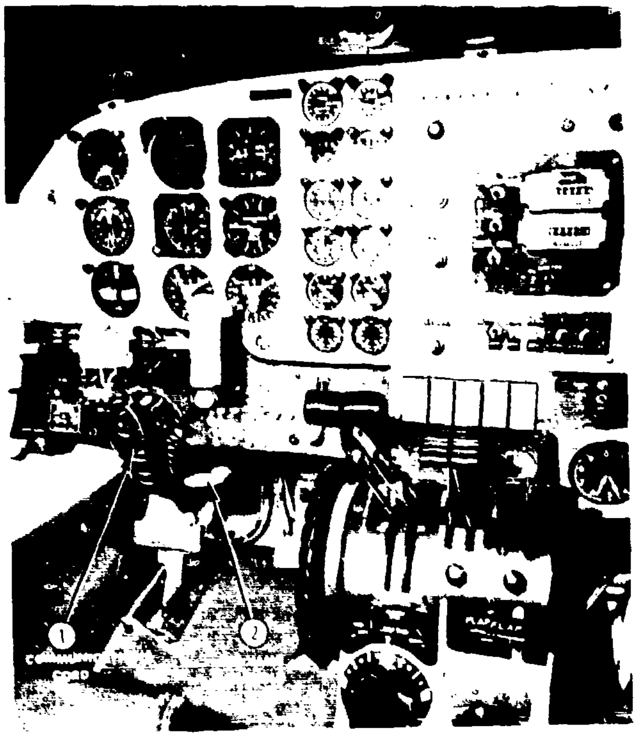

Photo 11. Instrument Panel, Pilot Side.

Noise

56. Noise in the cockpit area was evaluated qualitatively throughout the test program. Engine and propeller noise levels were considered to be excessive while operating at a continuous propeller speed of 2000 rpm. Noise levels were very uncomfortable after 2 hours of operation, even when wearing a properly fitted SPH-4 flying helmet. Noise levels at 1800 rpm propeller speed, the operating rpm of the commercial version Model 200 aircraft, were much lower, with no noticeable hearing discomfort. Further testing should be conducted to quantitatively measure and assess noise levels encountered during normal mission operations.

Toxicity

57. On every flight at altitudes above 15,000 feet Hp, a very light smoke filled the cabin and cockpit areas. Exposure times in excess of 30 minutes resulted in mild eye and throat irritation. Smoke in the cockpit and cabin areas is a deficiency. An EPR concerning this deficiency was submitted (ref 28, app A). Further testing should be conducted to determine the cause and the method of elimination of this unsatisfactory characteristic.

RELIABILITY AND MAINTAINABILITY

58. 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. This review was a limited, noninterference evaluation. Primarily, a qualitative evaluation was performed because the minimal number of program flight hours limited the opportunity to observe component repair and replacement. Formal removal or replacement tests were not performed. The aircraft was fully instrumented, a condition that resulted in maintenance complications which would not exist on an operational aircraft.

59. The items listed below are shortcomings which will affect the reliability and maintainability of the Model 200 aircraft. Equipment Performance Reports concerning these shortcomings were submitted (refs 29 through 37, app A).

a. The engine fuel transfer caution lights came on erroneously during flight.

b. The engine exhaust stacks cracked several times during testing (photo I).

c. The torque needles fluctuated ±25 ft-lb during flight.

d. The engine fire warning lights came on erroneously several times during flight (sunlight reflected inside the engine nacelle triggered one of the photoconductive cells).

e. The rudder boost pneumatic cylinder hoses to the flow control box connection points are identical, making it possible to connect these hoses backwards.

f. The installed yaw damper system malfunctioned repeatedly during the conduct of these tests.

g. Ground towing of the test aircraft in the normal mission configuration resulted in the bending of the nose gear steering idler link.

h. The propeller synchrophaser could not match propeller speeds at 2000 rpm (the primary governor stop limit), resulting in the requirement for manual adjustment of the propeller levers to obtain a synchronized rpm setting.

i. Engine exhaust gases used to continually 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 (photo J).

60. Normal maintenance procedure requires a 10-foot-high stand or ladder to perform the preflight and postflight inspections of the T-tail area. In addition, a tail stand was required to prevent the airplane from rocking back on its tail while performing ground maintenance operations during the normal mission configuration (aft cg) tests.

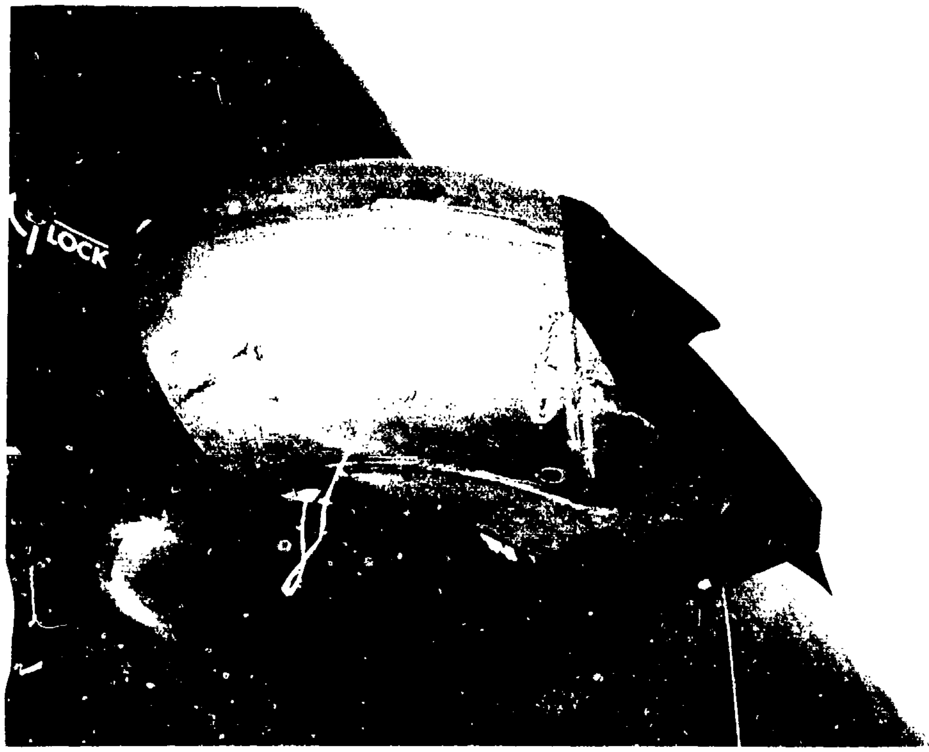

Photo I. Engine Exhaust Stack.

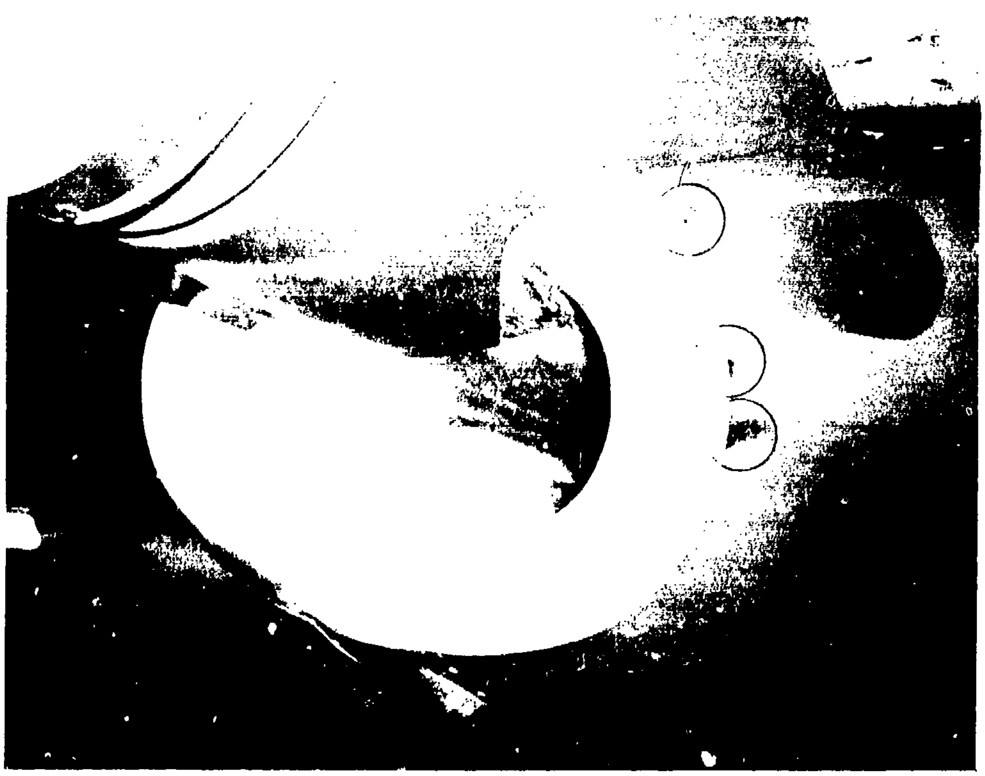

Photo J. Engine Inlet Cowling.

CONCLUSIONS

GENERAL

61. The following conclusions were reached upon completion of the BAC Model 200 aircraft APE I:

a. In the normal mission configuration at the military maximum gross weight of 15,000 pounds and an aft cg limit at FS 197.4, the Model 200 aircraft has inadequate single-engine performance and degraded longitudinal handling qualities.

b. The programmed stall warning system provided an excellent stall warning margin for each of the various test configurations and is an enhancing design feature (para 38).

c. The rudder boost greatly reduced pilot workload during asymmetric power conditions and is an enhancing feature (para 39).

d. The main landing gear braking characteristics were excellent and are an enhancing feature of this aircraft (para 45).

e. Four deficiencies and eighteen shortcomings were noted during these tests.

DEFICIENCIES AND SHORTCOMINGS

62. The following deficiencies were identified:

a. The pitch attitude instability evident during loading and ground operations of the aircraft in the normal mission configuration (aft cg) (para 41).

b. The requirement to maintain a continuous 2000-rpm propeller speed during ground operations (para 43).

c. The loss of the primary attitude and heading gyros when the propeller speed is less than 2000 rpm (para 49).

d. Smoke in the cockpit and cabin areas (para 57).

63. The following shortcomings were identified:

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

b. The long-term trimmability was poor (para 20).

c. The elevator control force versus airspeed gradient was reversed in the power approach configuration (para 22).

d. The Dutch-roll oscillations were lightly damped and easily excited (para 30).

e. The single lap belt adjustment provided is inadequate for use with shoulder straps (para 53).

f. The propeller feather range markings on the control console were improper (para 54).

g. There was no three-position push-to-talk switch (para 55).

h. The copilot was unable to monitor VHF communications channels selected on his signal distribution panel while his transmit-select was in the intercom position (para 55).

i. During electrical extension of the ice vanes, the communications cord to the pilot control wheel prevented the ice vane handles from extending fully (para 55).

j. The engine fuel transfer caution lights came on erroneously during flight (para 59a).

k. The engine exhaust stacks cracked several times during testing (para 59b).

l. The torque needles continually fluctuated ±25 ft-lb during flight (para 59c).

m. The engine fire warning lights came on intermittently several times during flight (sunlight reflected inside the engine nacelle triggered one of the photoconductive cells) (para 59d).

n. The rudder boost pneumatic cylinder hoses to the flow control box connection points are identical, making it possible to connect these hoses backwards (para 59e).

o. The installed yaw damper system malfunctioned repeatedly during the conduct of these tests (para 59f).

p. Ground towing of the test aircraft in the normal mission configuration resulted in the bending of the nose gear steering idler link (para 59g).

q. The propeller synchrophaser could not match propeller speeds at 2000 rpm (the primary governor stop limit), resulting in the requirement for manual adjustment of the propeller levers to obtain a synchronized rpm setting (para 59h).

r. Engine exhaust gases used to continually 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 covering (para 59i).

RECOMMENDATIONS

64. The deficiencies identified during this evaluation must be corrected (para 62).

65. The shortcomings should be corrected (para 62).

66. The aft cg limit of the normal mission configuration should be moved forward (para 22).

67. Consideration should be given to installing an autopilot system to reduce pilot workload (para 22).

68. A reliable yaw damper or an autopilot system which would reduce lateral-directional pilot workload should be installed (para 30).

69. Consideration should be given to installing a weather radar system in production aircraft to enhance the all-weather capability of the aircraft (para 49).

70. A variable-frequency AC generator should be installed on this aircraft to permit use of lower propeller speeds for taxiing, cruise economy, and reduced noise levels (para 52).

71. A UHF radio should be installed in lieu of one of the three installed VHF radios, as this would increase flight communications flexibility and provide an alternate radio for use with military communications networks (para 55).

72. Further testing should be conducted to quantitatively measure and assess noise levels encountered during normal mission operations (para 56).

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. Prime Item Development 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 Aircraft 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.

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

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

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

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

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

20. EPR, USAASTA, 74-21-1, 20 March 1974.

21. EPR, USAASTA, 74-21-2, 20 March 1974.

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

23. EPR, USAASTA, 74-21-5, 20 March 1974.

24. EPR, USAASTA, 74-21-6, 20 March 1974.

25. EPR, USAASTA, 74-21-7, 20 March 1974.

26. EPR, USAASTA, 74-21-8, 20 March 1974.

27. EPR, USAASTA, 74-21-9, 20 March 1974.

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

29. EPR, USAASTA, 74-21-10, 20 March 1974.

30. EPR, USAASTA, 74-21-11, 20 March 1974.

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

32. EPR, USAASTA, 74-21-13, 20 March 1974.

33. EPR, USAASTA, 74-21-14, 20 March 1974.

34. EPR, USAASTA, 74-21-15, 20 March 1974.

35. EPR, USAASTA, 74-21-16, 20 March 1974.

36. EPR, USAASTA, 74-21-17, 20 March 1974.

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

APPENDIX B. DESCRIPTION

GENERAL

1. The Model 200 aircraft has the general structure and space arrangements of the BAC Super KingAir Model 200 aircraft. Three views of the test aircraft are shown in photos B-1, B-2, and B-3. General specifications are listed below.

Dimensions

Wing span	54 ft, 6 in.
Horizontal stabilizer span	18 ft, 5 in.
Length	43 ft, 9 in.
Heigh' 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.

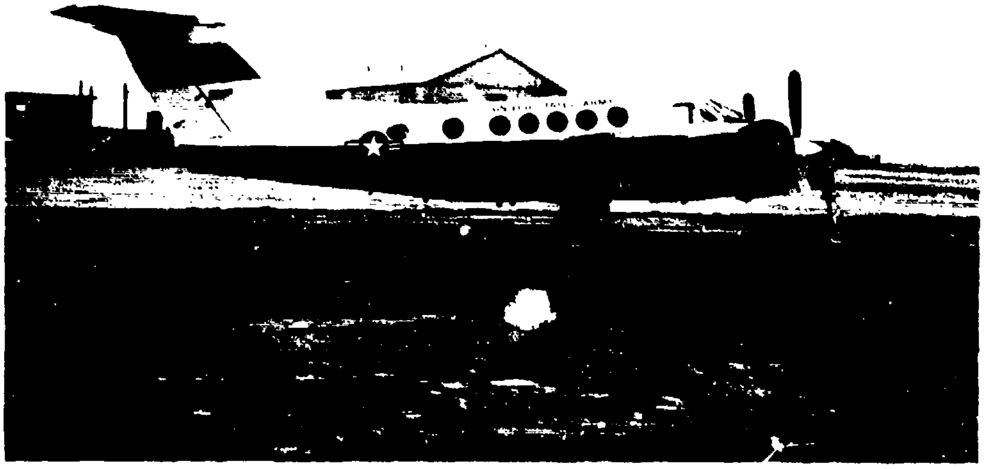


Photo B-1. Right Side View, Model 200 Aircraft.

Photo B-2. Front View, Model 200 Aircraft.

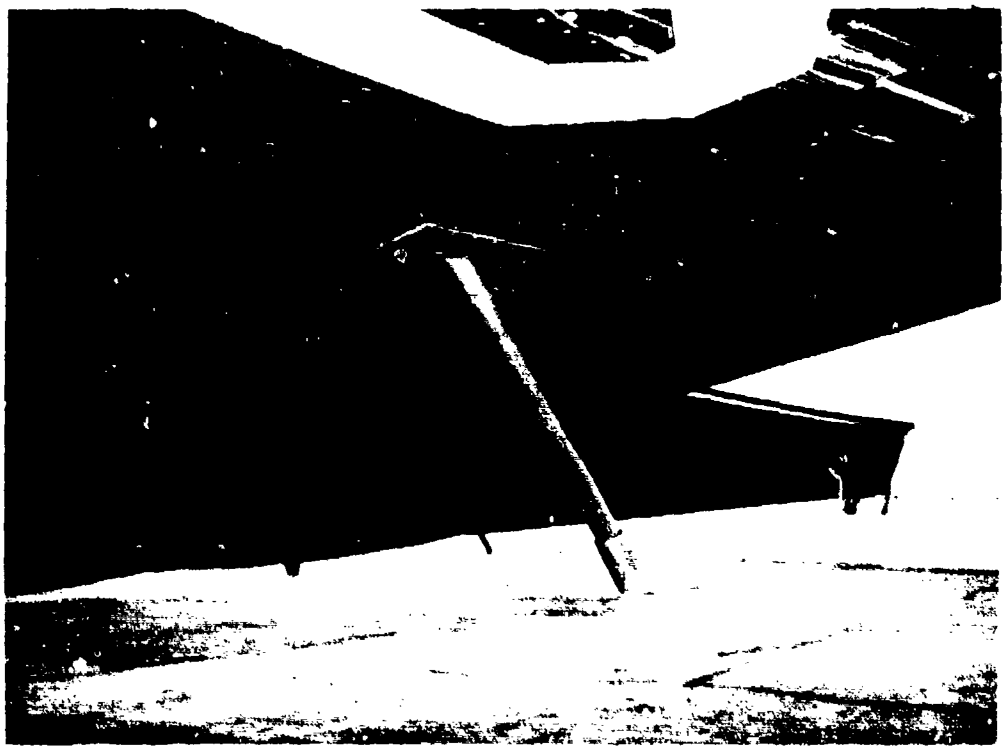


Photo B-3. Left Side Front Quarter View, Model 200 Aircraft.

FLIGHT CONTROL SYSTEM

2. The Model 200 aircraft is provided with conventional dual controls for the pilot and copilot. The flight control system is reversible. The elevator and rudder control surfaces are of conventional design. The aileron control surface has a 28 inch x 1-1/2 inch metal sandwich added to the trailing edge adjacent to the trim tab to aid lateral control effectiveness (photo B-4). The elevators and ailerons are operated by conventional control wheels interconnected by a T-column. The rudder pedals are interconnected by a linkage below the floor. These systems are connected to the control surfaces through closed cable bell crank systems. Rudder, elevator, and aileron trim are adjustable with controls mounted on the center pedestal. Position indicators for each of the trim tabs are integrated with their respective controls. An elevator bob-weight and downspring has been incorporated to lighten longitudinal control forces in flight. A control lock is provided which permits positive locking of the control column, rudder pedals and the engine power controls.

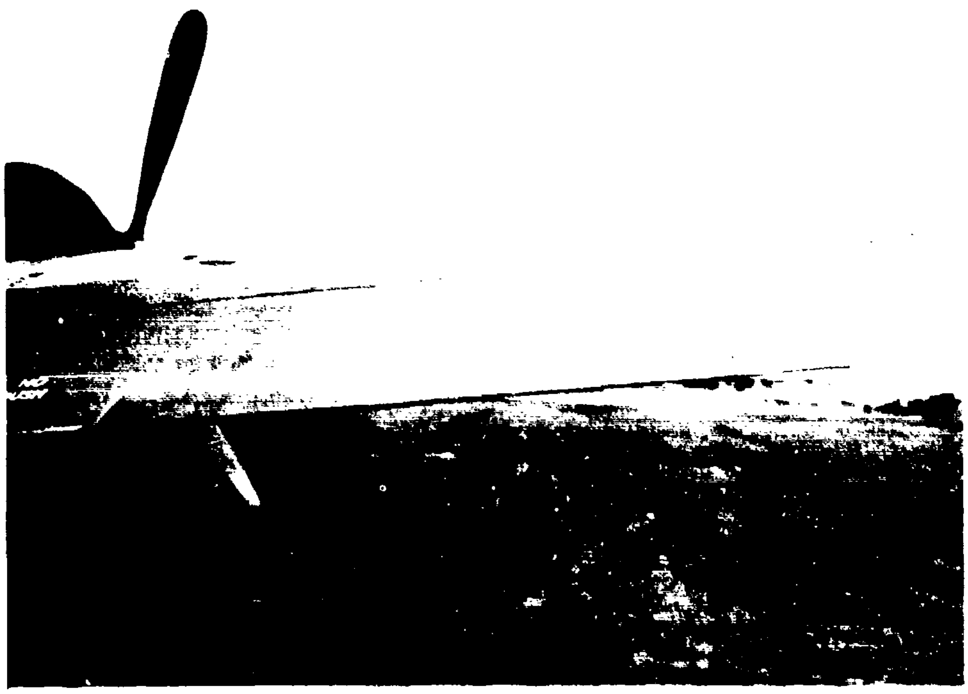


Photo B-4. Right Aileron Control Surface Modification.

3. A rudder boost system is provided to assist in maintaining directional control during asymmetrical thrust conditions, such as engine failure or a large variation of power between the engines. Incorporated in the rudder cable system are two pneumatic rudder boosting servos that actuate the cables to provide rudder pressure to help compensate for asymmetrical thrust. The system is operated by sensing

differential pressure between each of the engine bleed air systems. The system is operated by a toggle switch located on the pedestal below the rudder trim wheel. A functional check of the system may be obtained during the conduct of normal engine run-up procedures.

4. A yaw damper system is provided to assist in maintaining directional stability. The system components include a yaw sensor, amplifier, and control valve. Regulated air pressure from the control valve is directed to the same pneumatic servos used for the rudder boost system. The system is controlled by a toggle switch adjacent to the rudder boost switch on the pedestal. The circuit of the yaw damping system is interrupted by the landing gear safety switch while the airplane is on the ground and will not operate in this condition. The system may be used at any altitude and is required for flight above 17,000 feet.

ELECTRICAL SYSTEM

5. The aircraft electrical system is a 28 VDC single conductor system with the negative lead of each power source grounded to the main aircraft structure. DC electrical power is provided by one 34 ampere-hour, air-cooled, 20-cell nickel-cadmium battery and two 300-ampere starter/generators connected in parallel. Two three-phase, 8.5-KVA generators provide AC power to the aircraft and special equipment busses. These generators are housed in blisters, one on each engine nacelle (photos B-5, B-6, and B-7). The system is capable of supplying power to all subsystems that are necessary for normal operation of the airplane. A hot battery bus is provided for emergency operation of certain essential equipment and the cabin entry threshold light. Power to the main bus from the battery is through the battery relay, controlled by a switch placarded BAT-ON-OFF, located on the left subpanel. Power to the main bus system from the DC generators is controlled by a generator control panel which includes the following features: generator voltage regulator, generator paralleling, generator line contractor control, generator overvoltage protection, generator feeder ground fault protection, and reverse current protection. The DC generators are controlled by switches, placarded L DC GEN and R DC GEN, located on the left subpanel. Each AC generator is controlled by a generator control panel located beneath the floor aft of the main spar. Each control panel includes a voltage regulator and current transformer which regulates the voltage output of the AC generator, and a power monitor that isolates the AC generator whenever voltage is not within 95/117 volts AC and the frequency is not between 375/425 Hz. The AC generators are controlled by switches, placarded L AC GEN and R AC GEN, located on the left subpanel.

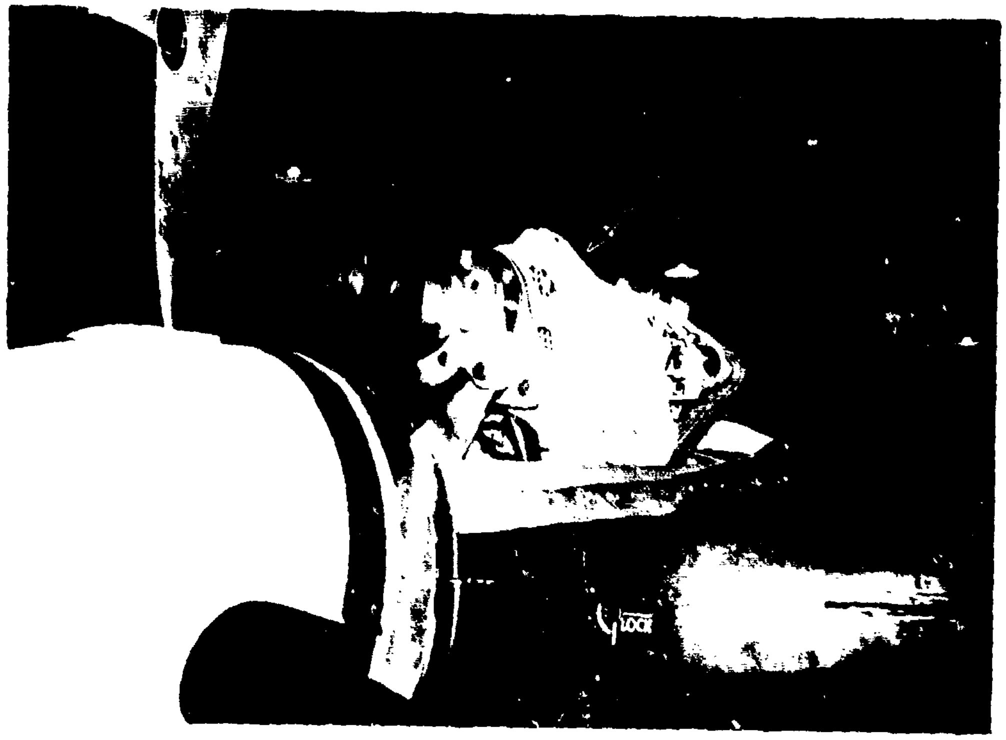


Photo B-5. AC Generator (cover removed).

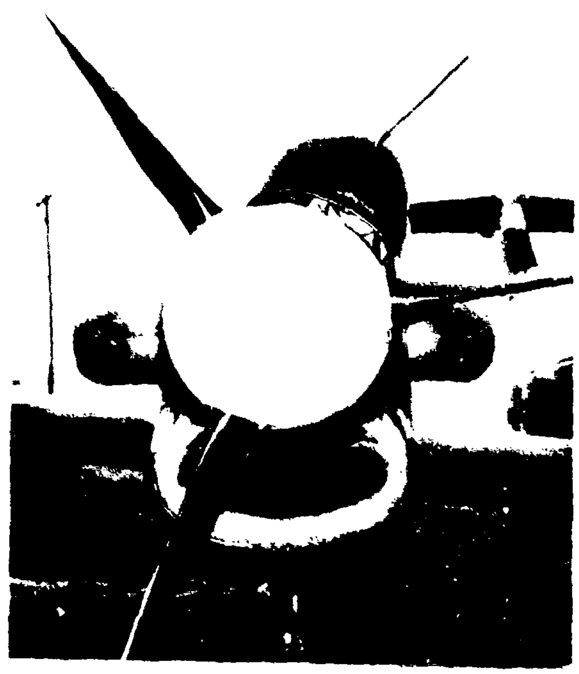


Photo B-6. AC Generator, Front View.

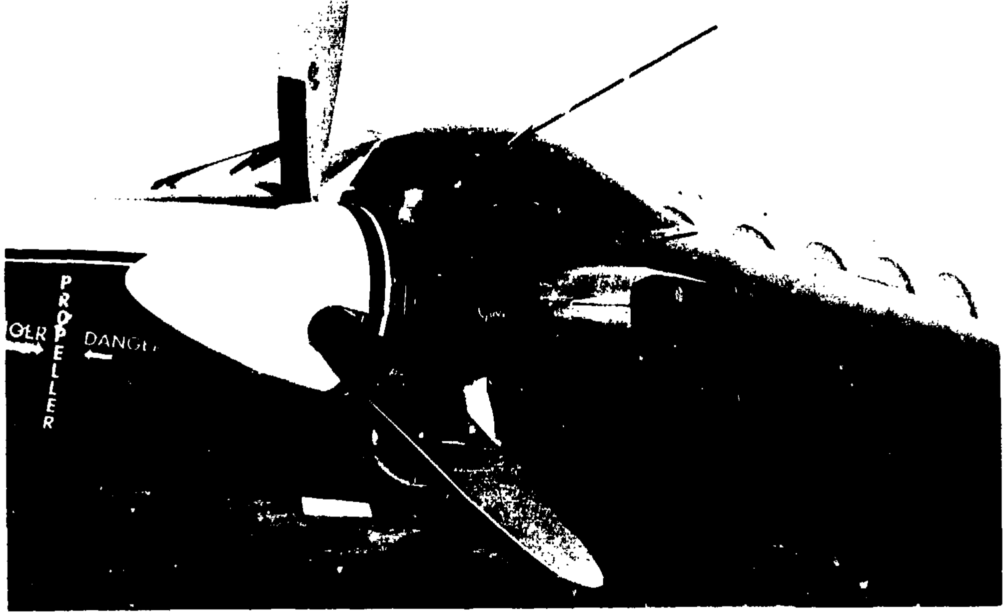


Photo B-7. AC Generator, Side View.

ENVIRONMENTAL SYSTEM

6. The environmental system consists of the bleed air pressurization, heating and cooling systems, and their associated controls. The cabin pressure vessel is designed for a normal working pressure differential of 6.0 psi, which will provide a cabin pressure altitude of 3870 feet at an airplane altitude of 20,000 feet. It will provide a nominal cabin altitude of 9840 feet at an airplane altitude of 31,000 feet. A mixture of bleed air from the engines and ambient air is available for cabin pressurization at a rate of approximately 10 to 15 pounds per minute. This air mixture also passes through a heating flow control unit in each nacelle and is ducted into the cabin to provide heating. An air-to-air heat exchanger helps regulate the temperature of the bleed air. Cabin air conditioning is provided by a refrigerant gas vapor-cycle refrigeration system consisting of a belt-driven engine-mounted compressor installed in the right engine (photo B-8). An environmental control section on the copilot subpanel provides for automatic or manual control of the environmental system. Pressurization controls are located on the pedestal and consist of a pressure setting rheostat and a cabin pressure/dump toggle switch.

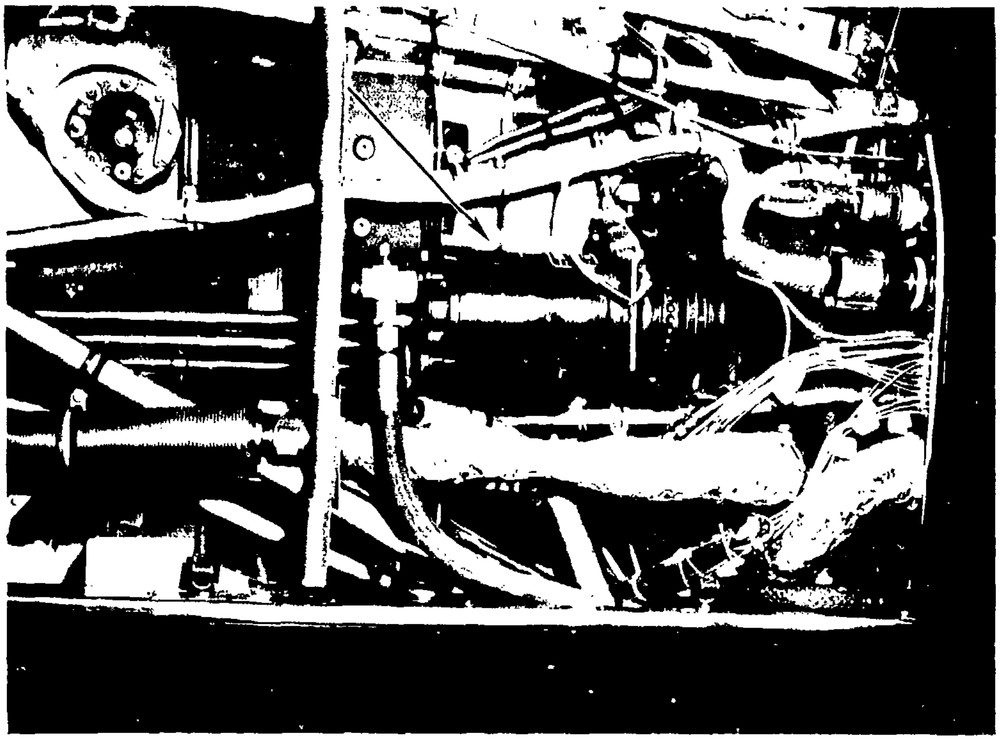


Photo B-8. Air Conditioning Compressor.

PROPULSION SYSTEM

7. The PT6A-41 engine, manufactured by United Aircraft of Canada, Ltd, has a three-stage axial, single-stage centrifugal compressor, driven by a single-stage reaction turbine. The power turbine, counterrotating with the compressor turbine, drives the output shaft. These engines produce 850 shaft horsepower each. Maximum continuous speed of the engine is 38,100 rpm, which equals 101.5 percent N_1. Prior to gear reduction, the turbine speed on the power side of the engine is 30,000 rpm at 2000 rpm propeller speed. The two engines installed on the test aircraft were experimental prototype engines (left engine serial number X70014, right engine serial number X70011).

8. The Hartzell propeller is of the full-feathering, constant-speed, counterweighted reversing type, controlled by engine oil pressure through single-action, engine-driven propeller governors. The propeller is three-bladed and flange-mounted to the engine shaft. Centrifugal counterweights, assisted by a feathering spring, move the blades toward the low rpm (high pitch) position and into the feathered position. Governor boosted engine oil pressure moves the propeller to the high rpm (low pitch) hydraulic stop and reversing position. The propellers have no low rpm (high pitch) stops; this allows the blades to feather after engine shutdown.

9. The propulsion system is operated by three sets of controls: the power levers, propeller levers, and condition levers (photo B-9). The power levers provide control of engine power from idle through takeoff power by operation of the gas generator (N_1) governor in the fuel control unit. When the power levers are lifted over the idle detent they control engine power through the beta and reverse ranges. The propeller levers are operated conventionally and control the constant-speed propellers through the primary governor. Normal operating range is 1600 to 2000 rpm. However, the propeller levers must remain full forward (2000 rpm) throughout all flight conditions on the CEFLY LANCER aircraft due to present configuration electrical power supply requirements (2000 rpm are required to operate the direct-drive 400-cycle 8.5-KVA generators). The condition levers control the flow of fuel at the fuel control outlet and select fuel cutoff, low-idle (52 percent N_1), and high-idle (70 percent N_1) functions.

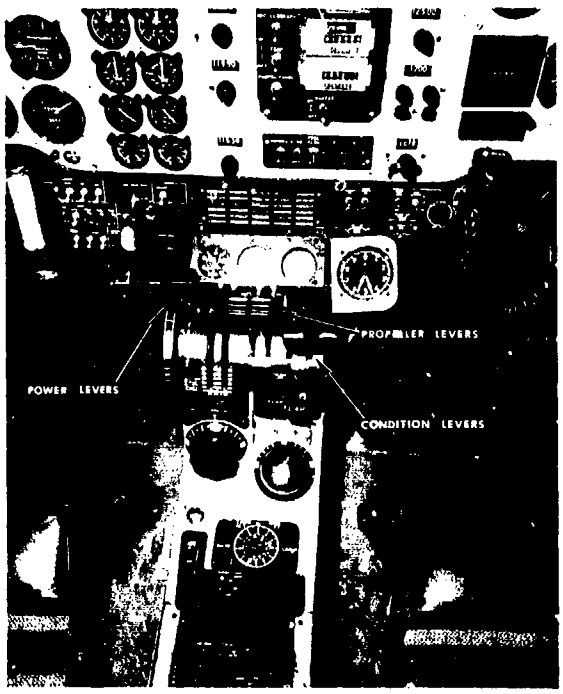


Photo B-9. Propulsion Control Console.

FUEL SYSTEM

10. The fuel system consists of two separate systems connected by a valve-controlled cross-feed line. The separate fuel system for each engine is further divided into a main and an auxiliary fuel system. The main system consists of a nacelle tank, two wing leading-edge tanks, two box-section bladder tanks, and an integral (wet cell) tank, all interconnected to flow into the nacelle tank by gravity. This system of tanks is filled from the filler located near the wing tip.

The auxiliary fuel system consists of a center section tank with its own filler opening and an automatic fuel transfer system to transfer the fuel into the main fuel system. When the auxiliary tanks are filled, they will be used first. During transfer of auxiliary fuel, which is automatically controlled, the nacelle tanks are maintained full. A swing check valve in the gravity feed line from the outboard wing prevents reverse fuel flow. Normal gravity transfer of the main wing fuel into the nacelle tanks will begin when the auxiliary fuel is exhausted. The two systems are vented through a recessed ram vent coupled to a protruding heated ram vent on the underside of the wing adjacent to the nacelle.

11. A fuel dump system is provided on the CEFLY LANCER aircraft (photo B-10). A guarded toggle switch on the fuel control panel activates the dump system. A check valve is opened and fuel is dumped from each of the separate fuel systems, utilizing gravity feed. Approximately 1500 pounds of fuel can be jettisoned during a 10-minute period. The dump may be terminated at the pilot's discretion, using the dump toggle switch.

Photo B-10. Fuel Dump Outlet.

MISCELLANEOUS SYSTEMS

Landing Gear

12. The tricycle high-flotation landing gear is retracted and extended by a 28-volt split field motor. A close-up view of the dual main gear wheels is shown in photo B-11. The gear motor is controlled by a switch located on the pilot subpanel, which must be pulled out of a detent to initiate retraction or extension. In the retracted position, the nose gear is fully enclosed and sealed by the nose gear doors. The main gear is partially exposed, with 8 inches of the dual main gear wheels exposed through a cut-out portion of the main gear doors. Manual extension may be accomplished in the event of a failure of the electrically operated system. The dual main gear tires and nose wheel tire are identical in size (6.50 x 10), but the main gear tires are 6-ply and the nose gear tire is 4-ply. Dual hydraulic brakes are operated by depressing the toe portion of either the pilot or copilot rudder pedals. Shuttle valves permit braking by either pilot or copilot.

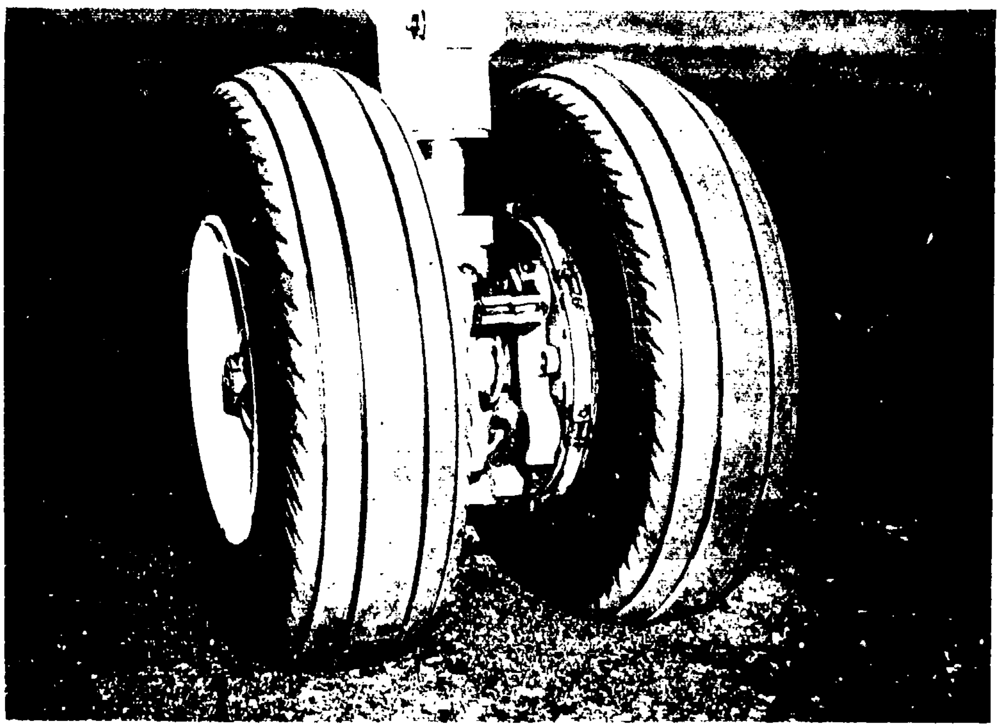


Photo B-11. Main Gear Wheels.

Annunciator System

13. The annunciator system consists of a warning annunciator panel (with red readout) centrally located in the glare shield and a caution/advisory annunciator panel (CAUTION yellow, ADVISORY green) located on the center subpanel.

Adjacent to the warning annunciator panel on the glare shield is a press-to-test switch to test the lights, and the pilot and copilot red WARNING and yellow CAUTION flashers. Individual function lights are of the word readout type. In the event of a fault, a signal is generated and applied to the respective channel in the appropriate annunciator panel. If the fault requires the immediate attention of the pilot, the fault warning lights on the glare shield will flash. The flashing fault warning lights may be extinguished by pressing the face of the light to reset the circuit. The illuminated fault indication on the warning annunciator panel will remain on if the fault is not, or cannot be, corrected. If an additional fault occurs, the appropriate light on the annunciator panel will illuminate and the warning flashing light will again illuminate.

Fire Detection System

14. A fire detection system is installed to provide immediate warning in the event of fire at the engine compartment. The system consists of three photoconductive cells in each engine nacelle, a control amplifier on a panel on the forward pressure bulkhead, two annunciator warning lights, placarded FIRE L ENG and FIRE R ENG, two fire extinguisher control switches, with lenses placarded L ENG FIRE - PUSH TO EXT, R ENG FIRE - PUSH TO EXT, located on the glare shield, a test switch on the copilot subpanel, and a circuit breaker placarded FIRE DET, on the right sidewall. Photoconductive cells, sensitive to infrared rays, are used as flame detectors. These cells are positioned in the engine compartments to receive direct and reflected rays, thus covering the entire compartment with three cells. Heat level and rate of heat rise are not factors in the sensing method. The cell emits an electrical signal proportional to the infrared intensity and ratio of the radiation striking the cell. To prevent stray light rays from signaling a false alarm, the control amplifier activates only when the signal reaches a preset alarm level, which illuminates the appropriate warning light in the warning annunciator panel. When the fire has been extinguished, the cell output voltage drops below the alarm level and the control amplifier resets. No manual resetting is required to reactivate the detection system.

Emergency Locator Transmitter

15. An emergency locator transmitter is installed on the right rear side of the fuselage to provide automatic emergency homing signals in the event of a crash or forced landing (photo B-12). The transmitter has a three-position toggle switch, ARM-ON-OFF, and the normal operation mode is the ARM position.

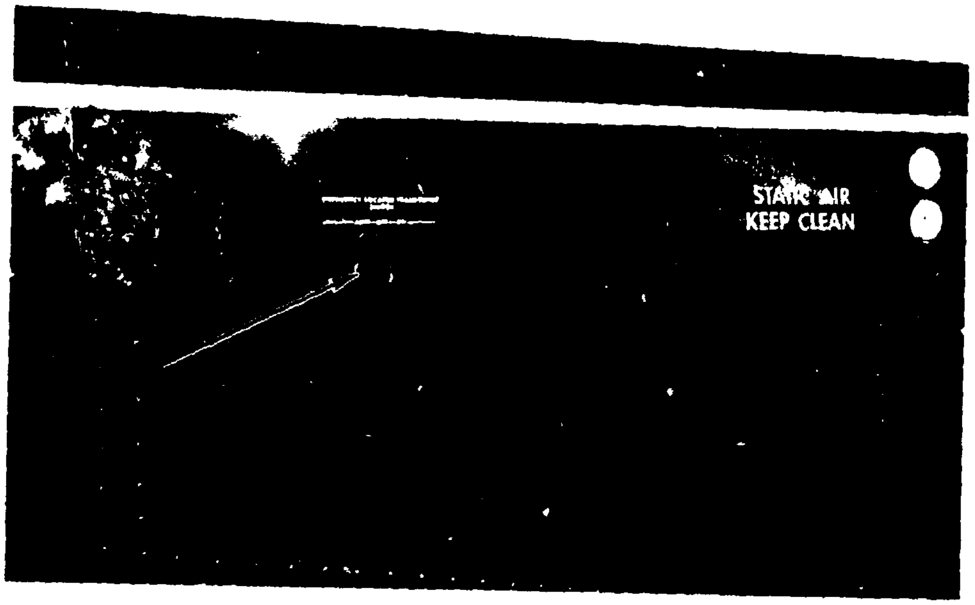


Photo B-12. Emergency Locator Transmitter Switch.

APPENDIX C. INSTRUMENTATION

1. The instrumentation in the BAC Model 200 aircraft, serial number 71-21058, was installed, calibrated, and maintained by BAC personnel. In addition to the instrumentation listed, the aircraft was equipped with a pitot-static boom which incorporated angle-of-attack and angle-of-sideslip vanes. Photos C-1, C-2, and C-3 show the instrument panel, photopanel, and oscillograph, respectively.

2. The left airspeed boom was a nonswivel type manufactured to Air Force Flight Test Center Drawing No. 59EDD261, Model No. 50-260. The right airspeed boom was a swivel type designed and manufactured by BAC. Photos C-4 and C-5 show the left and right airspeed booms. A list of test instrumentation showing the manufacturer, calibration range, and estimated parameter accuracies follows.

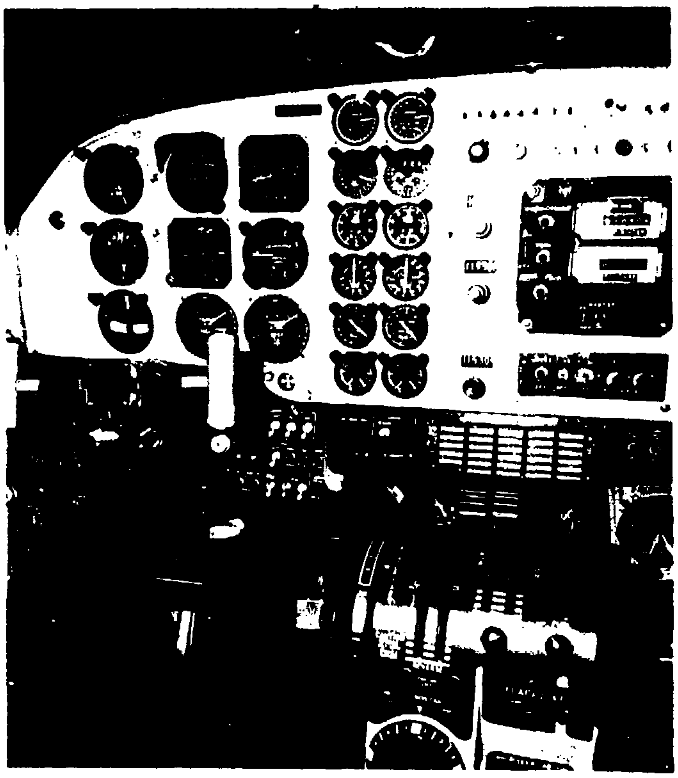


Photo C-1. Pilot Instrument Panel.

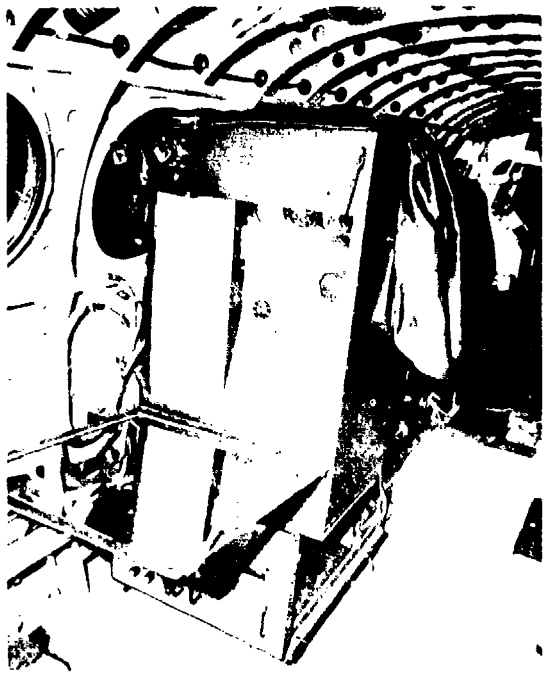


Photo C-2. Photopanel

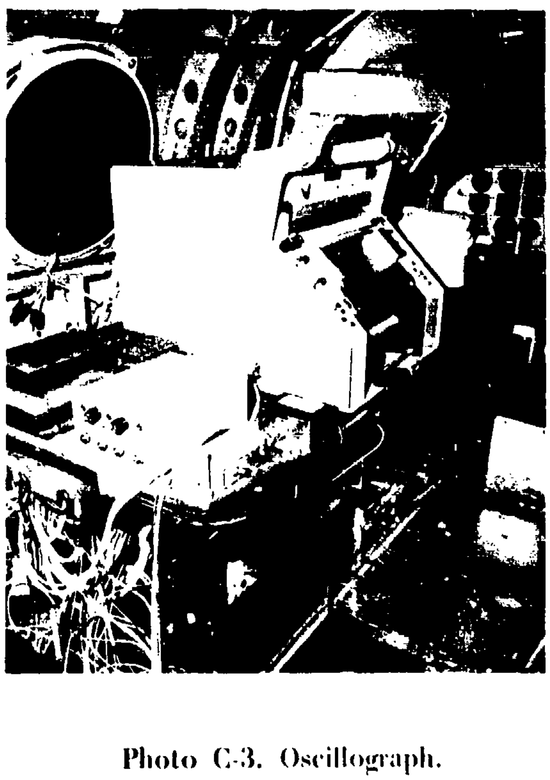


Photo C-3. Oscillograph.

Photo C-4. Airspeed Boom, Left Side.

Photo C-5. Airspeed Boom, Right Side.

Parameter	Manufacturer	Photopanel 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	522[illegible]	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
		Oscillograph		
Airspeed (left boom)	Statham	–	Zero to 350 kt	±1 kt
Altitude (left boom)	Statham	–	Zero to 35,000 ft	±20 ft
Rudder control surface position	Humphrey	–	26°, right, to 25.75°, left	±0.2°
Aileron control surface position	Humphrey	–	24.03° left aileron, up, to 17.06° left aileron, down	±0.2°
Elevator control surface position	Humphrey	–	19.68°, up, to 15.06°, down	±0.2°

Rudder control force	Beech	–	200 lb, left, to 200 lb, right	±1/2 lb
Aileron control force	Beech	–	100 lb, left, to 100 lb, right	±1/2 lb
Elevator control force	Beech	–	100 lb, push, to 100 lb, pull	±1/2 lb
Angle of attack	Buffalo/Statham	–	35°, nose down, to 30°, nose up	±1/4°
Angle of sideslip	Buffalo/Statham	–	35°, left, to 35°, right	±1/4°
Center-of-gravity normal acceleration	Statham	–	-5 to 5g	0.5g
Pitch attitude	Honeywell	10	80°, nose down, to 70°, nose up	±1/2°
Roll attitude	Honeywell	10	70°, left, to 70°, right	±1/2°
Yaw attitude	Electronic Specialty	5026	90°, left, to 90°, right	±1/2°
Pitch rate	Northrop	3	±30°/sec	±1°/sec
Roll rate	Northrop	3	±100°/sec	±1°/sec
Yaw rate	Northrop	3	±50°/sec	±1°/sec

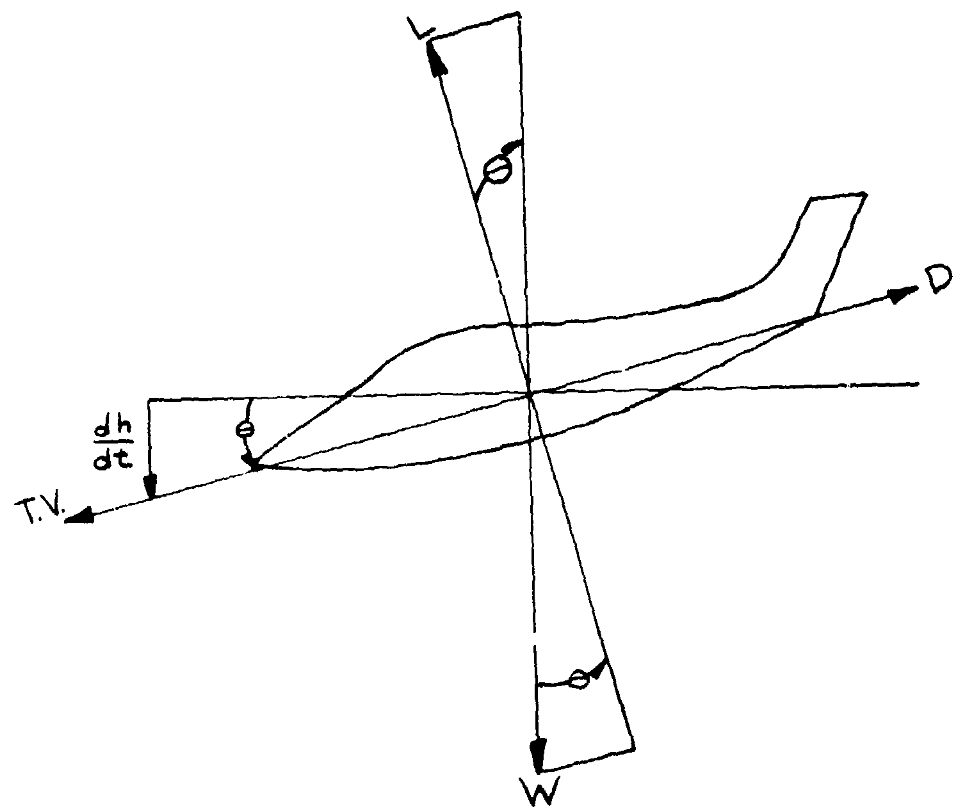
APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. This appendix contains some of the data reduction and analysis methods used to evaluate the BAC Model 200 CEFLY LANCER aircraft. The topics discussed include level flight, glide and climb performance, and dynamic stability. Past programs generally developed a drag polar relationship for specific flight conditions. However, the test points showed large deviation from the faired line at extreme altitudes (low versus high). The deviations were 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 develop the base line drag polar for the BAC Model 200 CEFLY LANCER aircraft. Level flight performance tests were conducted using the constant pressure altitude method and the sawtooth climb method was used for climb performance. All test data were converted into nondimensional coefficients which were used to develop the base line drag polar and the final generalized equations. The equations were then used to predict aircraft performance data at conditions not specifically tested.

Performance

2. The propeller feathered glide method 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_{MO}) (maximum operating airspeed) was investigated for a target altitude band. The following technique was used to develop the base line drag coefficient equation.



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

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

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

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

Where:

L = Lift force (lb)

W = Aircraft gross weight (lb)

θ = 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.

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

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

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

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

Where:

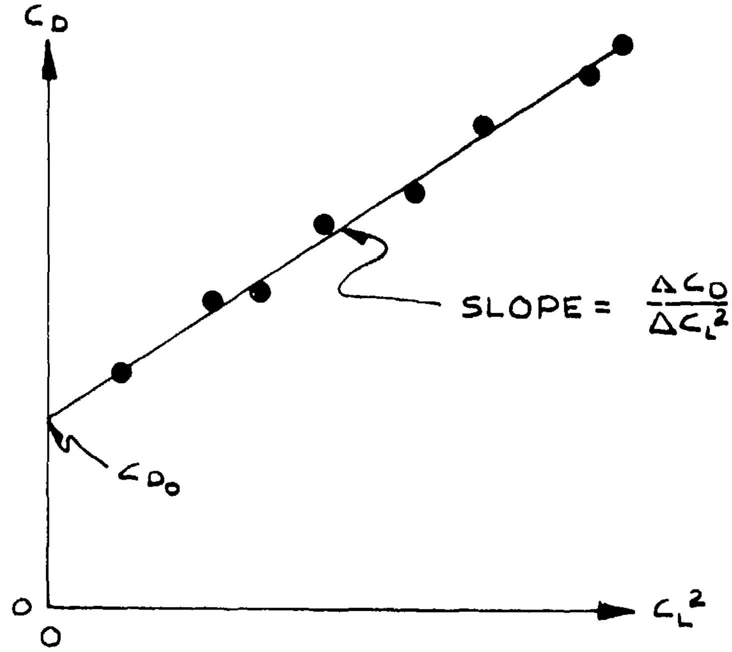
C_D = Coefficient of drag

$q = 1/2\ \rho\ V^2$ (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.



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

3. 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:

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

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

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

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

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

η_p = Propeller efficiency

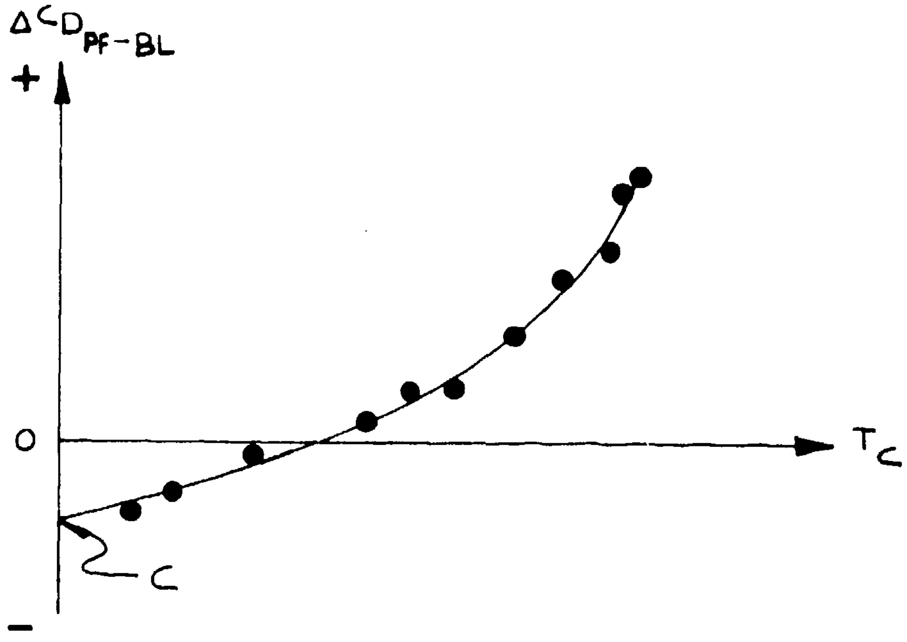
SHP = Shaft horsepower

F_n = Jet thrust (lb)

Q = Engine torque (ft-lb)

N_P = 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.

4. 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.

5. Climb performance tests (single and dual-engine) were conducted using the sawtooth climb method. 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 propeller feathered, while the right engine was operating at MCP. The aircraft was stabilized and trimmed at incremental airspeeds from 1.1 to $1.8V_S$ for ±1000 feet of the target altitude. The tape-line rate of climb and coefficients of drag, lift, and thrust were obtained from the recorded test data to determine the coefficients for the generalized equation.

6. The shaft horsepower available, fuel-flow rate, and net thrust of a PT6A-41 specification engine, including all installation losses, are presented in figures 52 through 55, appendix E. Figures 56 and 57 present the engine inlet pressure recovery data which were furnished by the airframe manufacturer. The UACL-furnished computer deck was used to calculate the performance for an installed specification engine. The computer deck 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, serial numbers X70014 and X70011, 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 chart was furnished by BAC and is presented in table D-1.

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}).

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

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	.689	.674	.652	.588				
.05	.569	.662	.715	.758	.777	.781	.784	.784	.778	.756	.723	.700	.681	.598				
.06	.590	.650	.725	.766	.794	.809	.814	.816	.812	.810	.786	.758	.701	.627				
07	.525	.635	.710	763	794	.821	.824	.834	.831	.831	.824	.809	.759	699				
.08	.503	.615	.698	.753	.792	.823	.832	.844	.847	848	.843	.835	.807	.749				
.09	481	.598	.681	740	784	.820	.831	.845	.852	.853	.852	.848	826	.786				
.10	.461	.574	666	.727	777	.810	.827	.842	.853	.857	859	.859	.841	.817	.719			
.12	.417	.534	.629	.698	.750	787	.814	.833	.847	.857	.862	.864	.858	.843	.786	.726		
14	.383	.498	588	.665	.723	.770	.797	.821	.837	.850	.859	.864	.869	.855	.818	.779	.729	
.16	.353	.461	.554	636	.699	746	.780	.806	.827	.842	.852	.858	.867	.863	.839	.811	.771	.726
18	.324	429	.519	.602	.670	.722	762	.790	.812	.830	.844	.853	.864	.865	.849	.831	.800	.760
20	299	.373	487	.570	.640	.668	.740	774	799	.820	.835	.845	.859	.865	.855	.843	.820	783
30	.204	.275	.343	411	.489	552	.612	.667	710	.743	.773	.792	.824	.857	.858	.853	.850	.840
.40											.697	.728	.777	.809	.840	.845	.848	.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

$\sigma \sim \rho/\rho_0$

N_p ~ PROPELLER SPEED (RPM)

D ~ PROPELLER DIAMETER (8.208 ft)

V_T ~ TRUE AIRSPEED (KNOTS)

J ~ ADVANCE RATIO

Table D-1. Propeller Efficiency.

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

$$\sigma = \left(T_o/T_a\right)\left(P_a/P_o\right)$$

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.

11. True airspeeds (V_T) were determined from the test altitude air density ratio (σ) and calibrated airspeed as follows:

$$V_T = \frac{V_{cal}}{\sqrt{\sigma}}$$

Airspeed Calibration

12. 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. 58, app E). Calibrated airspeeds (V_{cal}) were obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (ΔV_{pc}).

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

Weight and Balance

13. The aircraft weight and longitudinal cg were determined prior to each weight and/or cg configuration change. A typical internal ballast loading is shown in photo D-1. Weighing was accomplished using electronic scales located under the aircraft jack points with the crew on board at their designated stations.

Photo D-1. Internal Ballast.

Dynamic Stability

14. Dynamic stability characteristics were tested by using the techniques described in references 9, 11, and 13, appendix A. The data recorded from dynamic testing were presented as time histories of the pertinent parameters that describe the motion of the aircraft. Analyses of these time histories 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 two methods for all the conditions tested. These were the logarithmic decrement and time ratio method.

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

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

APPENDIX E. TEST DATA

INDEX

FIGURE 1

DUAL ENGINE CLIMB DRAG POLAR

BAC MODEL 200 USA S/N 71-21058

SYM	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
O	14180	187.3 (FWD)	13260	-29.0	2000	CRUISE	CLIMB
□	14370	187.3 (FWD)	2810	-12.5	2000	CRUISE	CLIMB

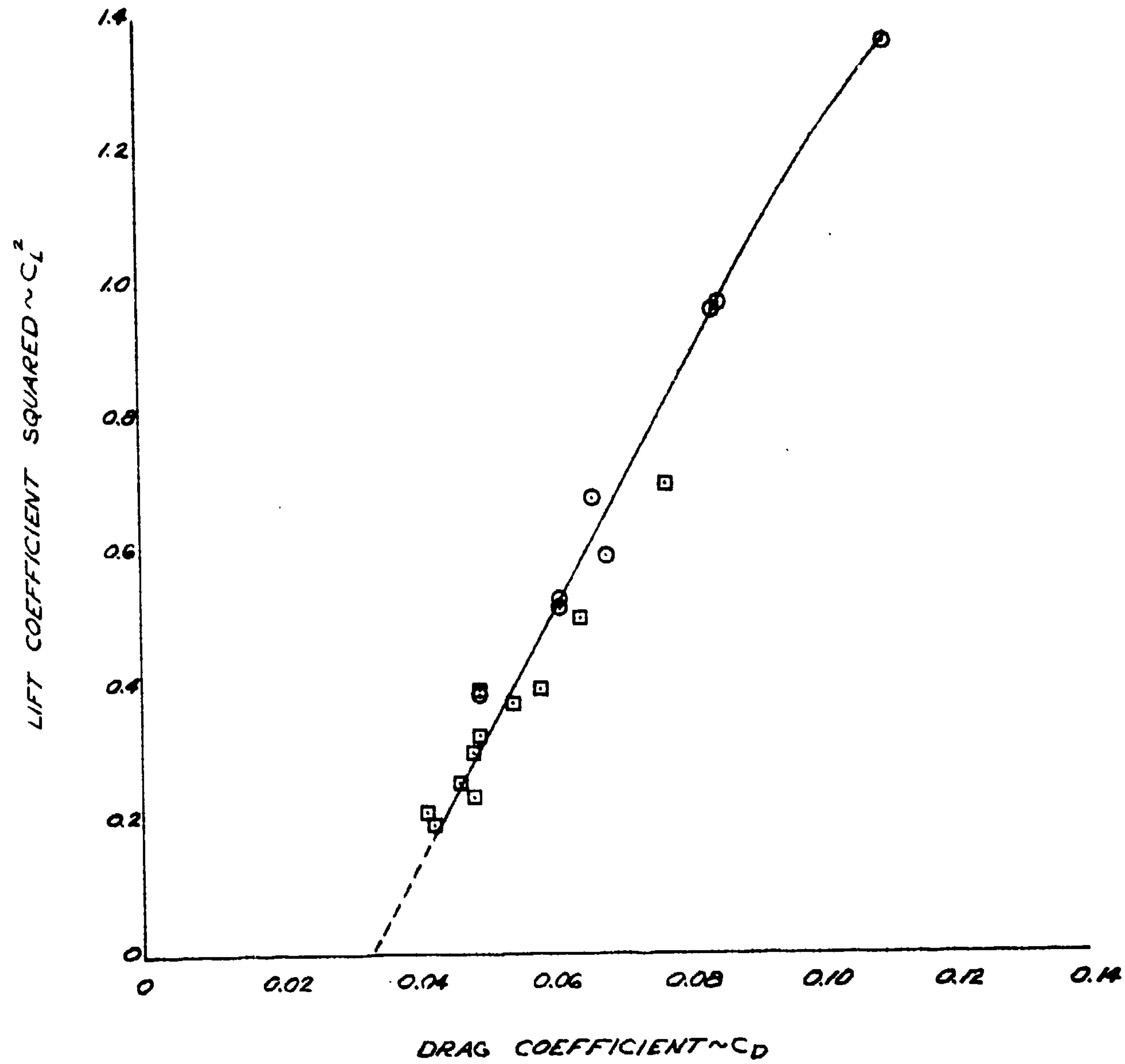


FIGURE 2
DUAL ENGINE CLIMB PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

STANDARD DAY
FORWARD CENTER OF GRAVITY
15000 POUNDS GROSS WEIGHT
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
AIRSPEEDS CALCULATED USING DRAG POLAR EQUATION AND
UACL ENGINE DATA

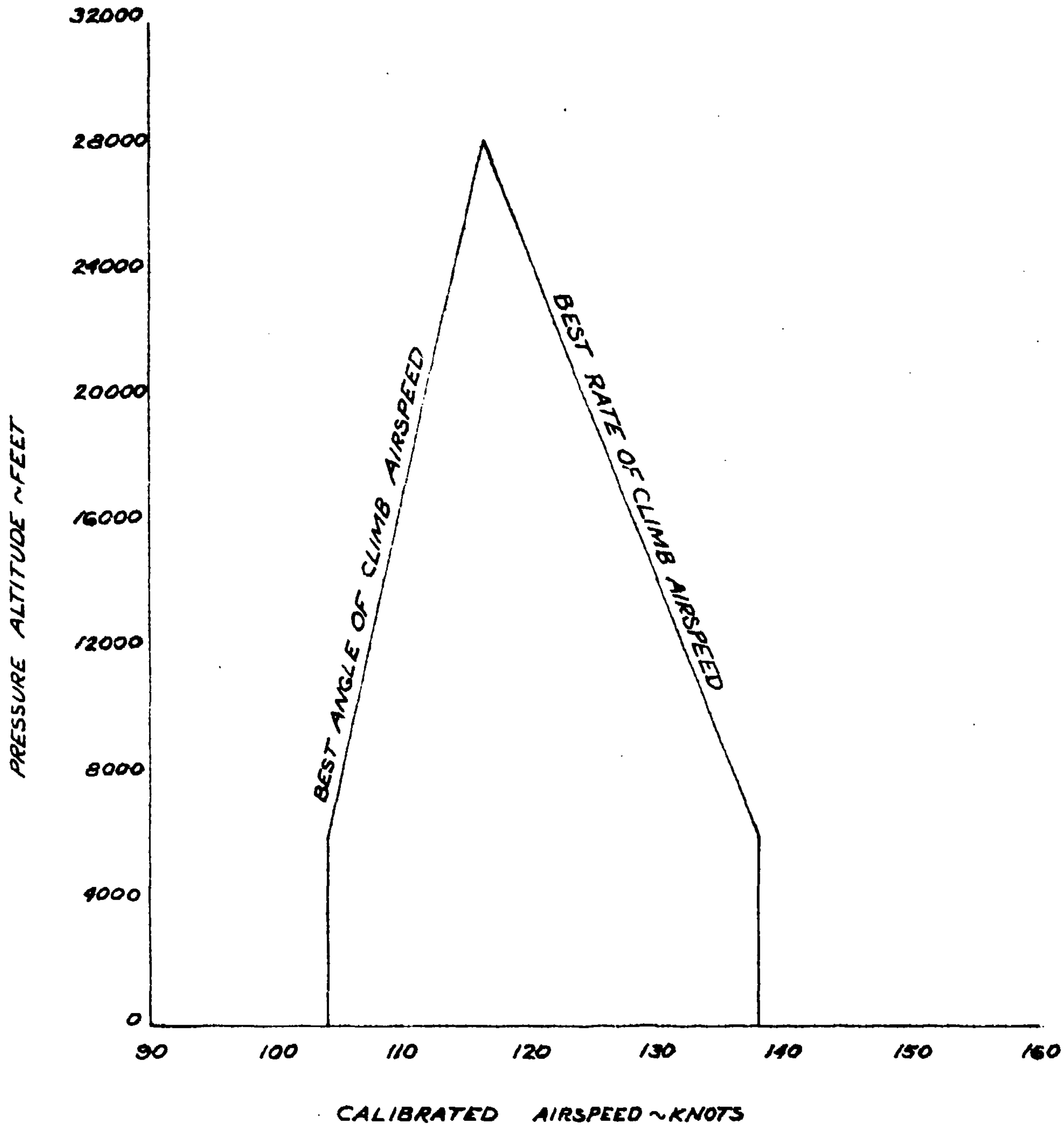


FIGURE 3
DUAL ENGINE CLIMB PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

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

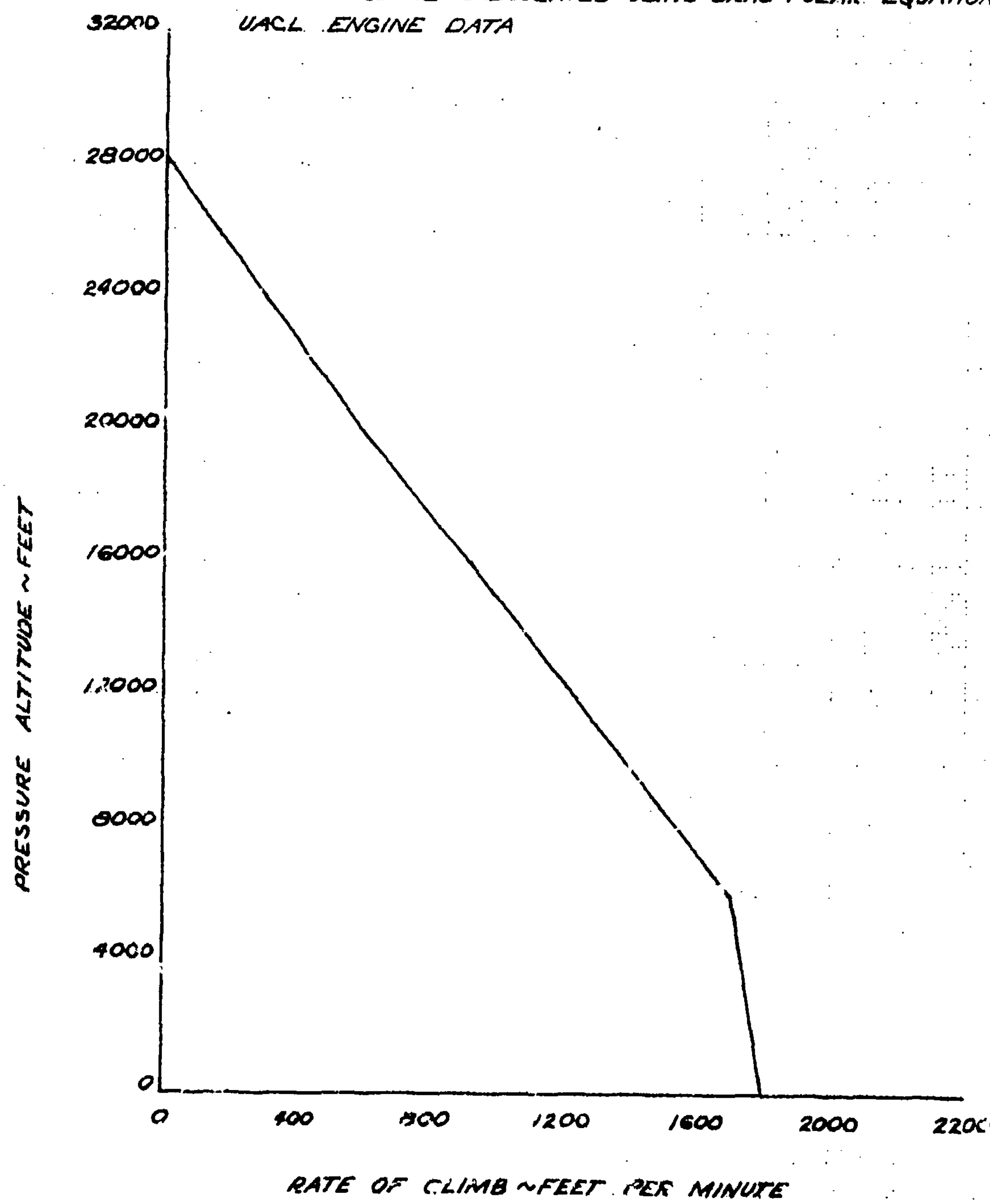


FIGURE 4
DUAL ENGINE CLIMB PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

STANDARD DAY
138 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

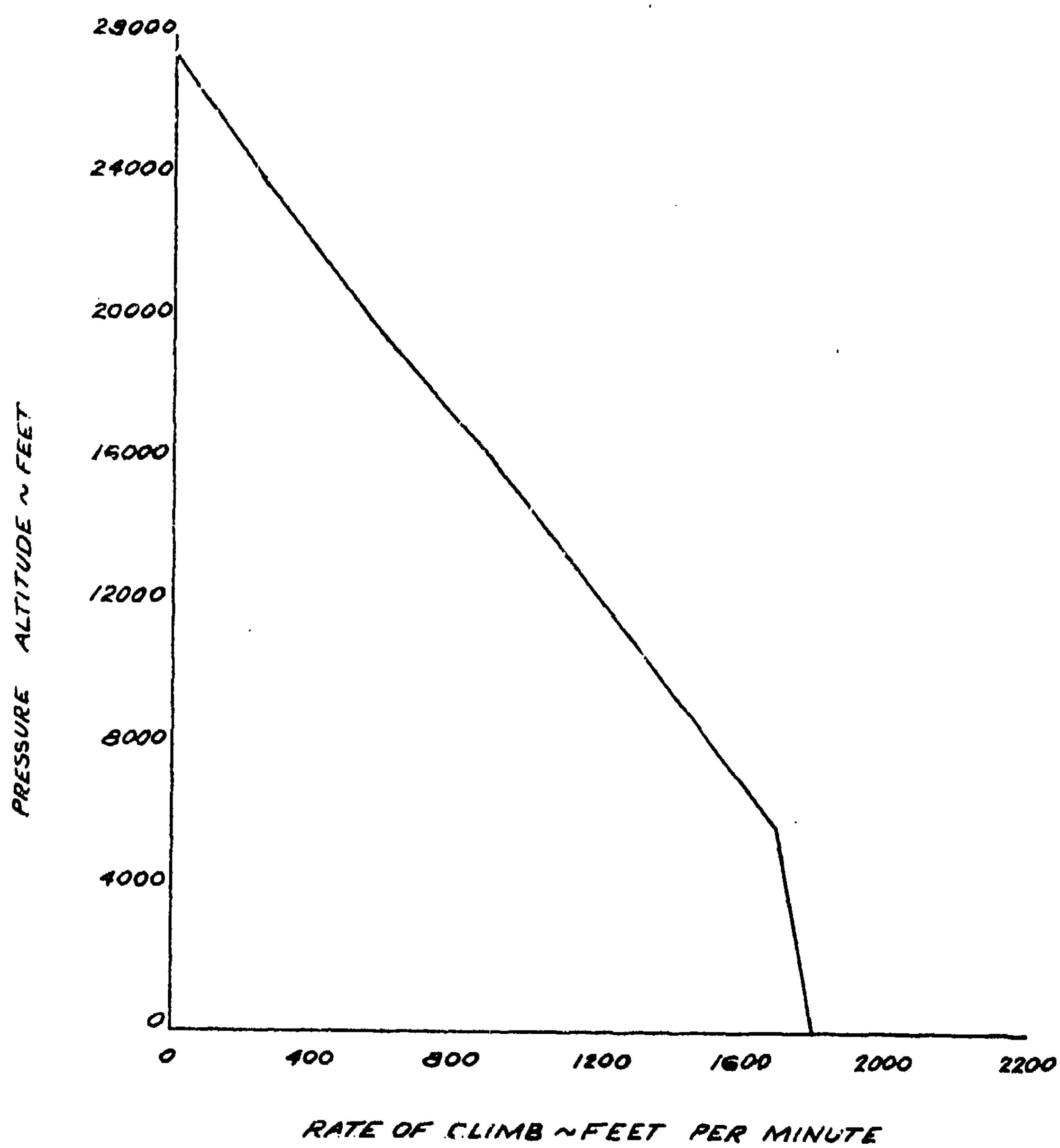


FIGURE 5

DUAL ENGINE CLIMB DRAG POLAR

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
14340	187.3	3350	-8.5	2000	TAKEOFF	CLIMB

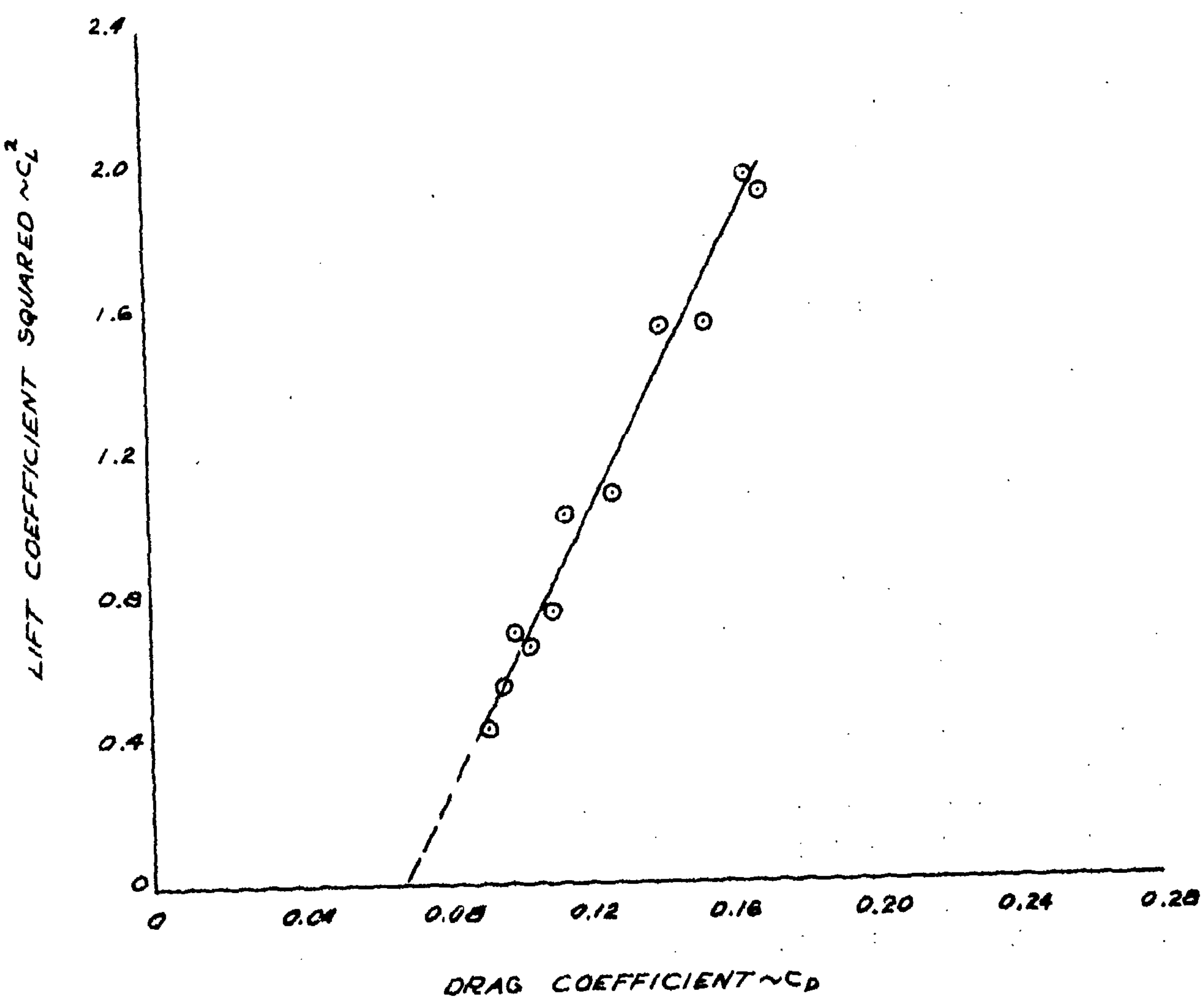


FIGURE 6

DUAL ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

STANDARD DAY
FORWARD CENTER OF GRAVITY
15000 POUNDS GROSS WEIGHT
TAKEOFF CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
AIRSPEEDS CALCULATED USING DRAG POLAR EQUATION AND UACL ENGINE DATA

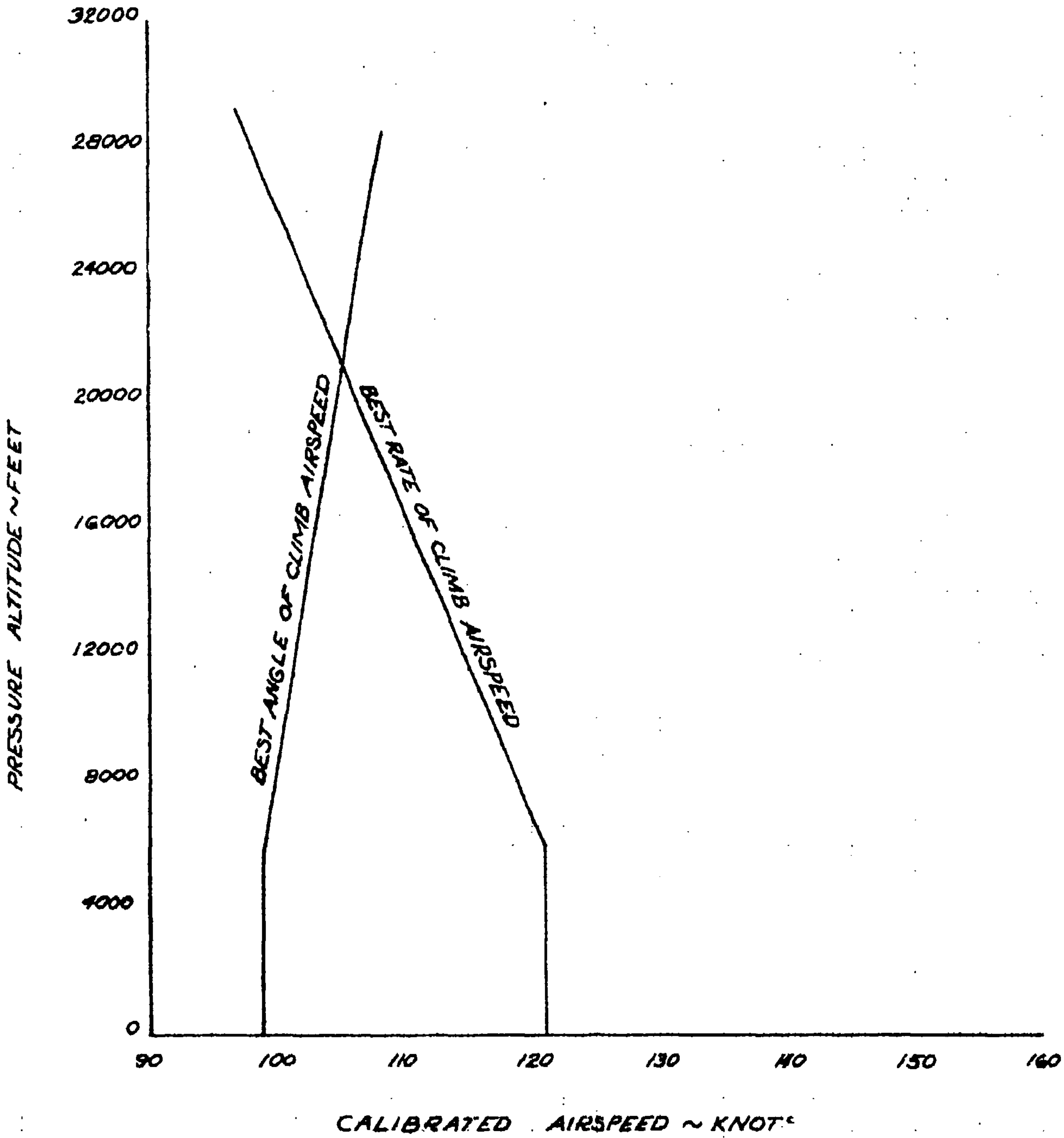


FIGURE 7

DUAL ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

STANDARD DAY

BEST RATE OF CLIMB AIRSPEED

TAKE OFF CONFIGURATION

FORWARD CENTER OF GRAVITY

15000 POUNDS GROSS WEIGHT

MAXIMUM CONTINUOUS SHAFT HORSEPOWER

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

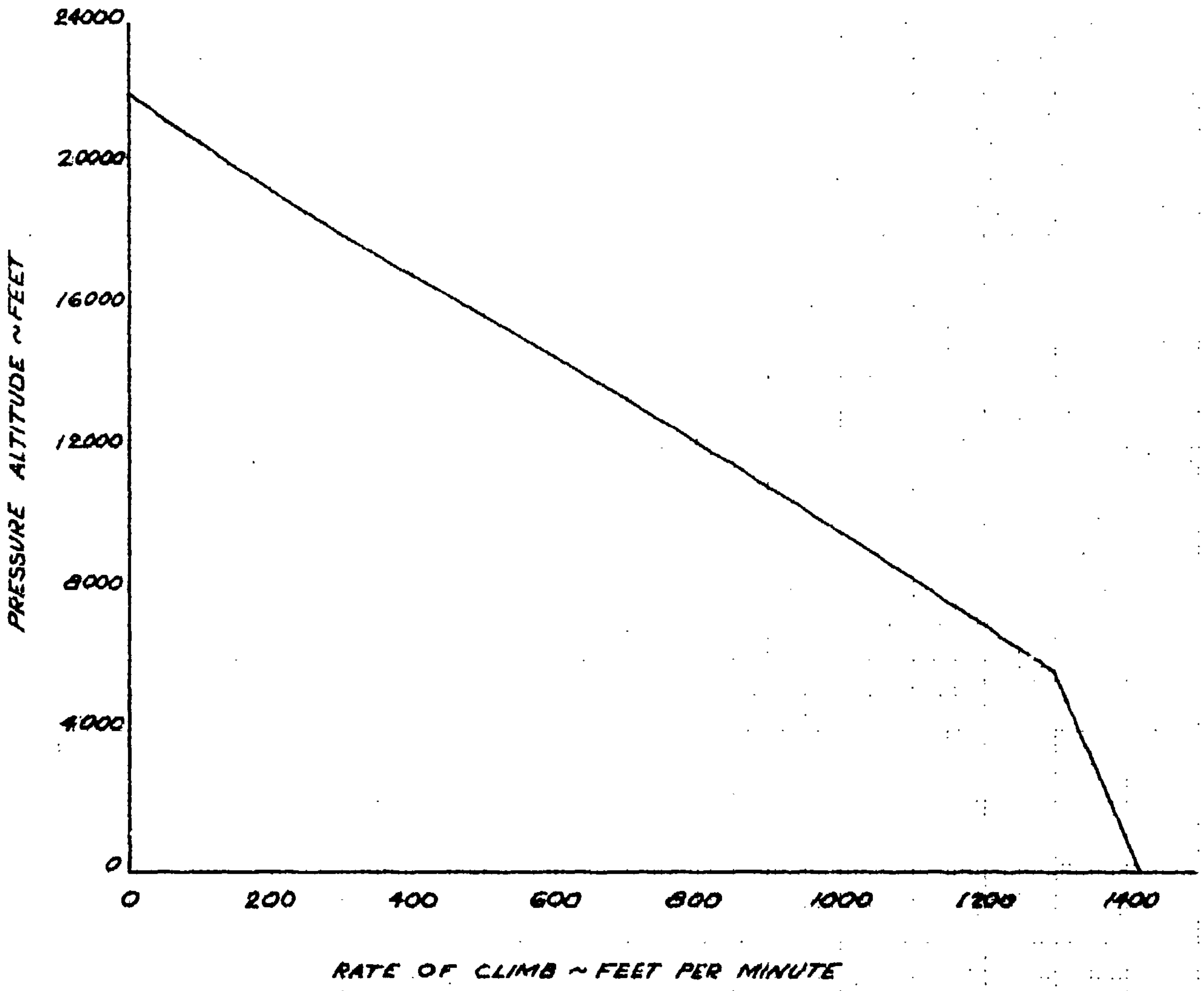


FIGURE 8

SINGLE ENGINE CLIMB DRAG POLAR

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
14800	187.4(FWD)	3200	-9.5	SINGLE ENGINE CRUISE	CLIMB

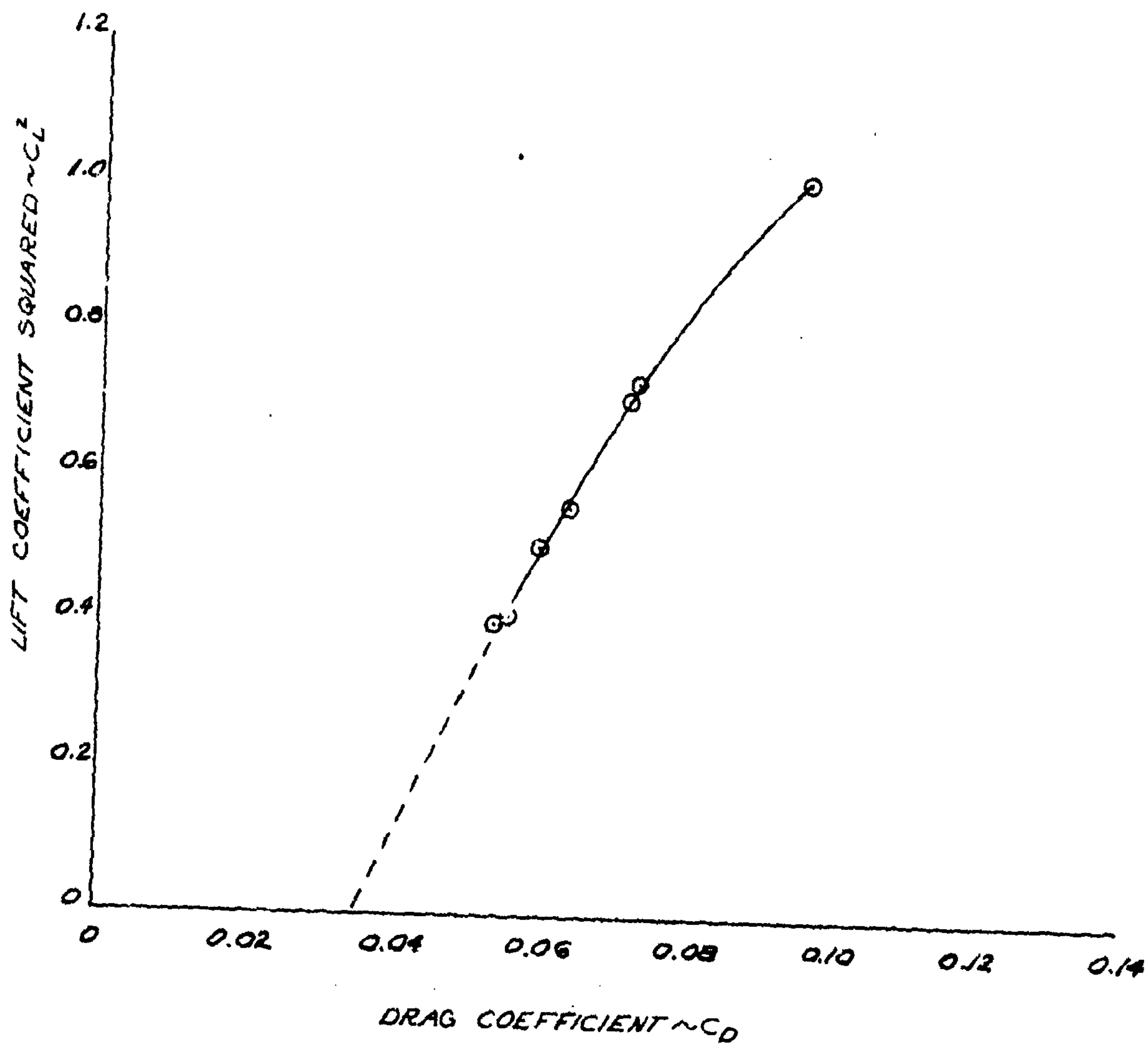


FIGURE 9

SINGLE ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

SEA LEVEL
FORWARD CENTER OF GRAVITY
2000 RPM PROPELLER SPEED
MAXIMUM CONTINUOUS POWER AVAILABLE

NOTE: HOT DAY DEFINITION OBTAINED FROM AIR FORCE-NAVY AERONAUTICAL (ANA) BULLETIN 421, REFERENCE MIL-C-8678 (AER)

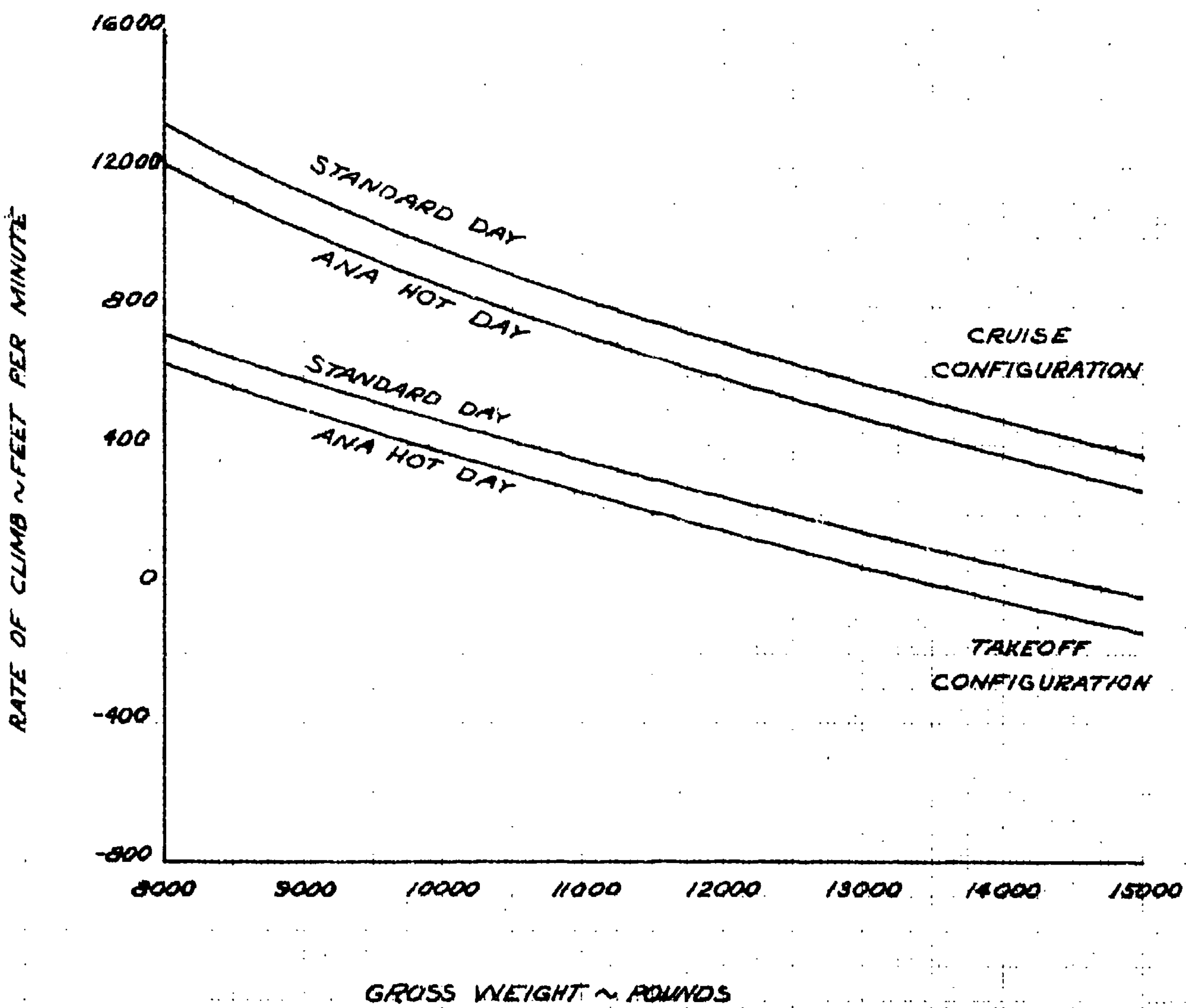


FIGURE 10

SINGLE ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

CRUISE CONFIGURATION

STANDARD DAY

FORWARD CENTER OF GRAVITY

2000 RPM PROPELLER SPEED

MAXIMUM CONTINUOUS POWER AVAILABLE

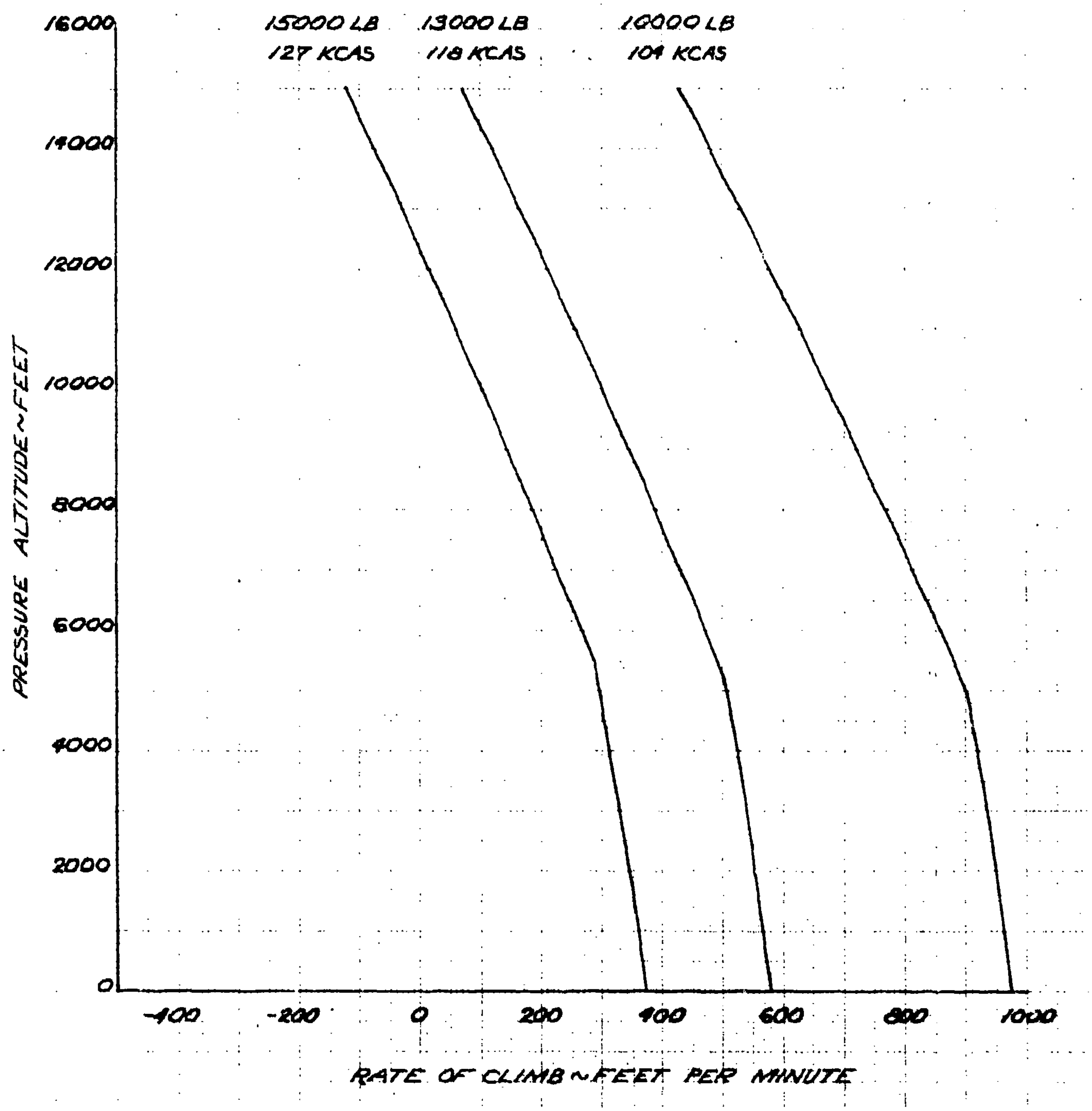


FIGURE 11

SINGLE ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

CRUISE CONFIGURATION

ANA HOT DAY

FORWARD CENTER OF GRAVITY

2000 RPM PROPELLER SPEED

MAXIMUM CONTINUOUS POWER AVAILABLE

NOTE: HOT DAY DEFINITION OBTAINED FROM AIR FORCE-NAVY AERONAUTICAL (ANA) BULLETIN 421, REFERENCE MIL-C-8678 (AER)

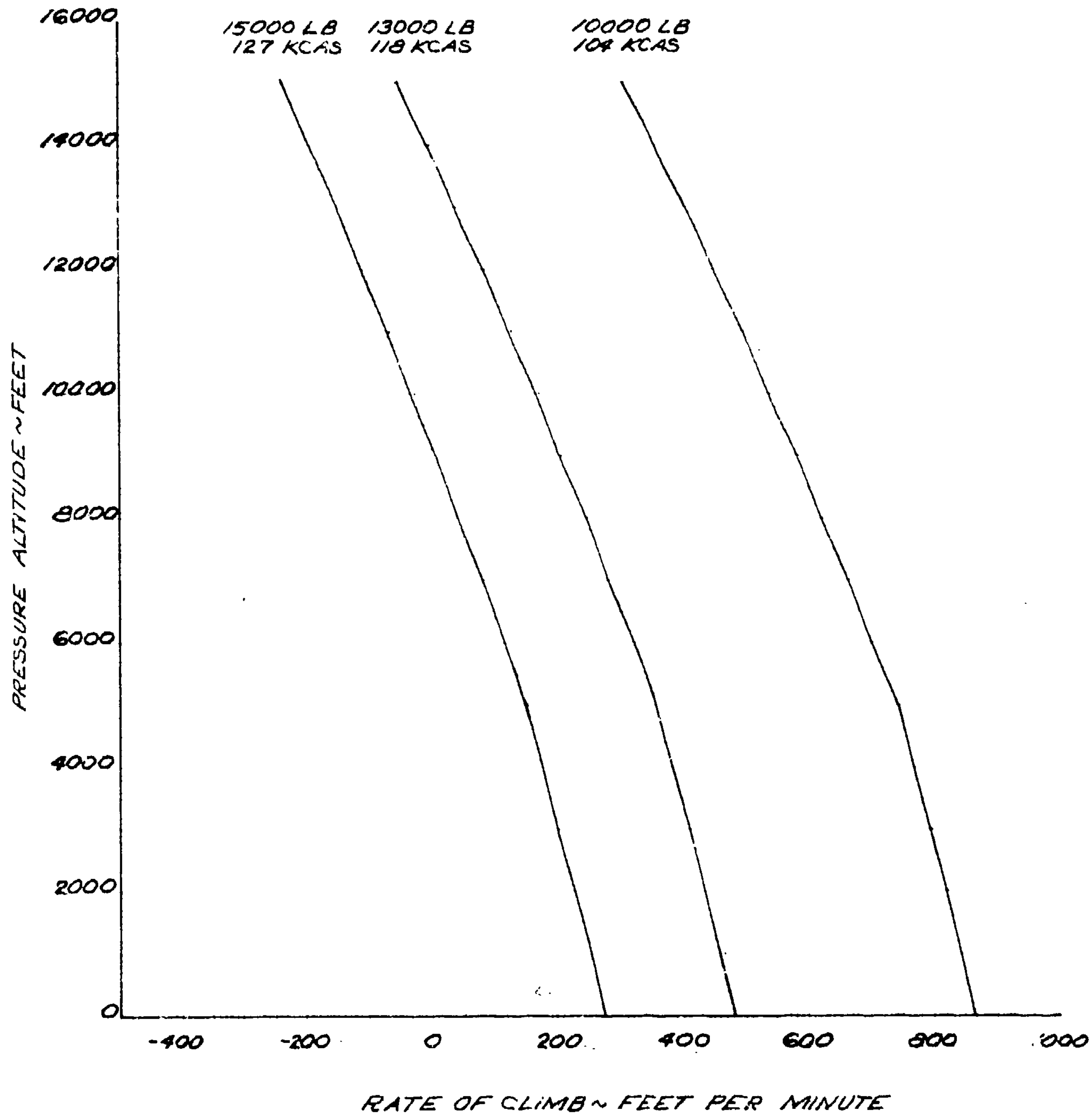


FIGURE 12

SINGLE ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

TAKEOFF CONFIGURATION
ANA HOT DAY
FORWARD CENTER OF GRAVITY
2000 RPM PROPELLER SPEED
MAXIMUM CONTINUOUS POWER AVAILABLE

NOTE: HOT DAY DEFINITION OBTAINED FROM AIR FORCE-NAVY AERONAUTICAL (ANA) BULLETIN 421, REFERENCE MIL-C-8678 (AER)

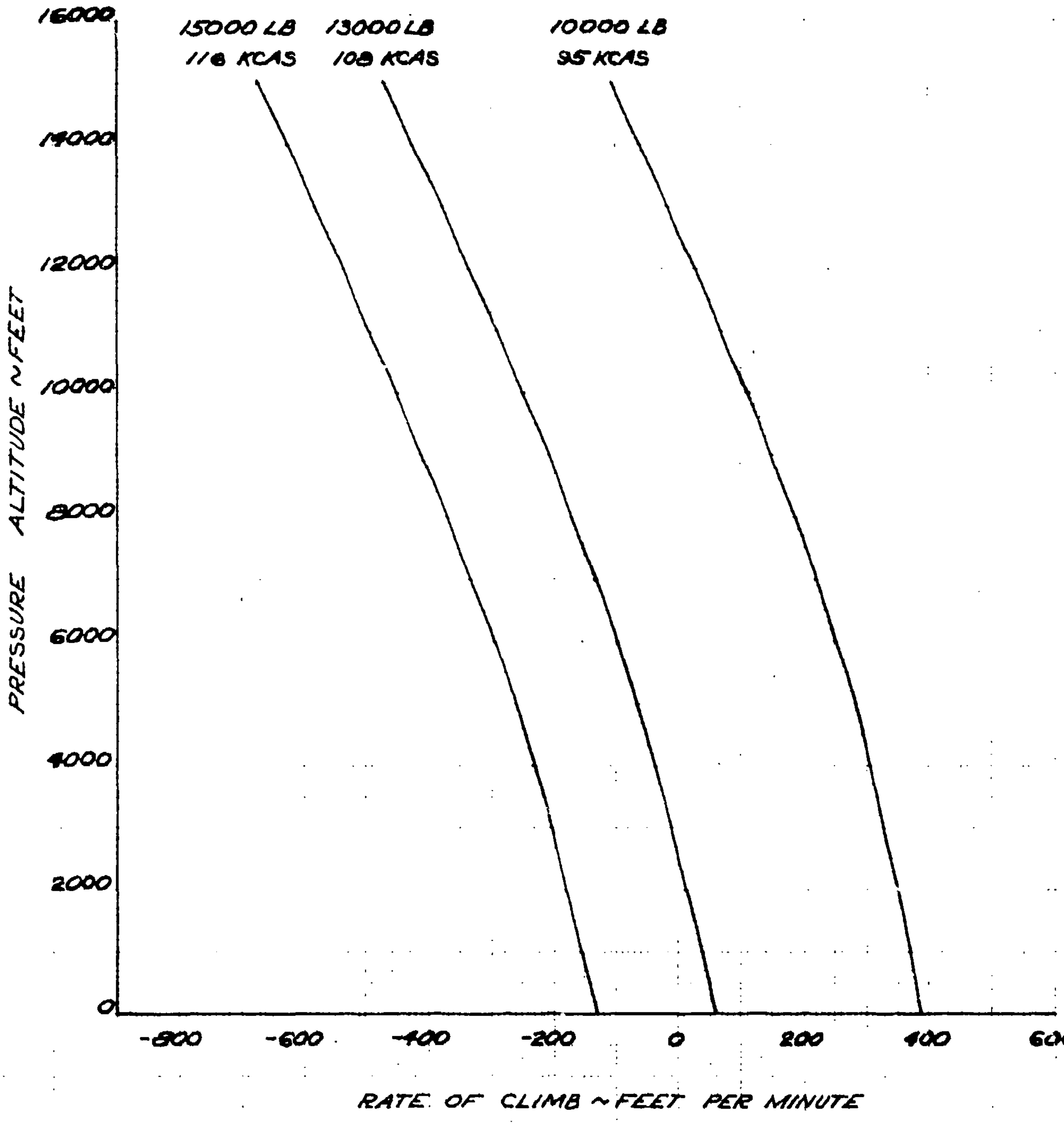


FIGURE 13
SINGLE ENGINE CLIMB PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

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

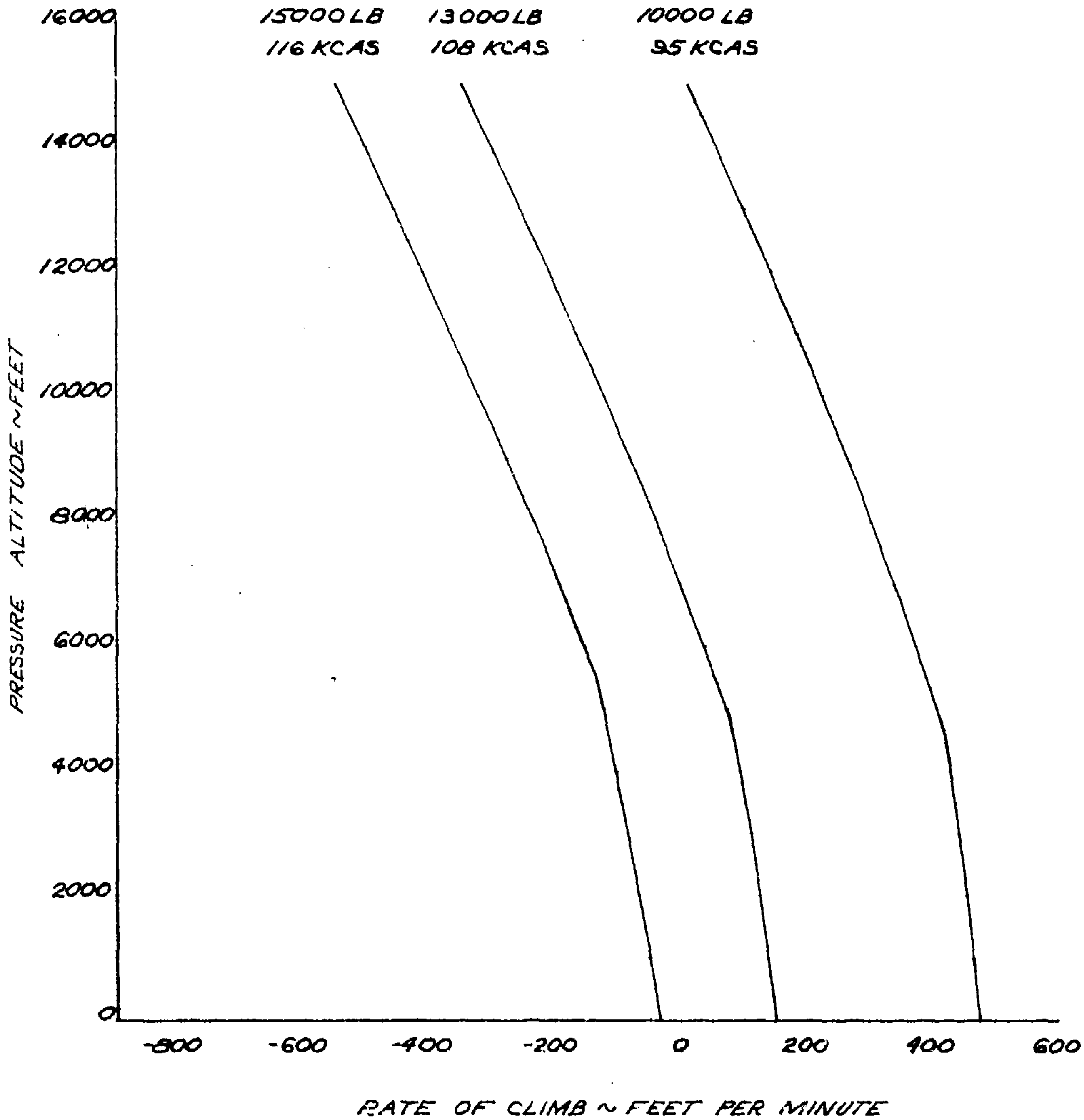


FIGURE 14

PROPELLER FEATHERED GLIDE DRAG POLAR

BAC MODEL 200 USA S/N 71-21058

SYM	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
○	11370	183.2 (FWD)	9560	-8.5	CRUISE POWER-OFF	GLIDE
□	11060	183.1 (FWD)	8890	-14.0	CRUISE POWER-OFF	GLIDE
△	14620	187.5 (FWD)	8460	-20.0	CRUISE POWER-OFF	GLIDE
◇	14820	187.4 (FWD)	9440	-9.5	CRUISE POWER-OFF	GLIDE
△	14760	187.9 (FWD)	11369	7.0	CRUISE POWER-OFF	GLIDE

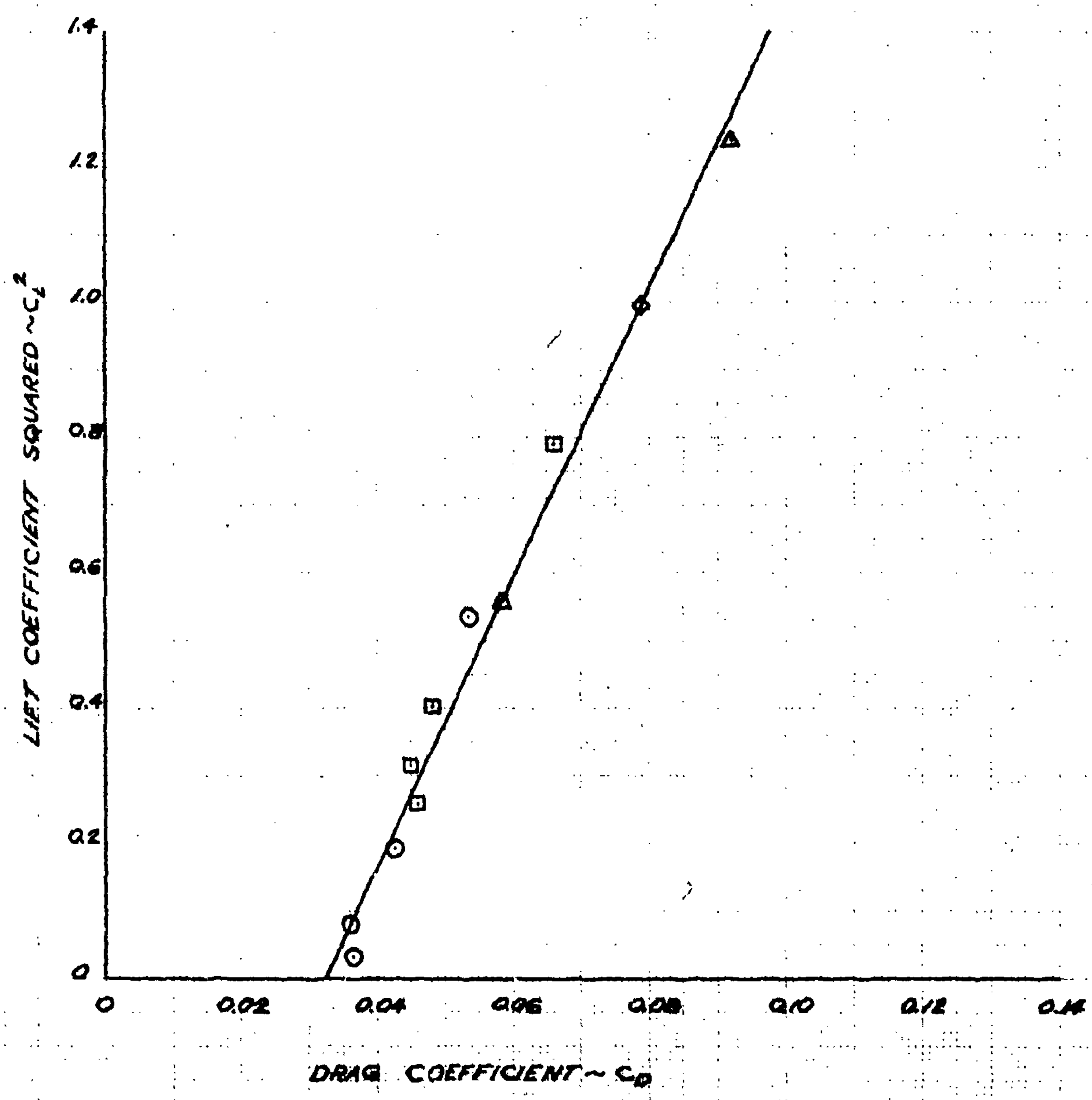


FIGURE 15

DUAL ENGINE LEVEL FLIGHT PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
11900	183.4(FWD)	15330	-12.0	2000	CRUISE	LEVEL FLIGHT

NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA

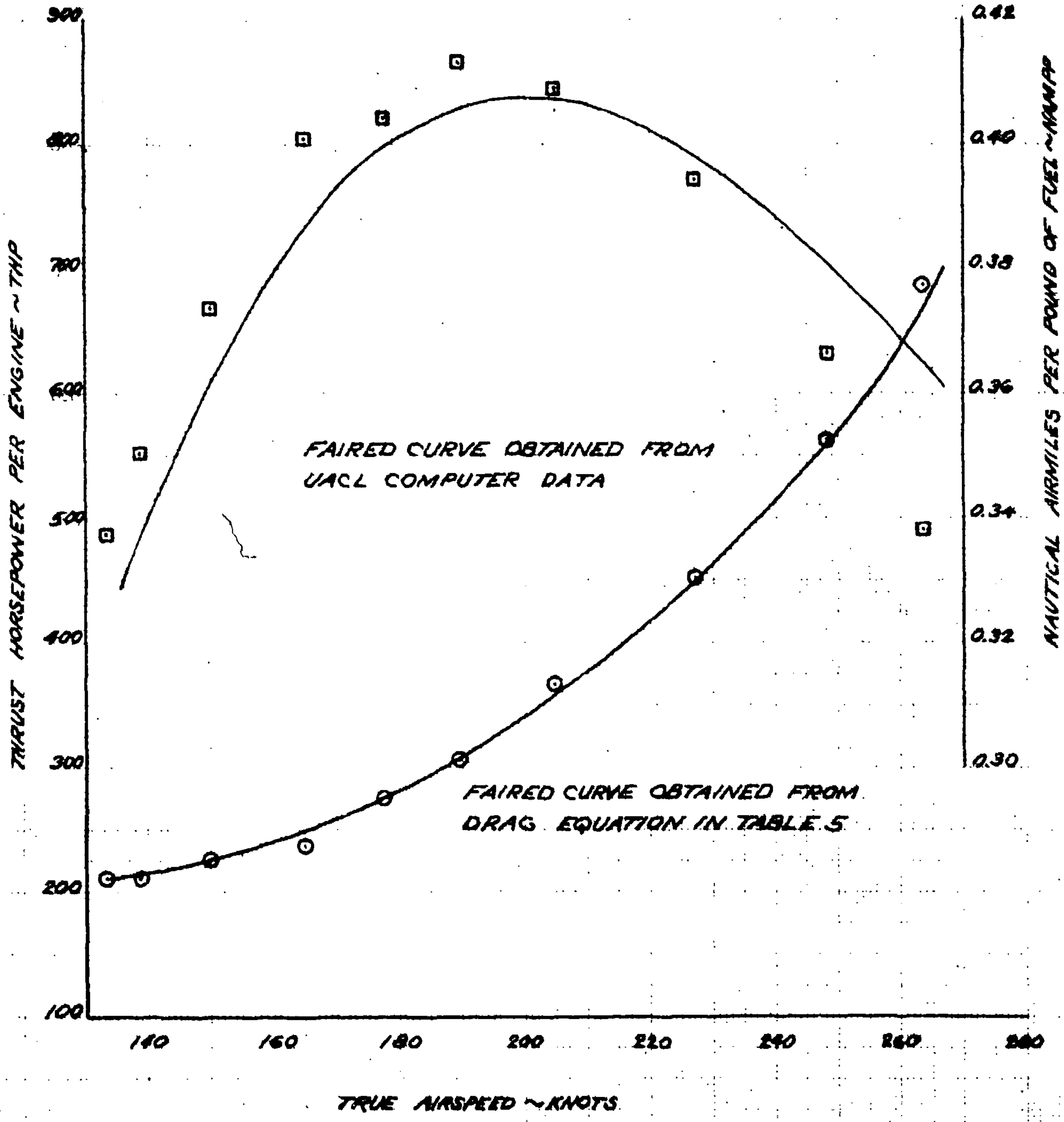


FIGURE 16
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	AVG PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
14600	188.1 (FWD)	13260	-23.0	2000	CRUISE	LEVEL FLIGHT

NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA

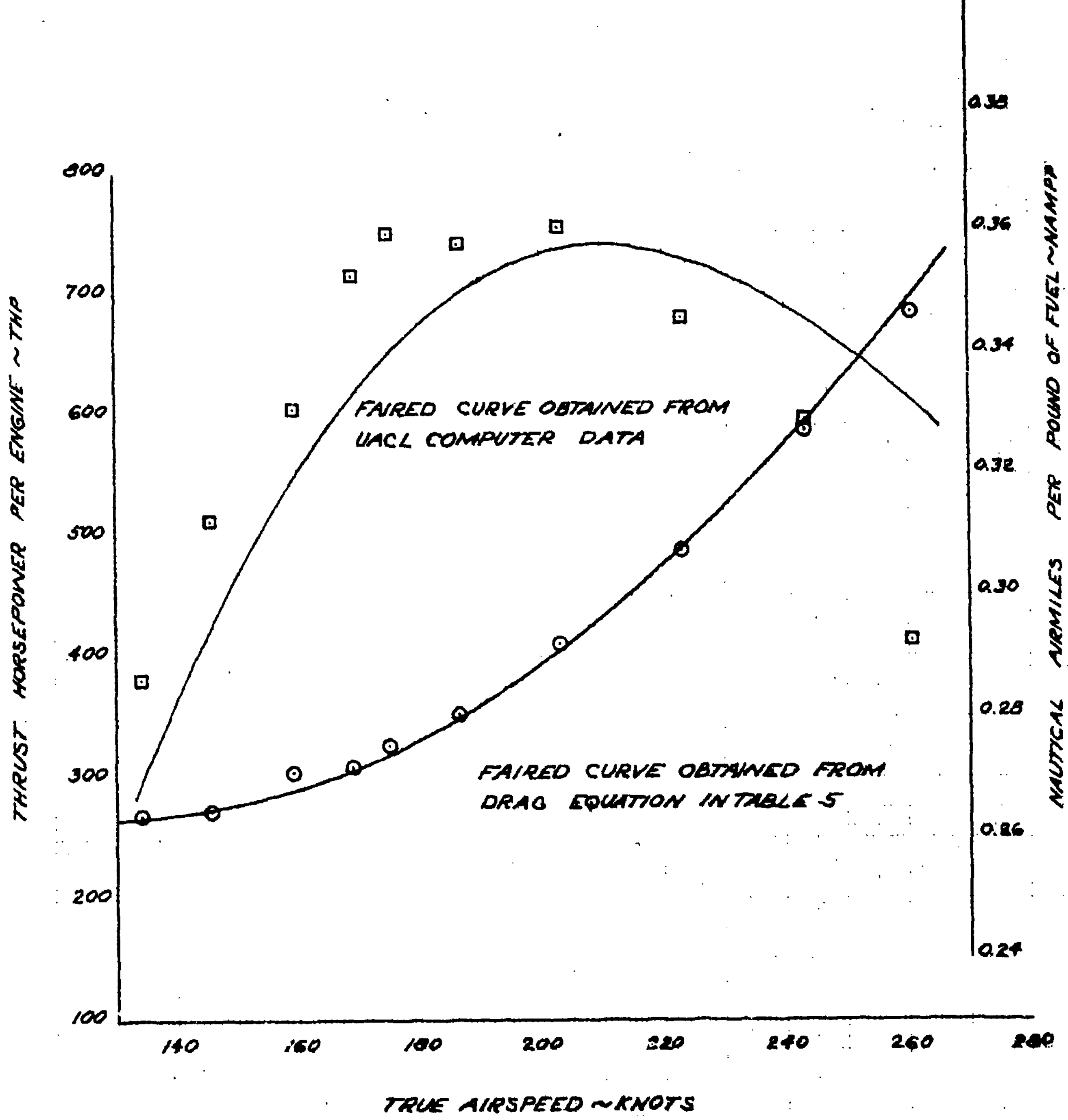


FIGURE 17

DUAL ENGINE LEVEL FLIGHT DRAG POLAR

BAC MODEL 200 USA S/N 71-21058

SYM	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
○	11900	183.4 (FWD)	15330	-12.0	CRUISE	LEVEL FLIGHT
□	11230	183.3 (FWD)	14620	-18.0	CRUISE	LEVEL FLIGHT
△	14600	187.6 (FWD)	13260	-23.0	CRUISE	LEVEL FLIGHT

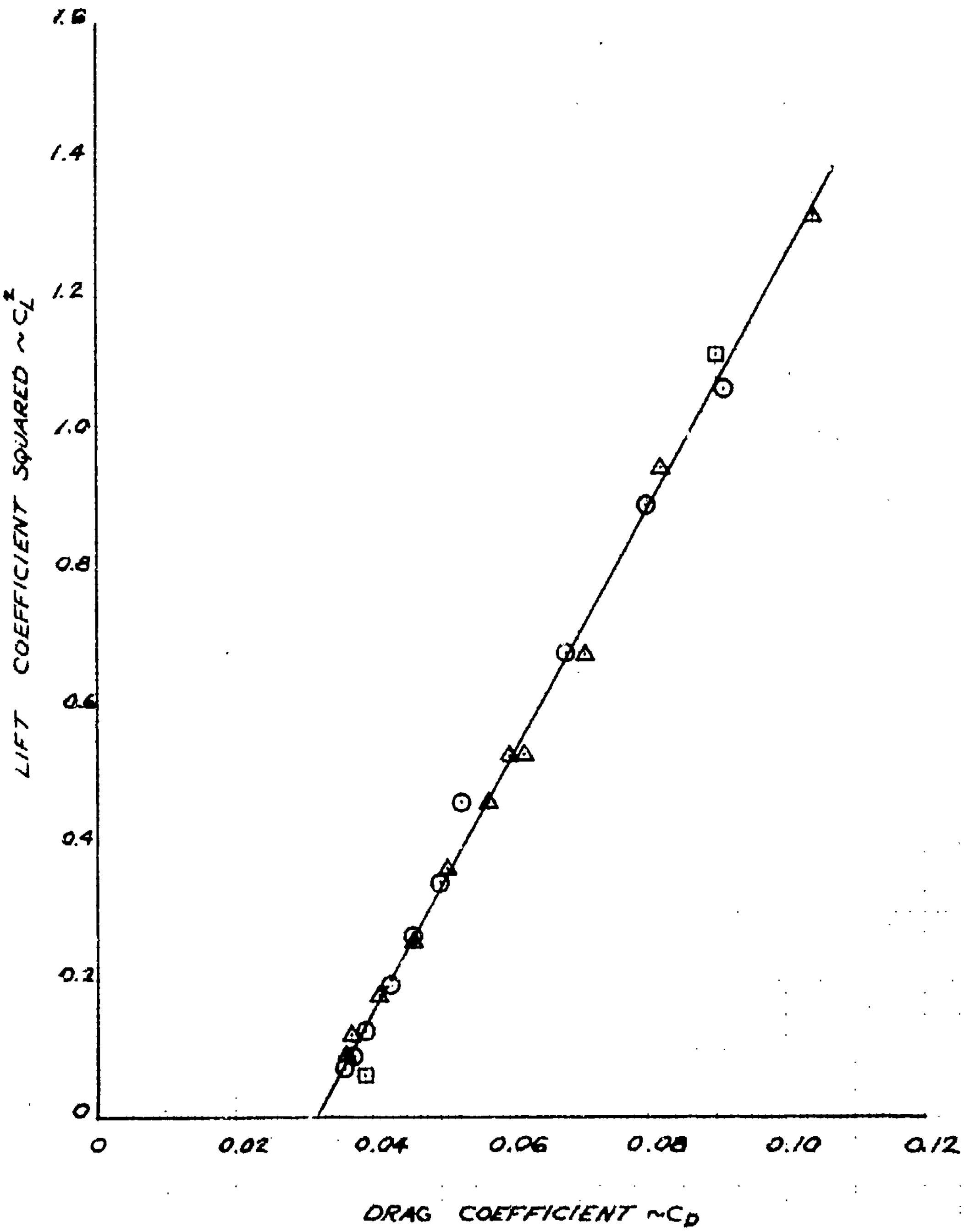


FIGURE 18

SINGLE ENGINE LEVEL FLIGHT PERFORMANCE

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION
14750	193.3 (AFT)	10220	-3.0	2000	CRUISE

NOTE: 1. NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
2. LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED

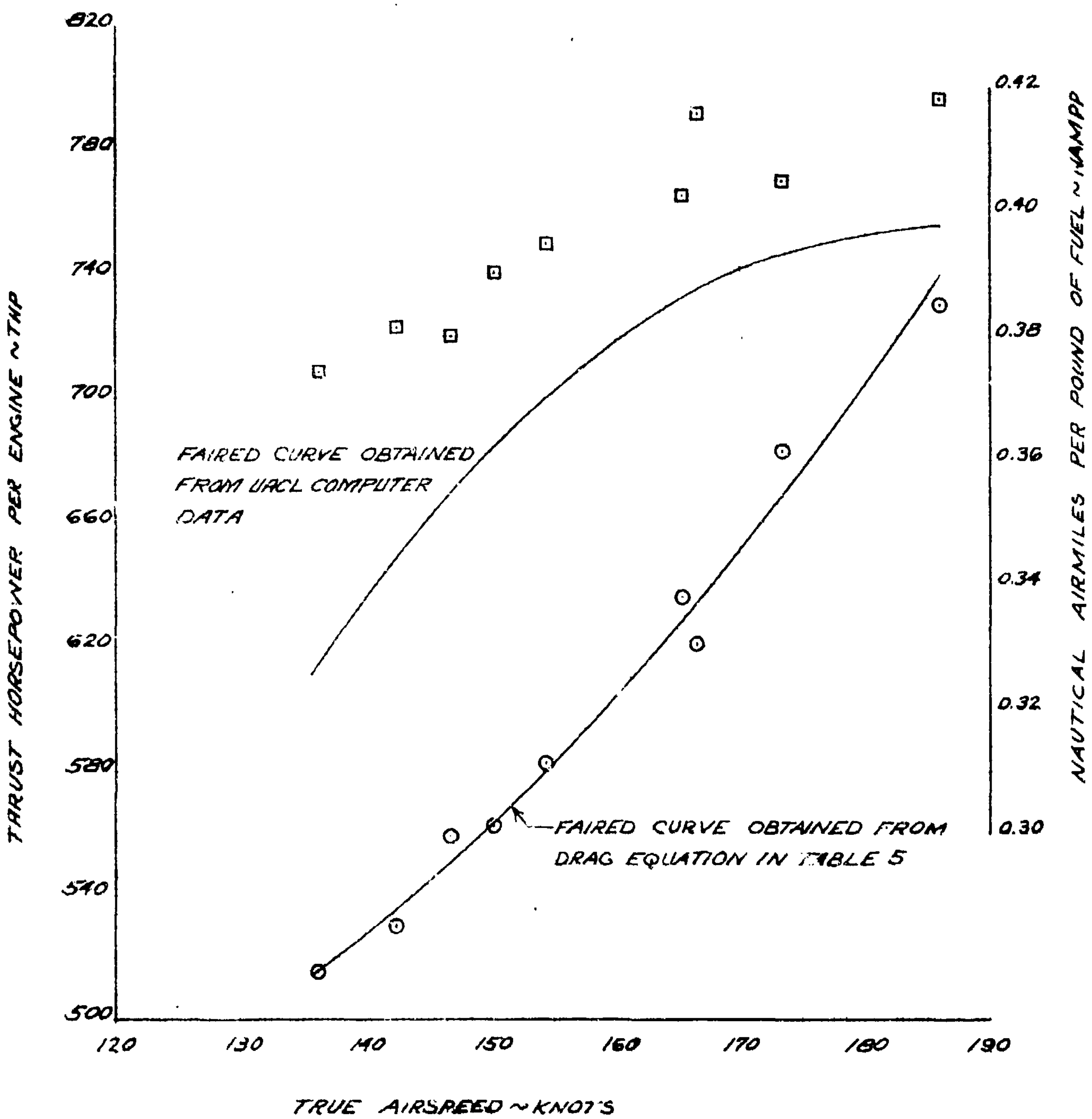


FIGURE 19

SINGLE ENGINE DRAG POLAR

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
14750	193.3 (AFT)	10220	-3.0	2000	CRUISE	LEVEL FLIGHT

NOTE: 1 LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED

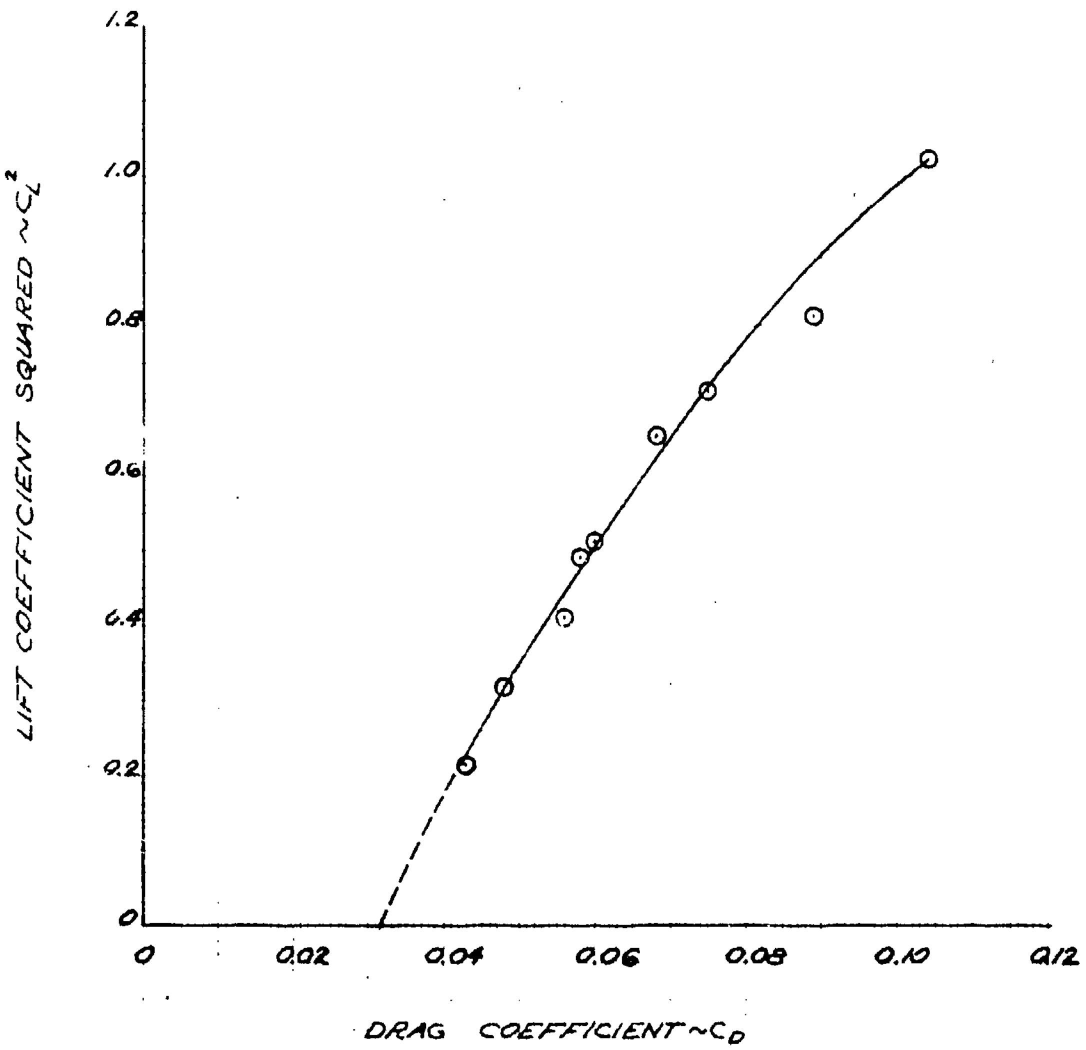


FIGURE 20
MAXIMUM LEVEL FLIGHT AIRSPEED (VH)
BAC MODEL 200 USA S/N 71-21058

STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
NOTE: MAXIMUM LEVEL FLIGHT AIRSPEED BASED ON
1. MAXIMUM CRUISE POWER AVAILABLE (FIGURES 52 THRU 55)
2. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
3. ACCESSORY LOSSES: 14.25 SHP< 10°C<19.29 SHP
4. BLEED AIR: 6.1 PPM<17°C<4.9 PPM
5. ENGINE INLET EFFICIENCY CURVE FIGURES 56 AND 57
6. PROPELLER SPEED = 2000 RPM

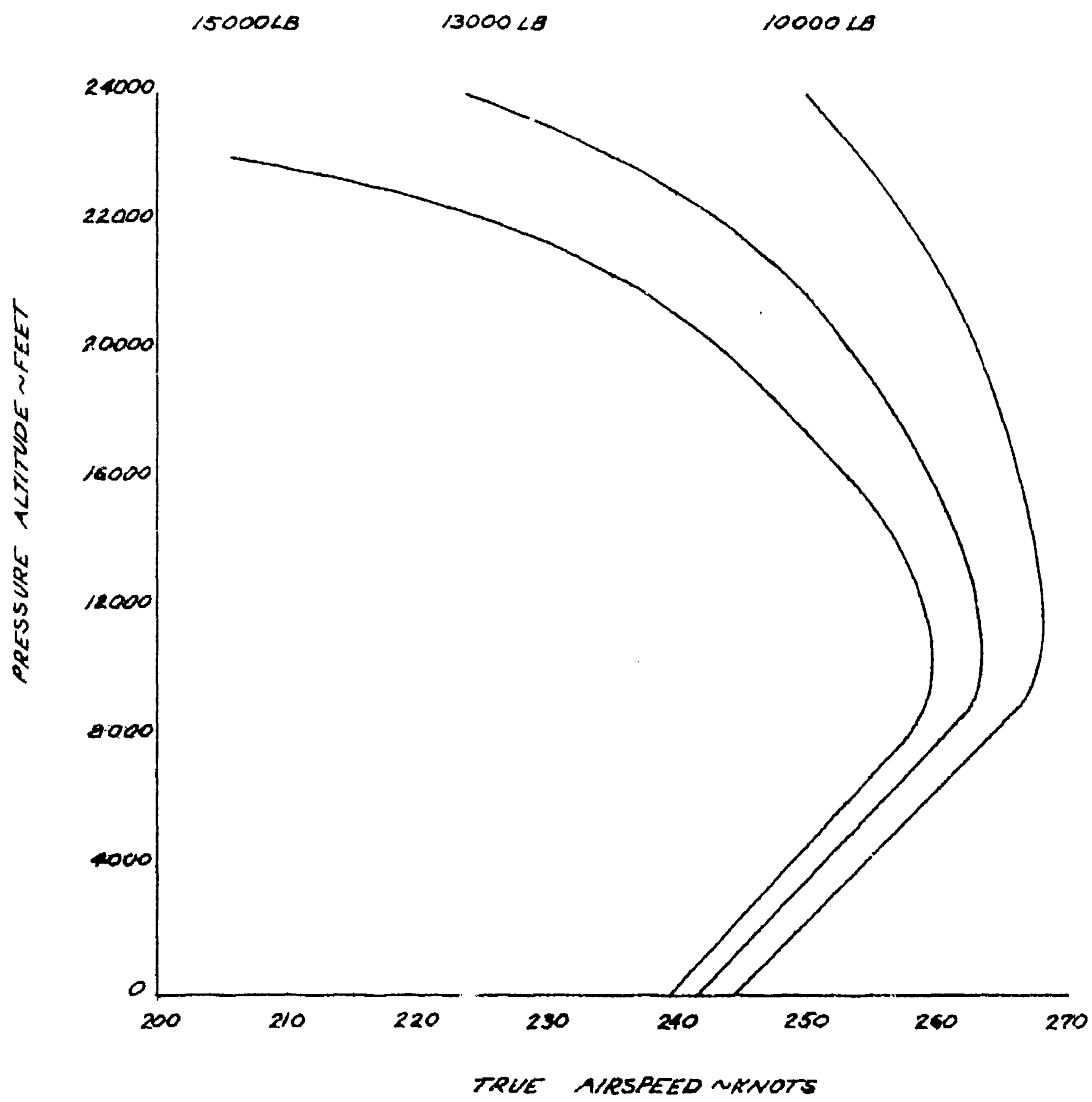
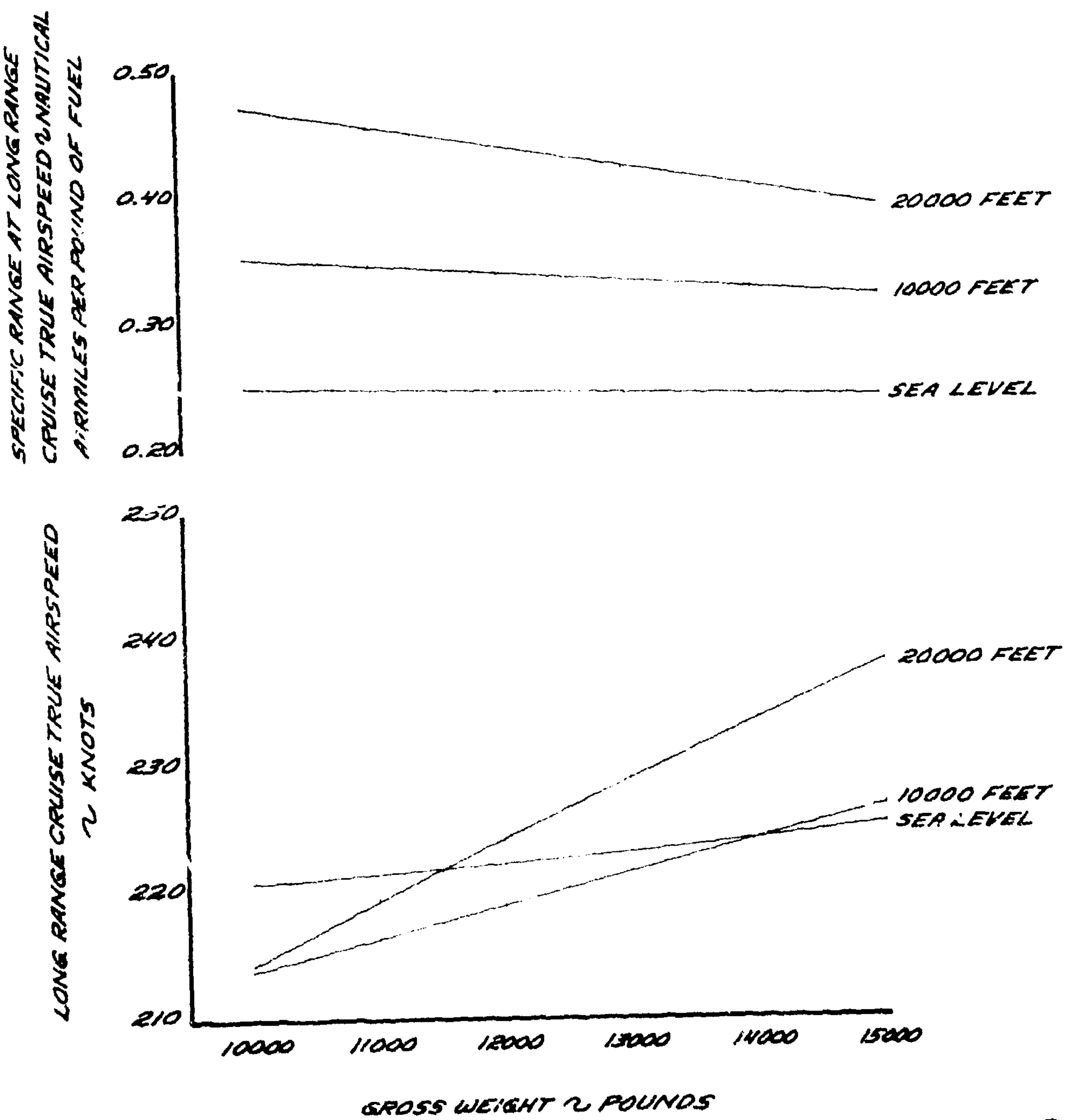


FIGURE 21

LEVEL FLIGHT RANGE SUMMARY

BAC MODEL 200 USA S/N 71-21058

NOTE: 1 CRUISE CONFIGURATION
2 STANDARD DAY CONDITIONS
3 FORWARD CENTER OF GRAVITY
4 PROPELLER SPEED = 2000 RPM
5 LONG RANGE CRUISE TRUE AIRSPEED DEFINED AT 0.99 OF MAXIMUM NAUTICAL AIRMILES PER POUND OF FUEL
6 SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA ON MINIMUM ENGINE



LD 5-74

FIGURE 22

MAXIMUM ENDURANCE SUMMARY

BAC MODEL 200 USA S/N 71-21058

STANDARD DAY CONDITIONS

FORWARD CENTER OF GRAVITY

2000 RPM PROPELLER SPEED

SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA

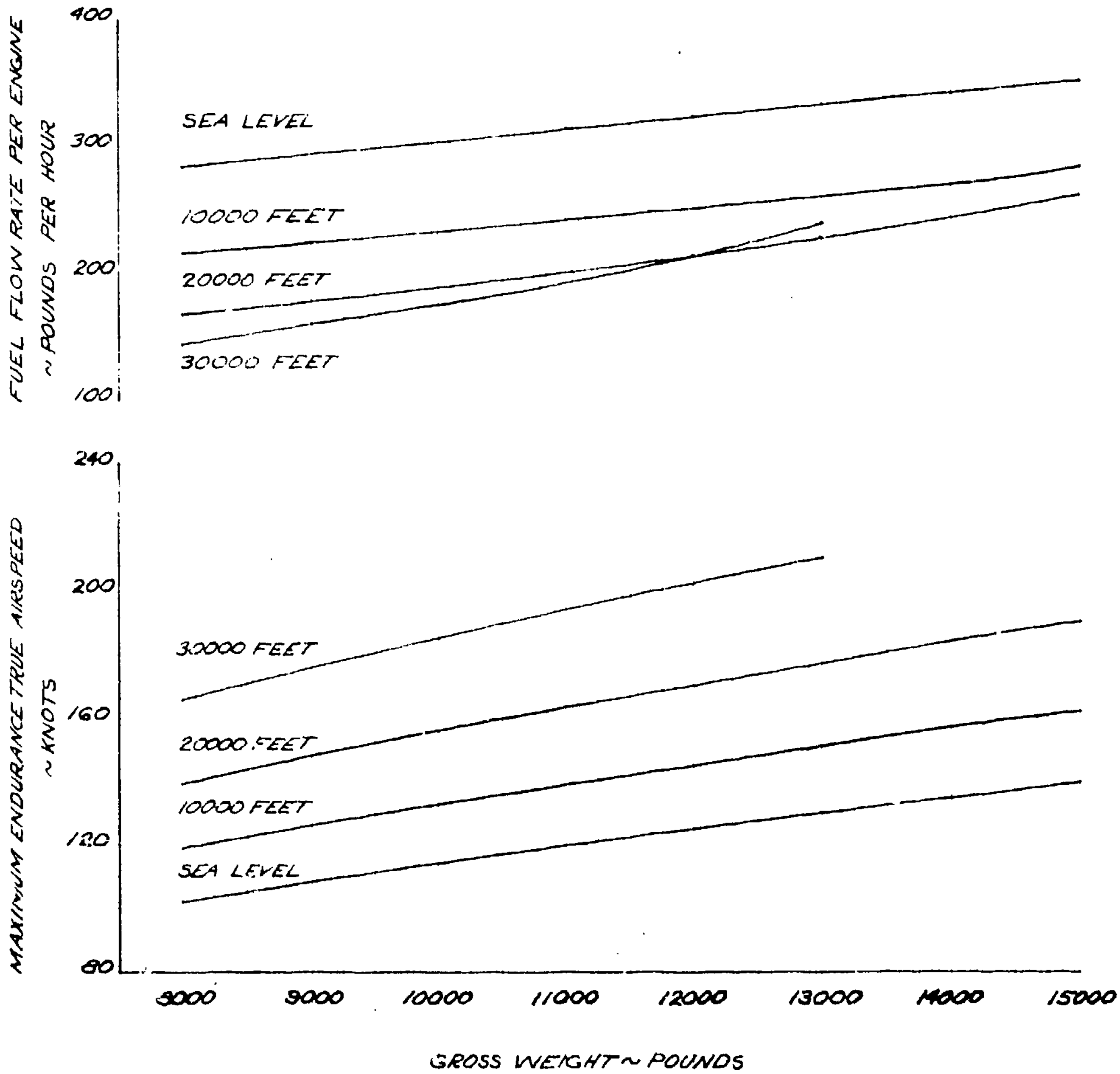
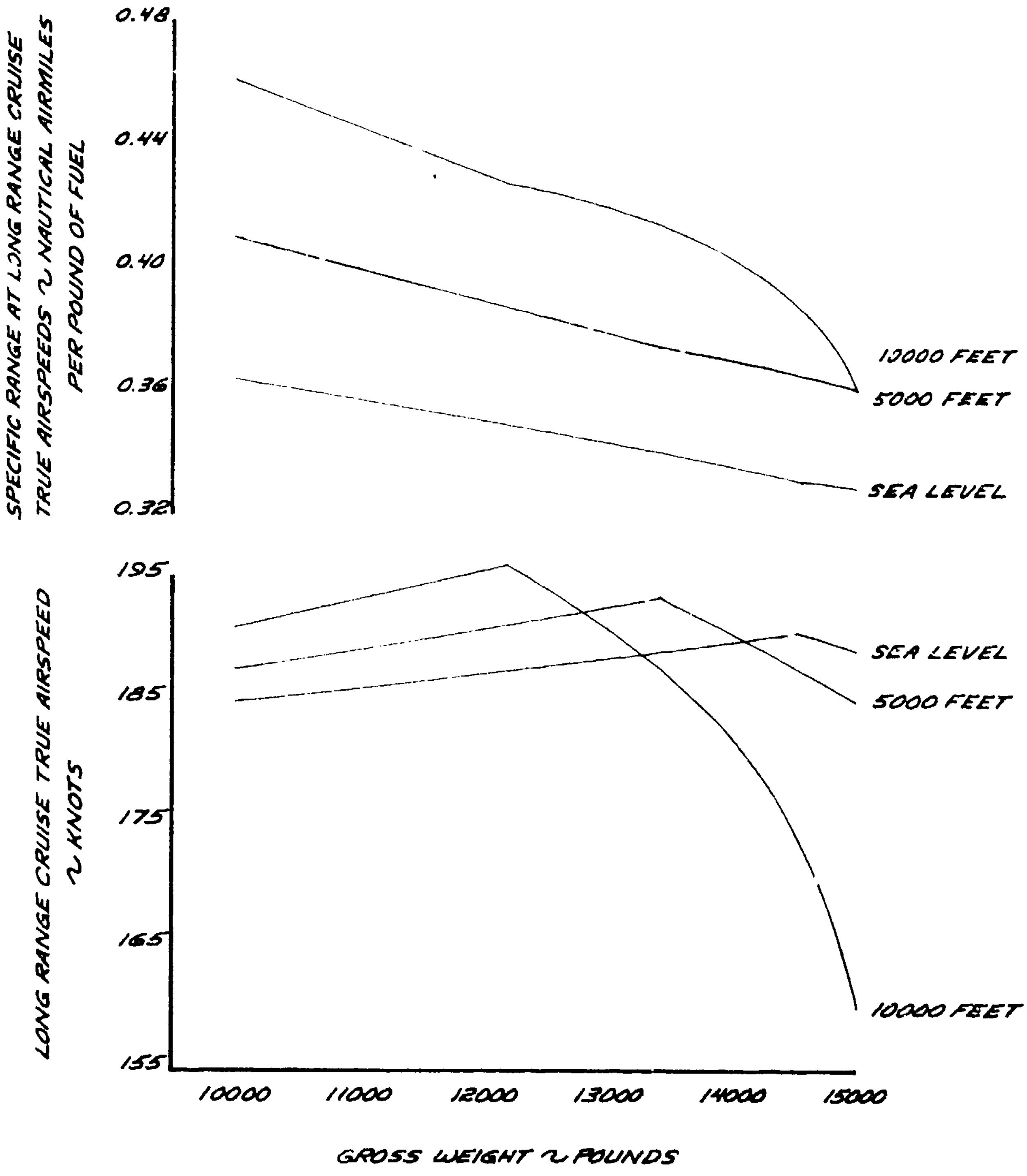


FIGURE 23

SINGLE ENGINE LEVEL FLIGHT RANGE SUMMARY

BAC MODEL 200 USA S/N 71-21058

NOTE: 1. CRUISE CONFIGURATION
2. STANDARD DAY CONDITIONS
3. FORWARD CENTER OF GRAVITY
4. PROPELLER SPEED = 2000 RPM
5. LONG RANGE CRUISE TRUE AIRSPEED DEFINED AT 0.99 OF MAXIMUM NAUTICAL AIRMILES PER POUND OF FUEL
6. SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA ON MINIMUM ENGINE



LD 9-74

FIGURE 24

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

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
14510	197.1 (AFT)	10970	3.5	CRUISE	LEVEL FLIGHT

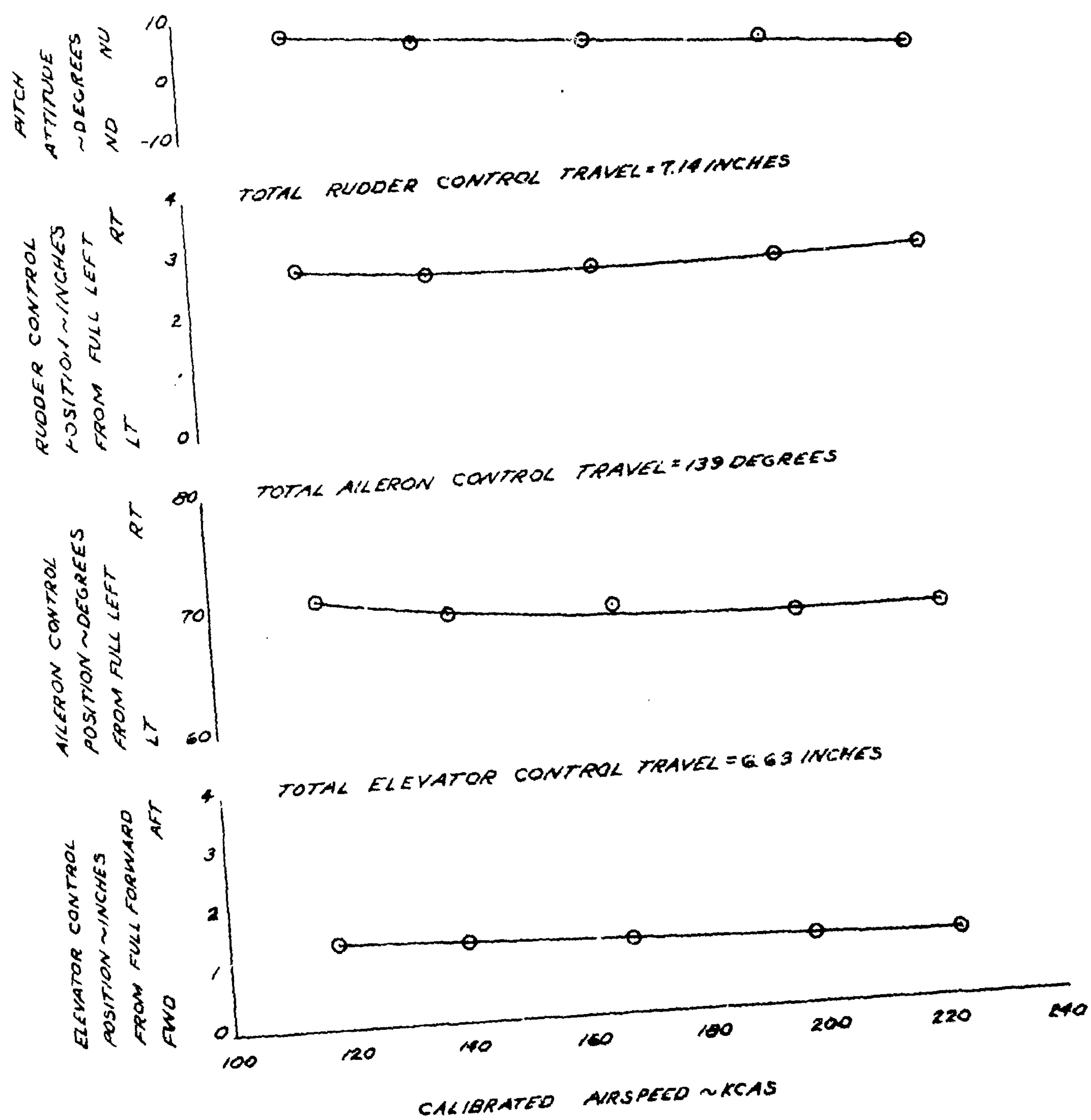


FIGURE 25
TRIMMER CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION	FLIGHT CONDITION
14570	197.1(AFT)	10970	3.5	2000	CRUISE	LEVEL FLIGHT

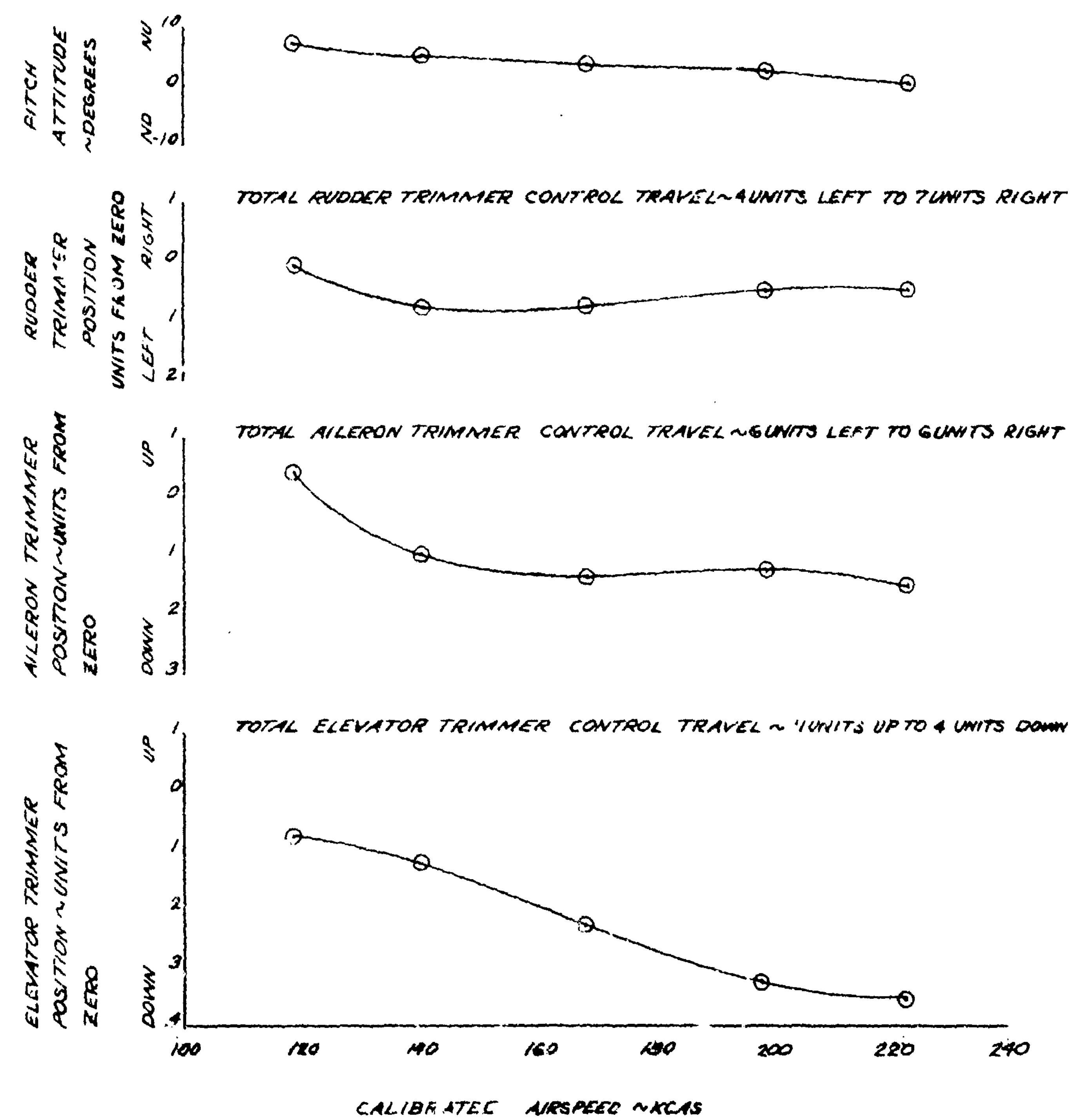


FIGURE 26

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	PROPELLER SPEED ~RPM	CONFIGURATION
14750	193.3(AFT)	10020	-3.0	2000	SINGLE ENGINE CRUISE

NOTE: LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED

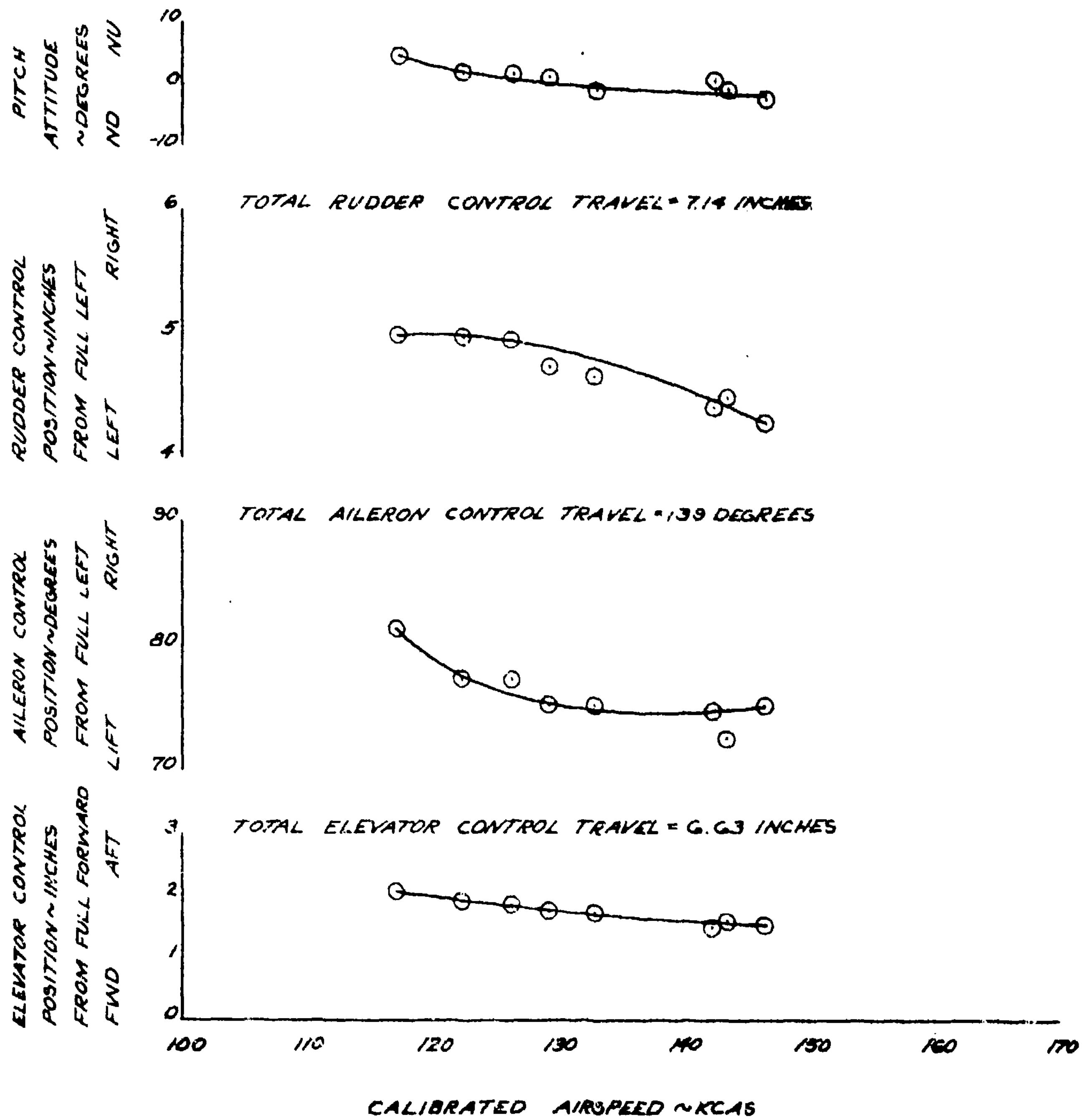


FIGURE 27
CONTROLLER CONTROL SURFACE RELATIONSHIP
BAC MODEL 200 USA S/N 71-21058

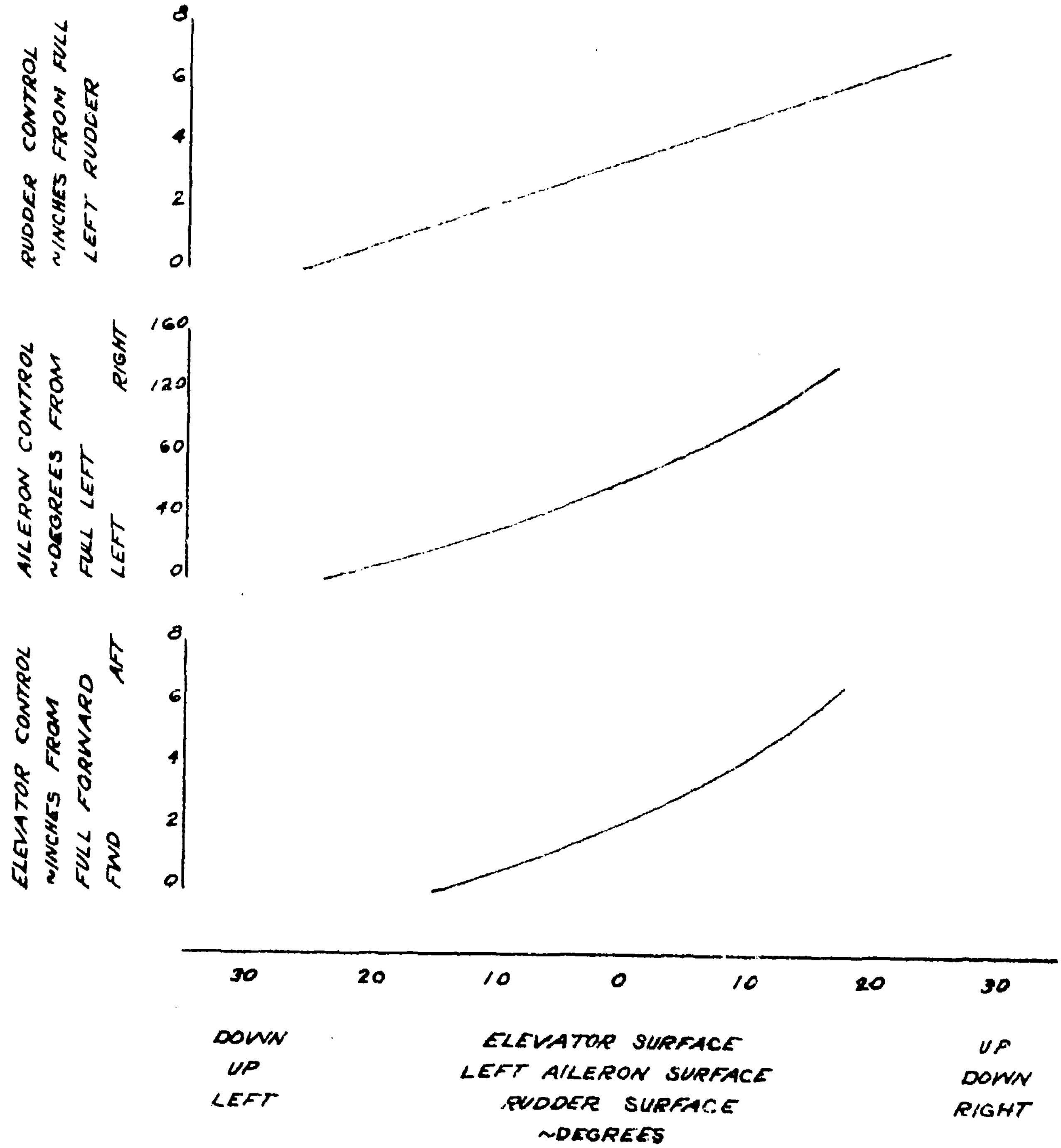


FIGURE 28

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
14670	187.6 (FWD)	2580	-13.0	CRUISE	LEVEL FLIGHT

NOTE: SHADED SYMBOLS DENOTE TRIM

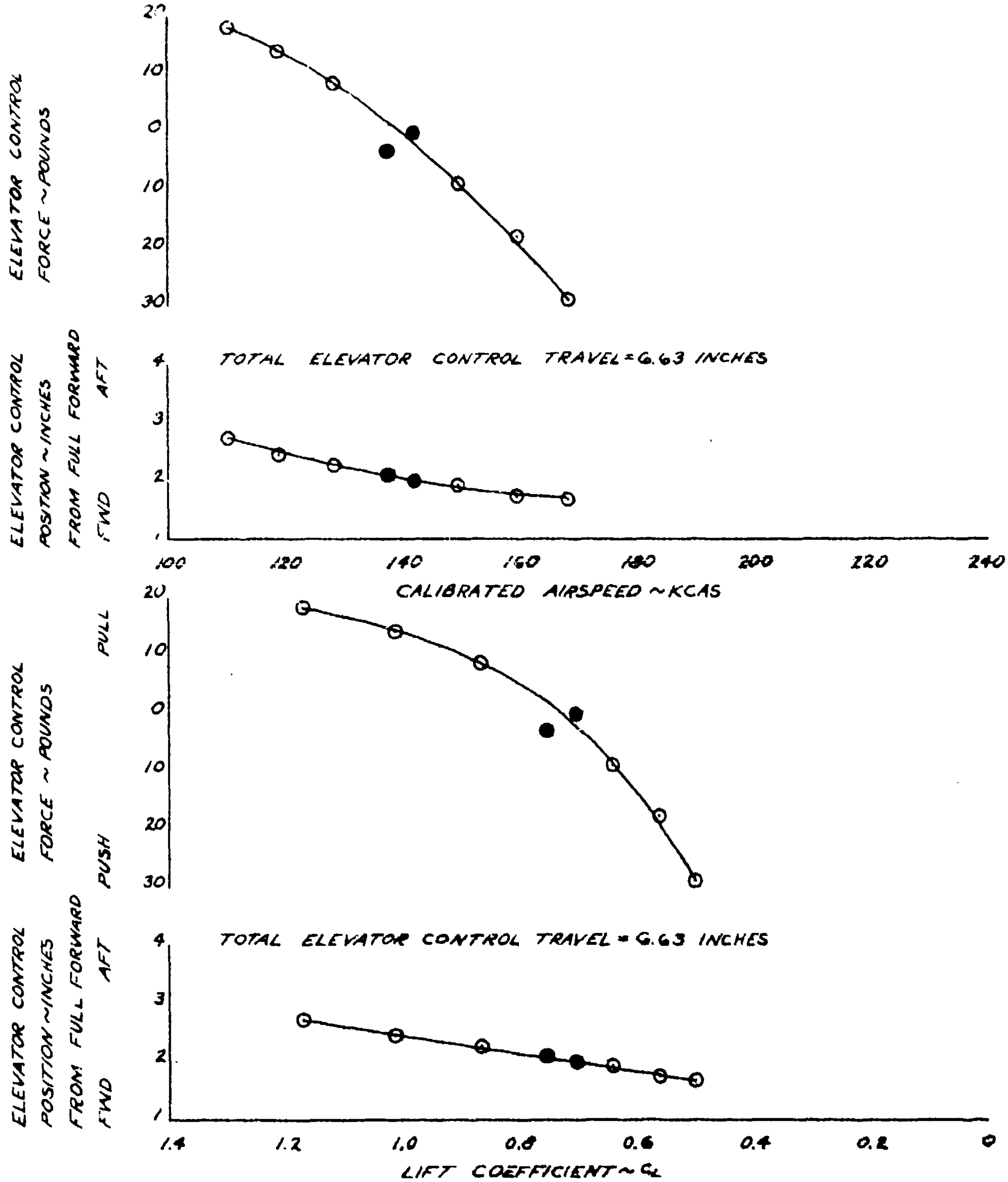


FIGURE 29

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 9'0 USA S/N 71-21068

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	A[illegible] DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14780	187.8 (FWD)	2660	-13.5	CRUISE	LEVEL FLIGHT

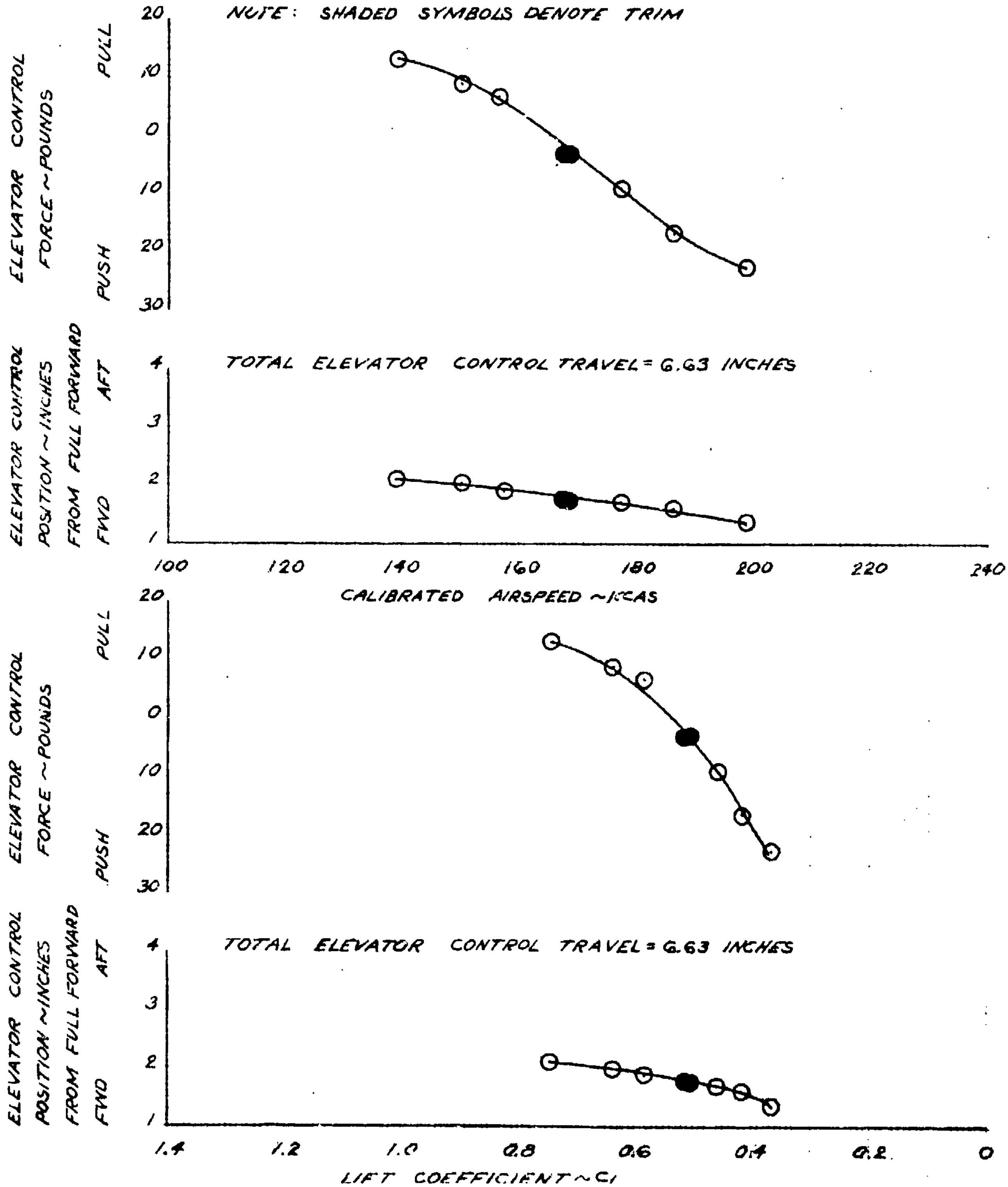


FIGURE 30

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL MODEL 200 USA 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14950	188.0(FWD)	2620	-14.0	CRUISE	LEVEL FLIGHT

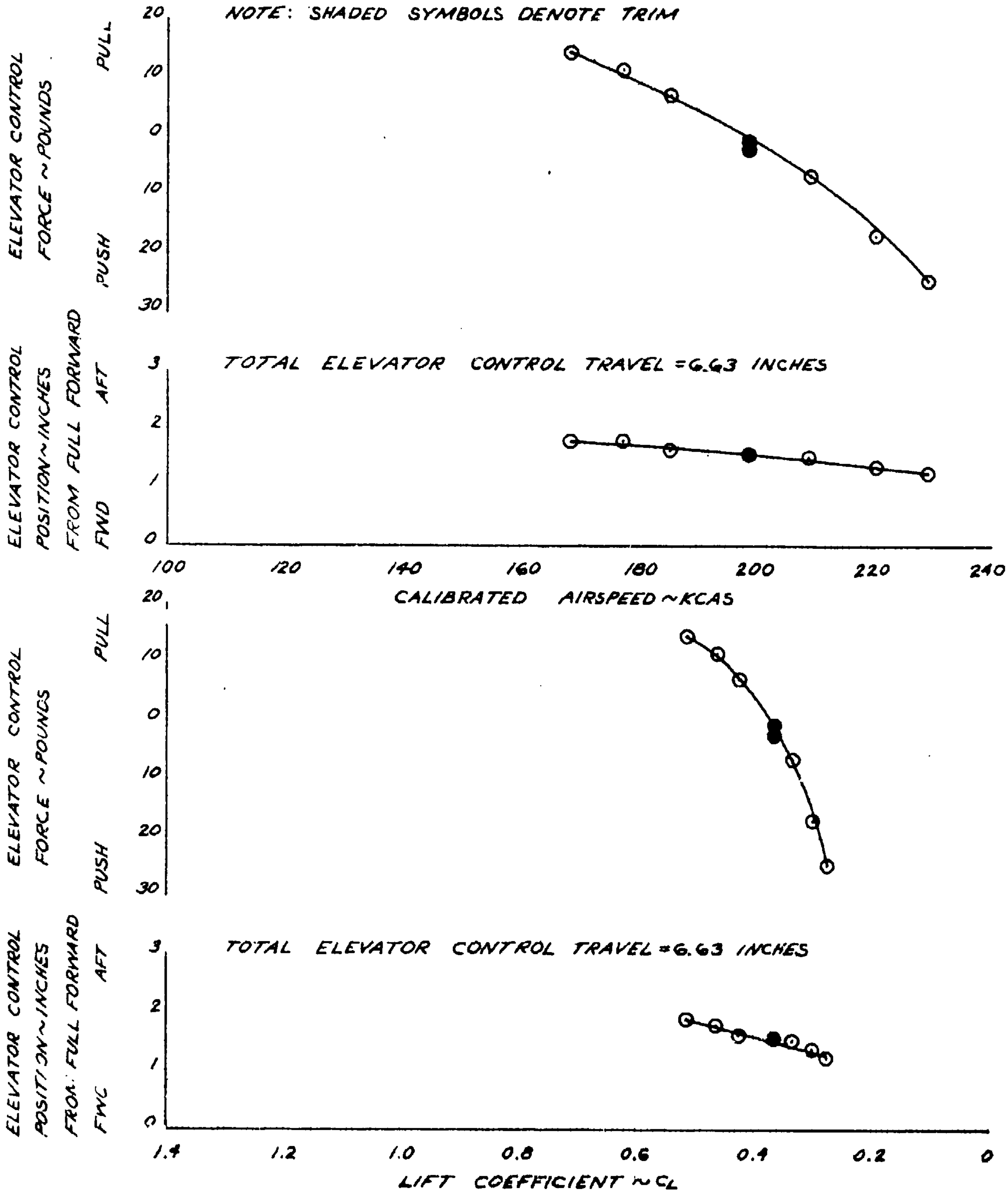


FIGURE 31

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
14560	187.5 (FWD)	3900	-4.0	POWER APPROACH	LEVEL FLIGHT

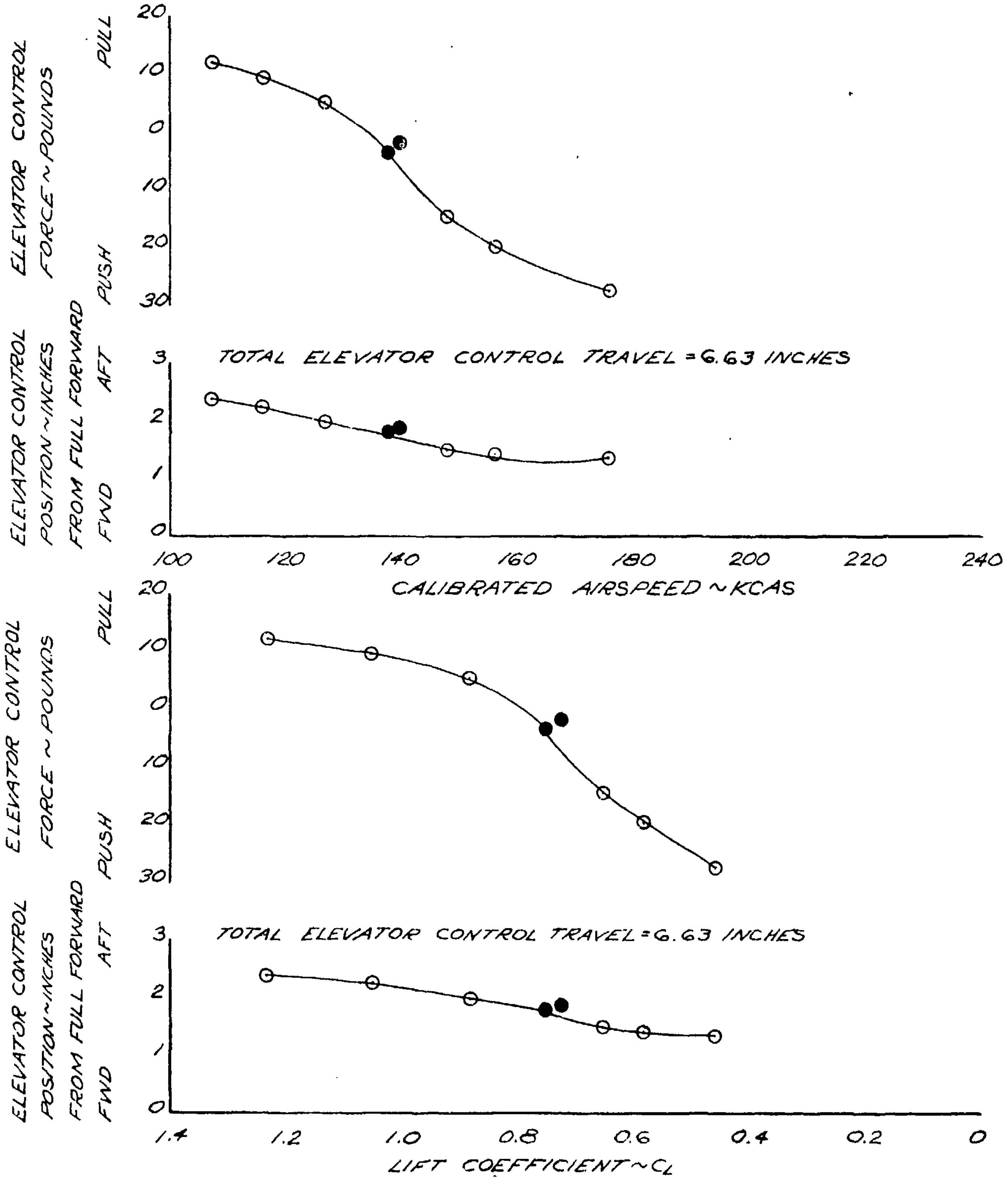


FIGURE 32

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
14780	197.2 (AFT)	5300	8.0	CRUISE	LEVEL FLIGHT

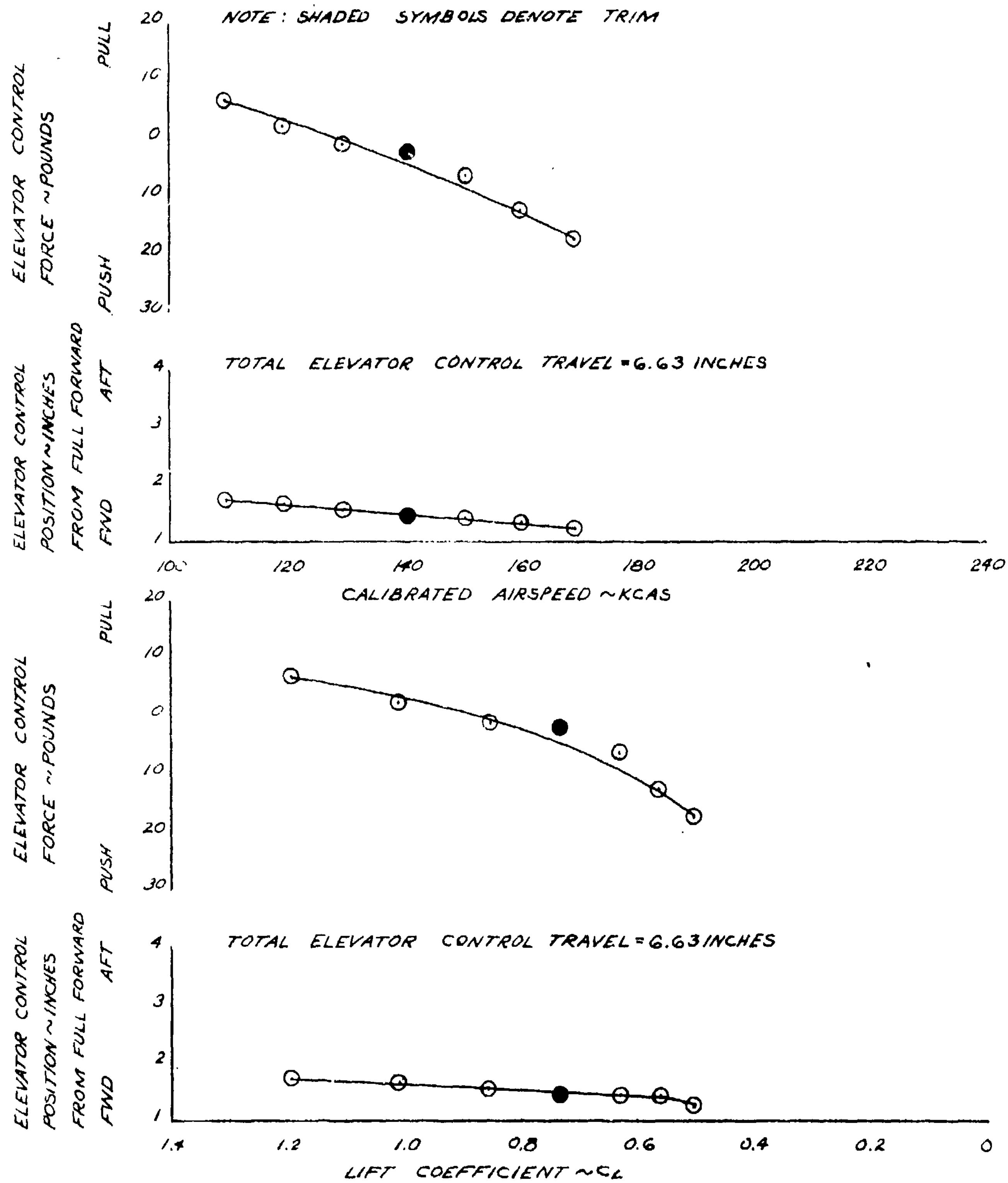


FIGURE 33

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
4880	197.3 (AFT)	5300	8.0	CRUISE	LEVEL FLIGHT

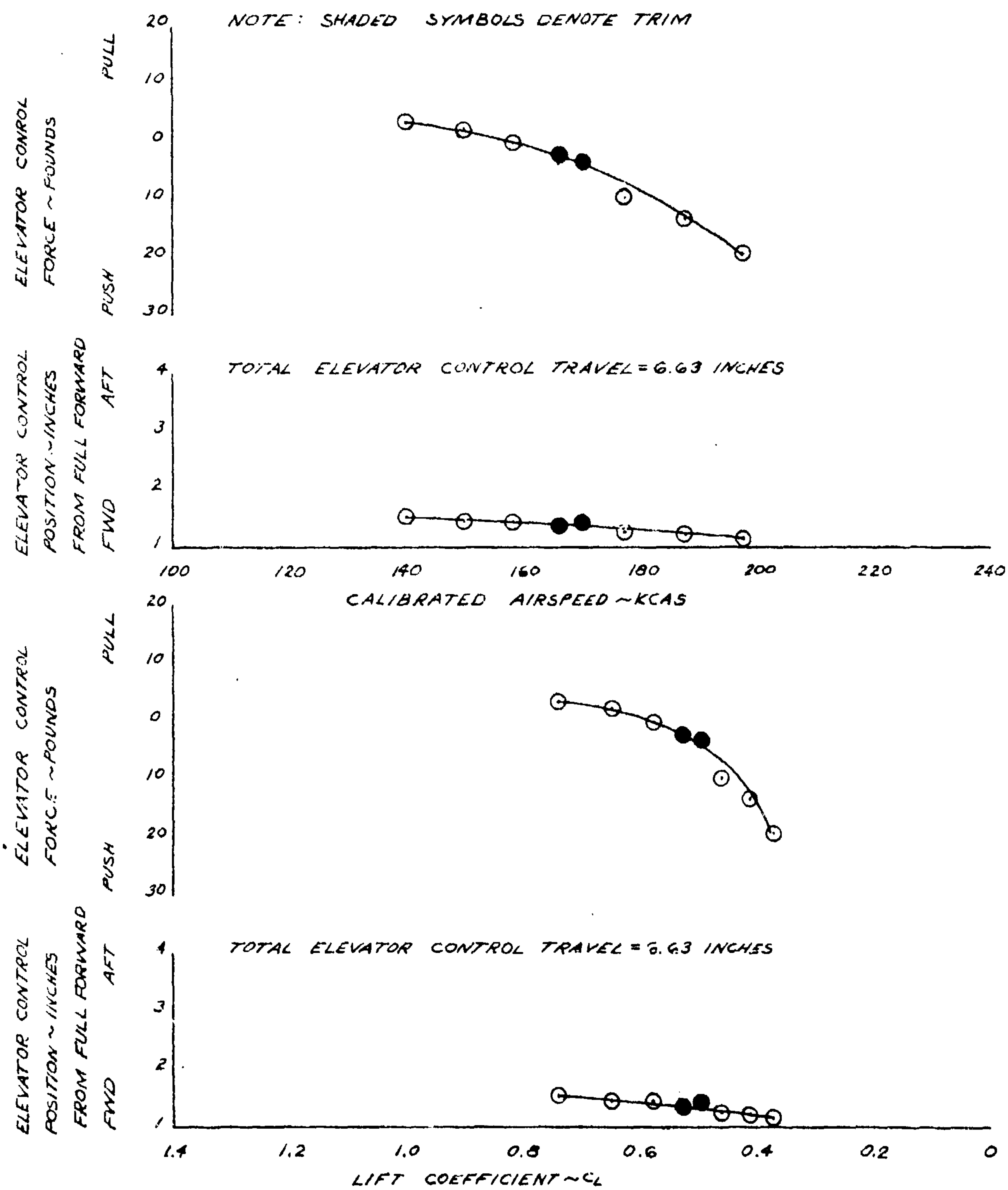


FIGURE 34

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
14990	197.4(AFT)	5120	6.5	CRUISE	LEVEL FLIGHT

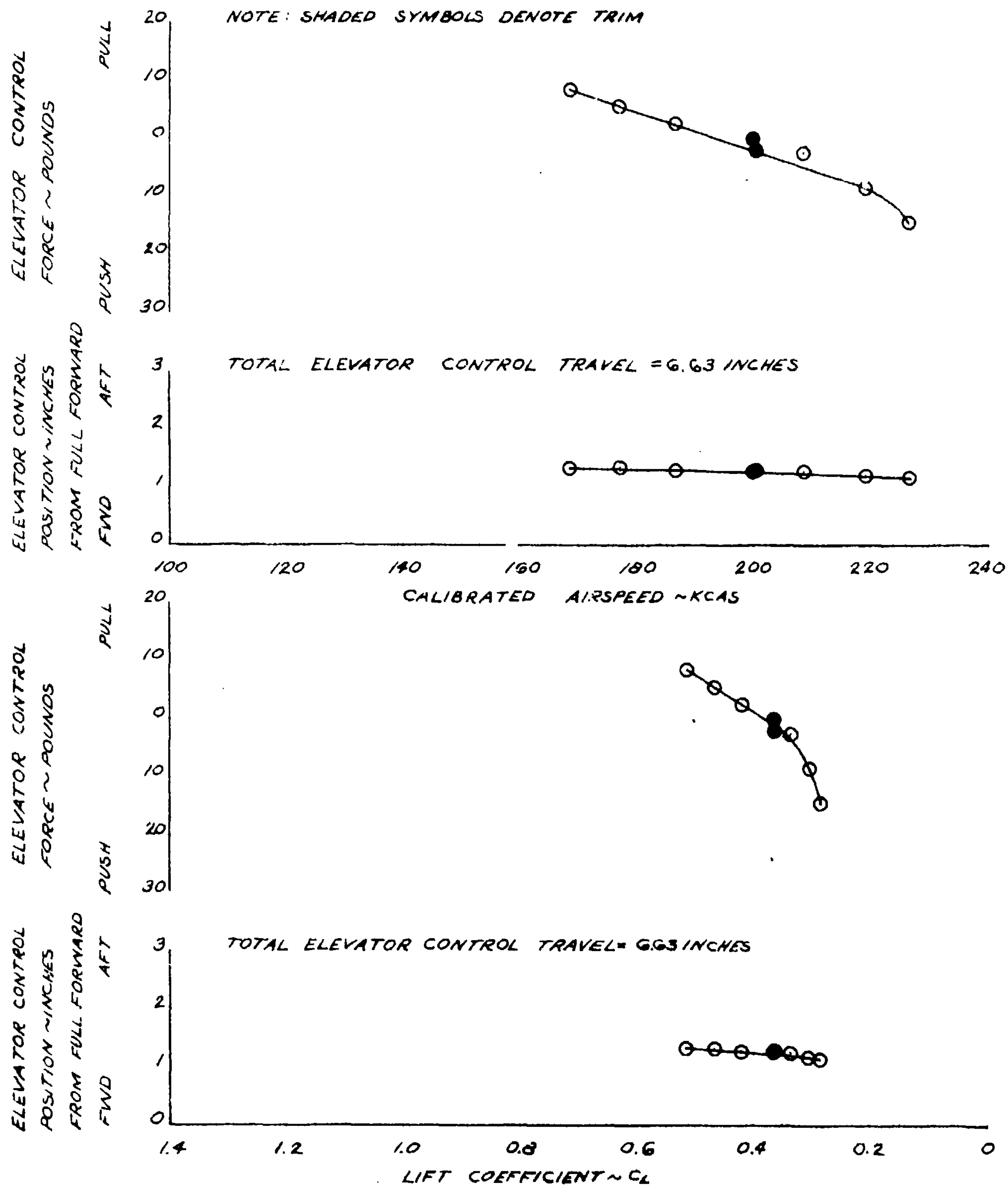


FIGURE 35

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
14630	197.2 (AFT)	5260	5	POWER APPROACH	LEVEL FLIGHT

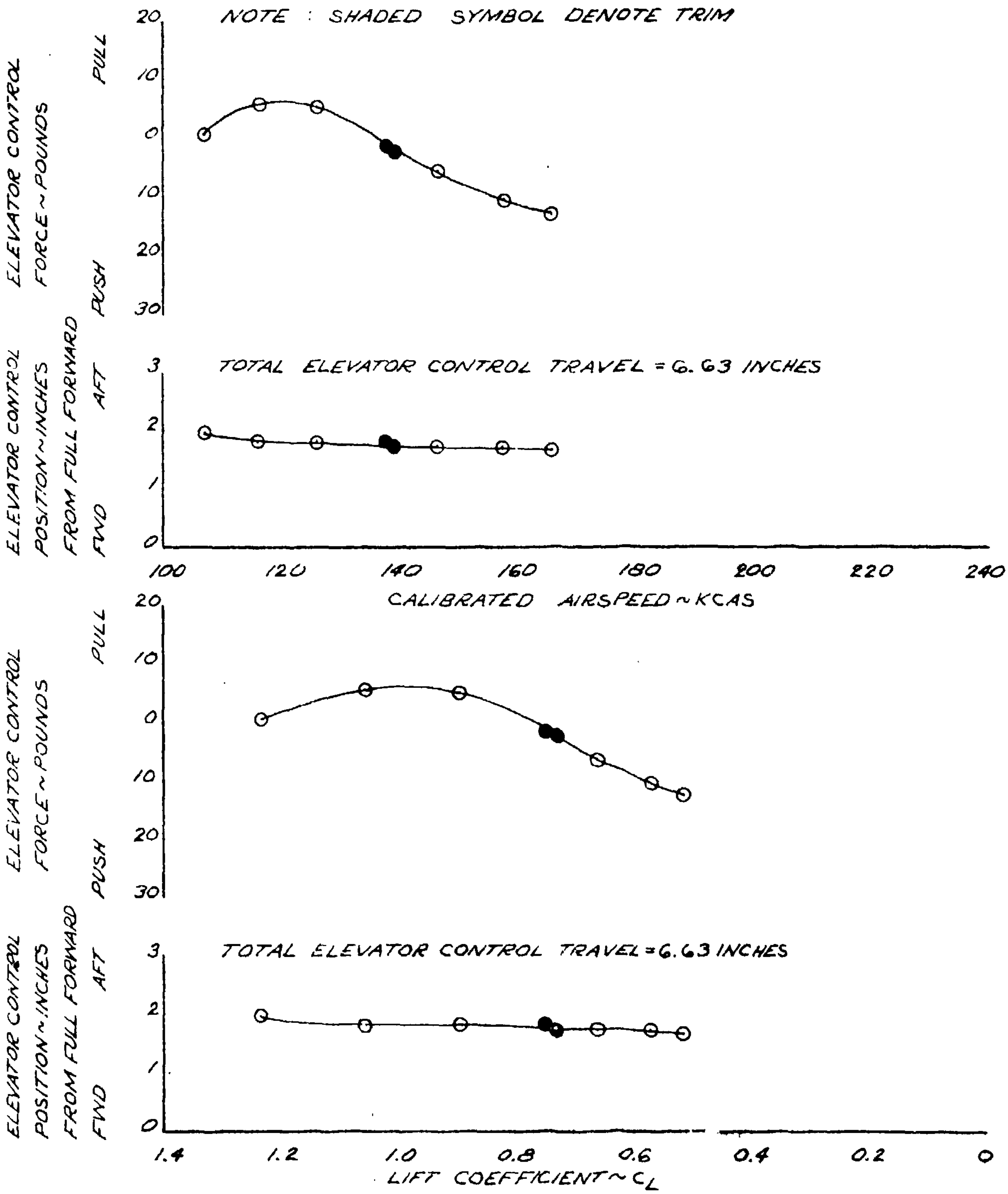


FIGURE 36

STATIC LATERAL-DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14330	197.2(AFT)	9840	-6.0	139	CRUISE	LEVEL FLIGHT

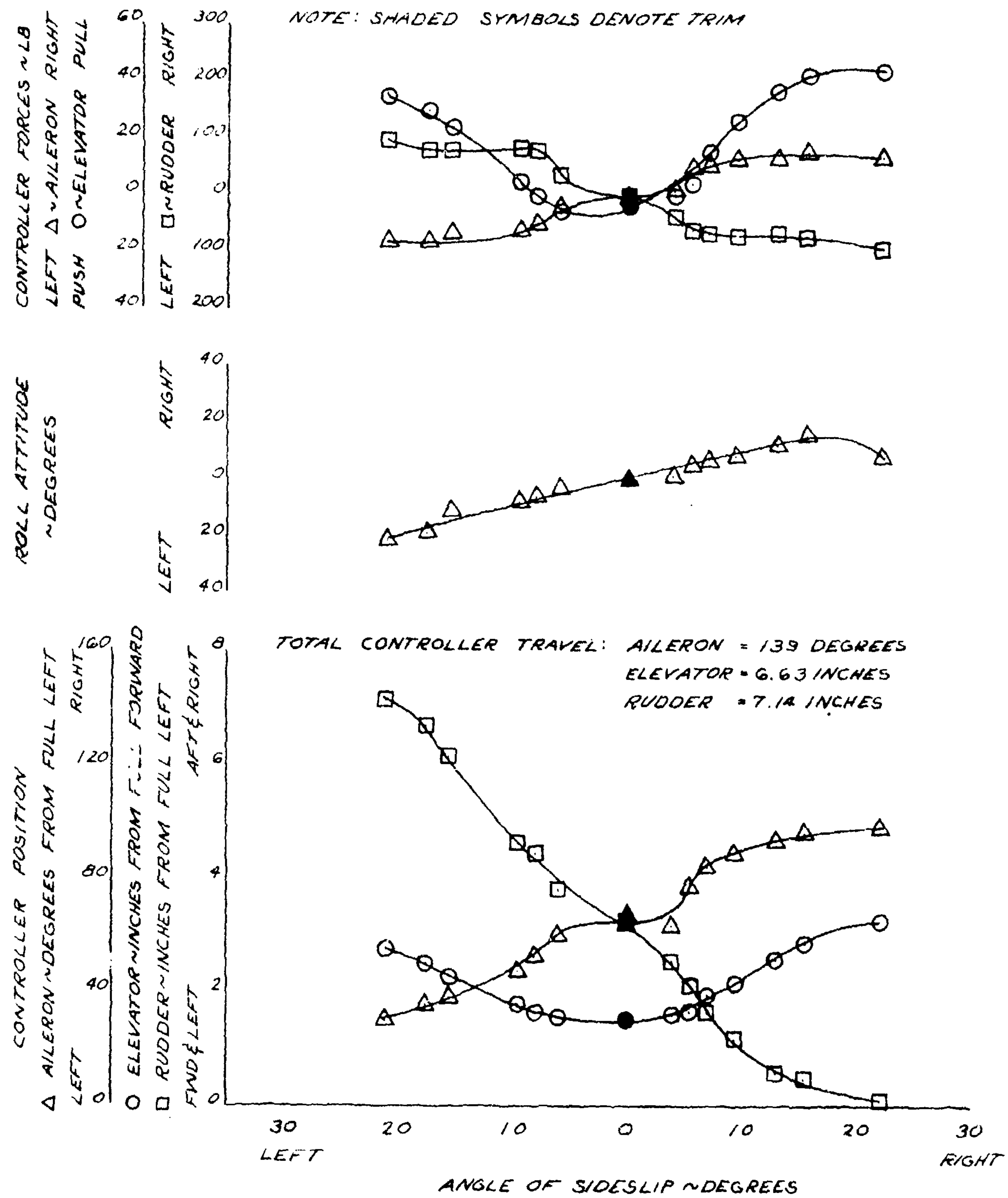


FIGURE 37

STATIC LATERAL-DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14780	197.2 (AFT)	9840	-6.0	168	CRUISE	LEVEL FLIGHT

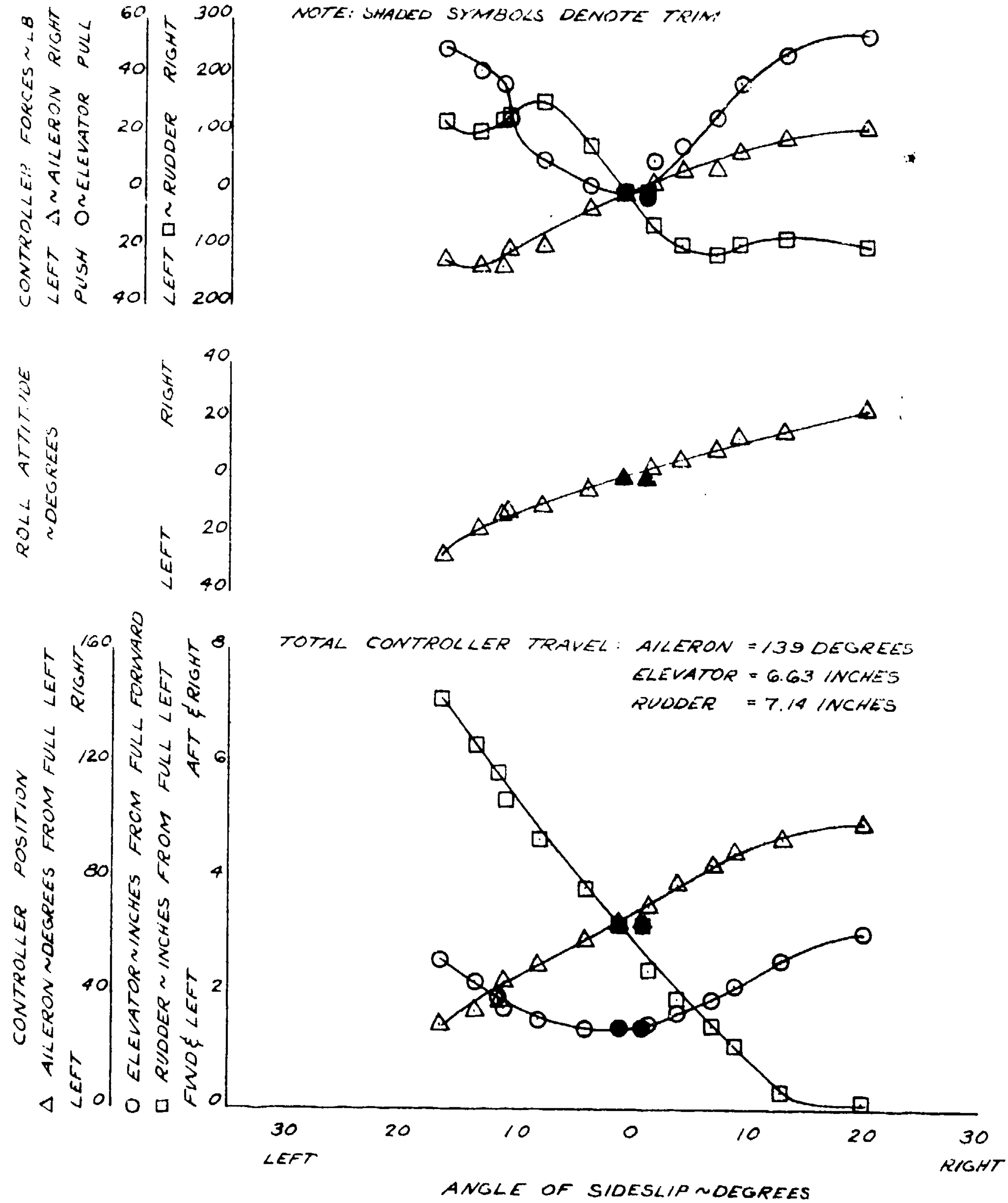


FIGURE 38

STATIC LATERAL-DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-21059

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14900	197.3(AFT)	10400	-6.0	196	CRUISE	LEVEL FLIGHT

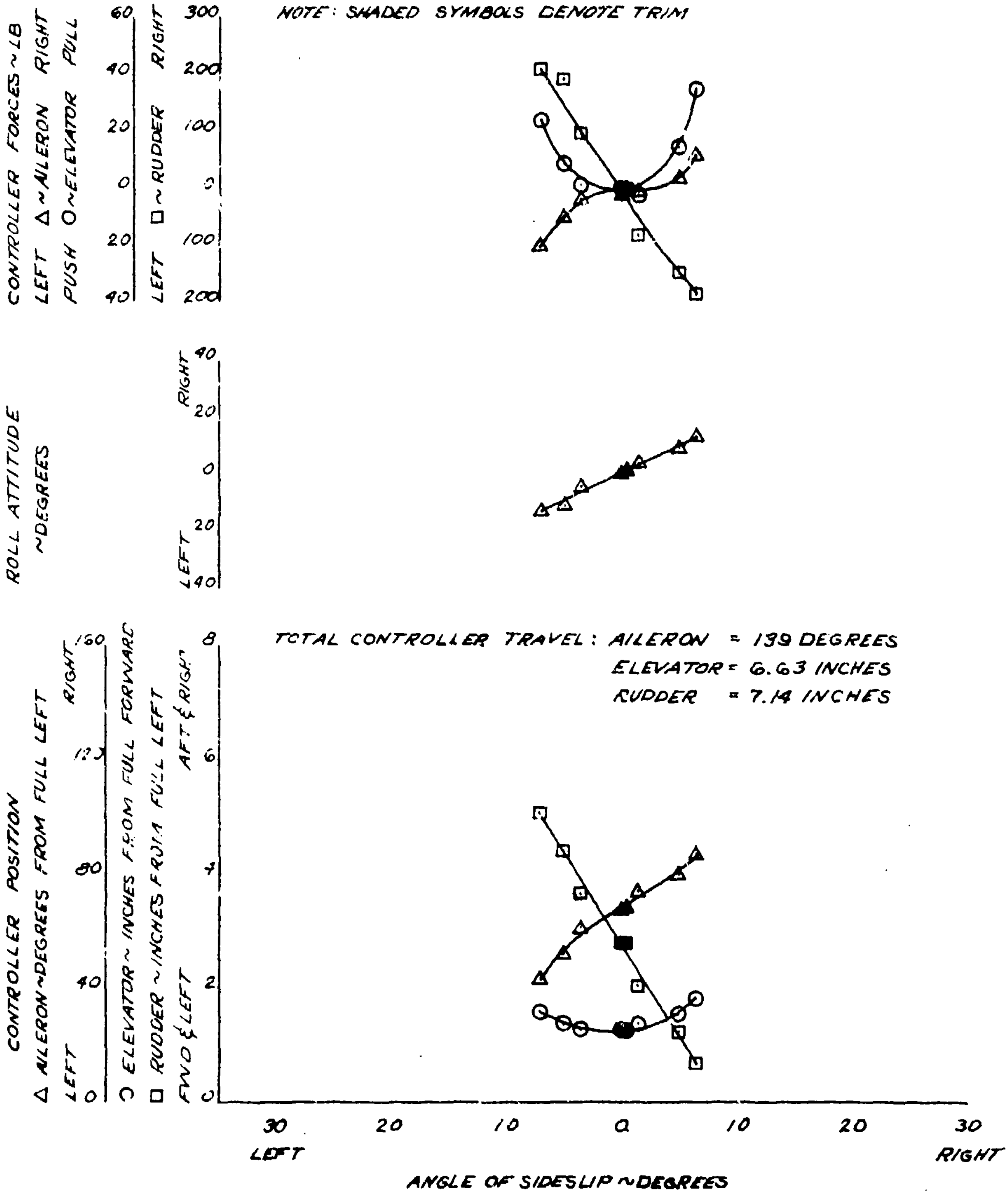


FIGURE 39

STATIC LATERAL-DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
13970	197.1 (AFT)	9840	-6.0	138	POWER APPROACH	LEVEL FLIGHT

NOTE: SHADED SYMBOLS DENOTE TRIM

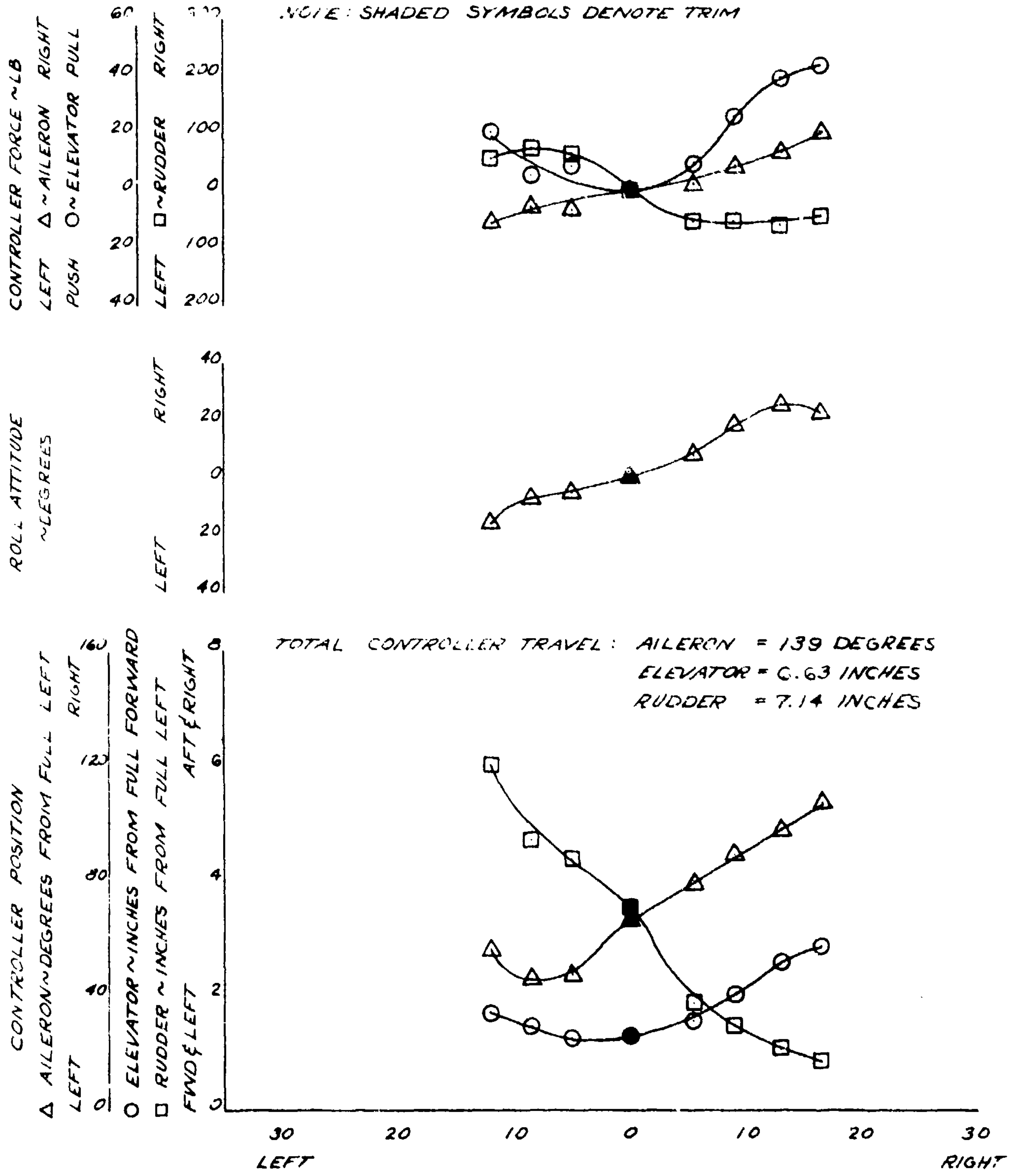


FIGURE 40

STATIC LATERAL-DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14190	197.2 (AFT)	10600	0.0	165	POWER APPROACH	LEVEL FLIGHT

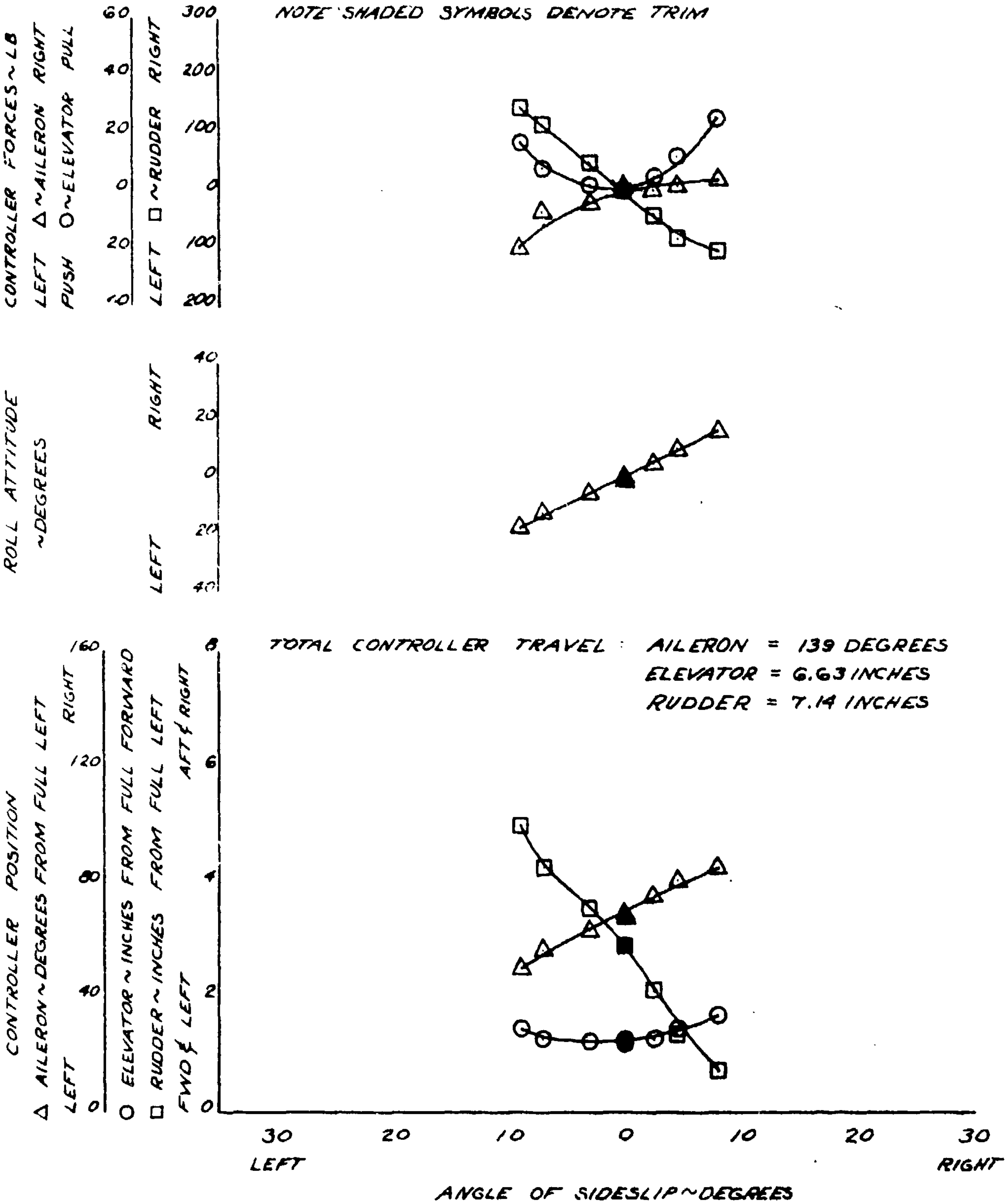


FIGURE 41

DYNAMIC LONGITUDINAL STABILITY (SHORT PERIOD RESPONSE)

BAC MODEL 200 USA S/N 71-21058

GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14080	197.0 (AFT)	10860	2.5	CRUISE	LEVEL FLIGHT

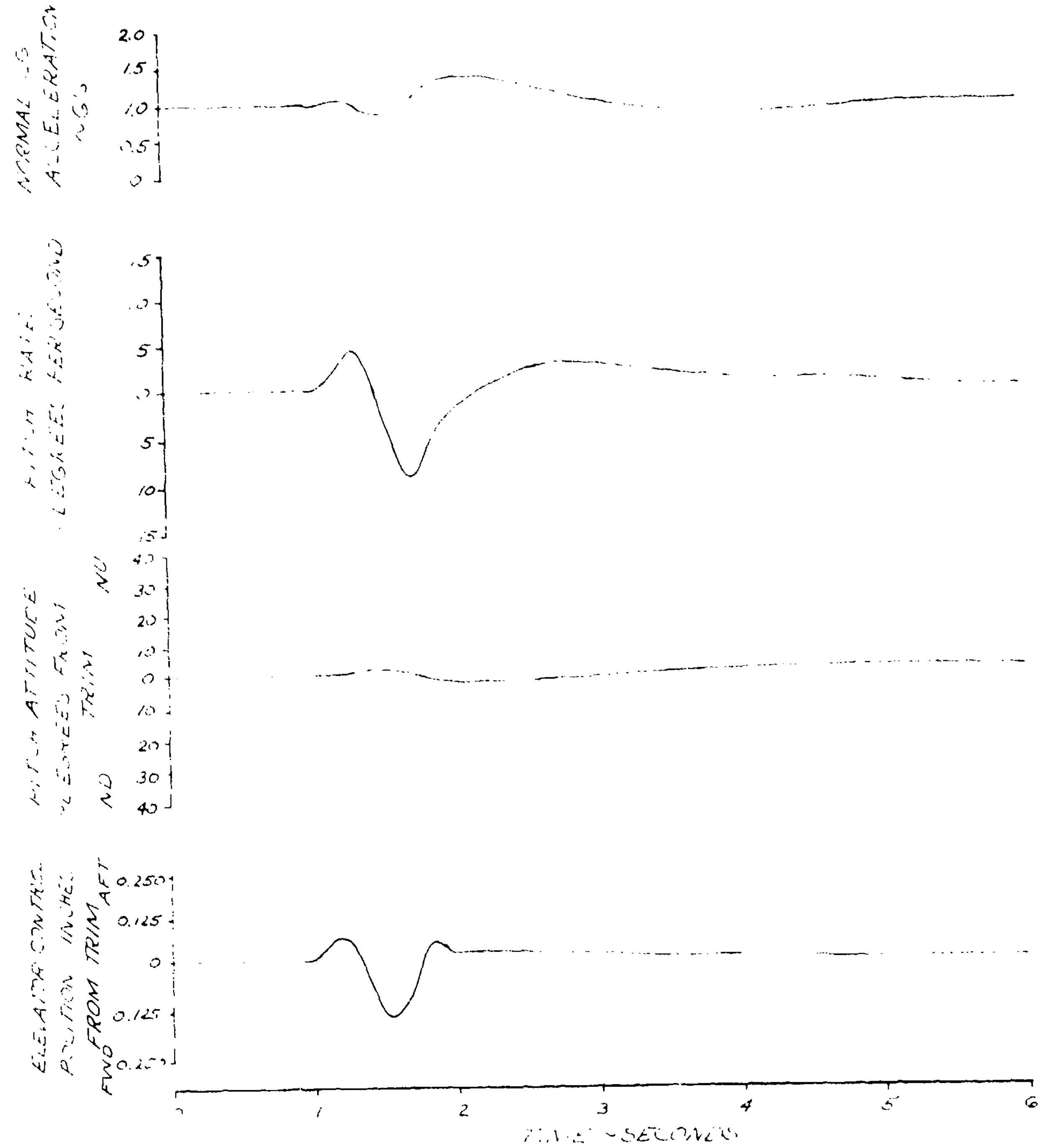


FIGURE 42
DYNAMIC LONGITUDINAL STABILITY (PHUGOID)
BAC MODEL 200 USA S/N 71-21058

TRIM AIRSPEED ~KCAS	GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	CONFIGURATION	FLIGHT CONDITION
198	14440	197.1 (AFT)	10680	0	CRUISE	LEVEL FLIGHT
166	13960	197.0 (AFT)	10420	-1.0	CRUISE	LEVEL FLIGHT
142	14130	197.1 (AFT)	10860	2.5	CRUISE	LEVEL FLIGHT

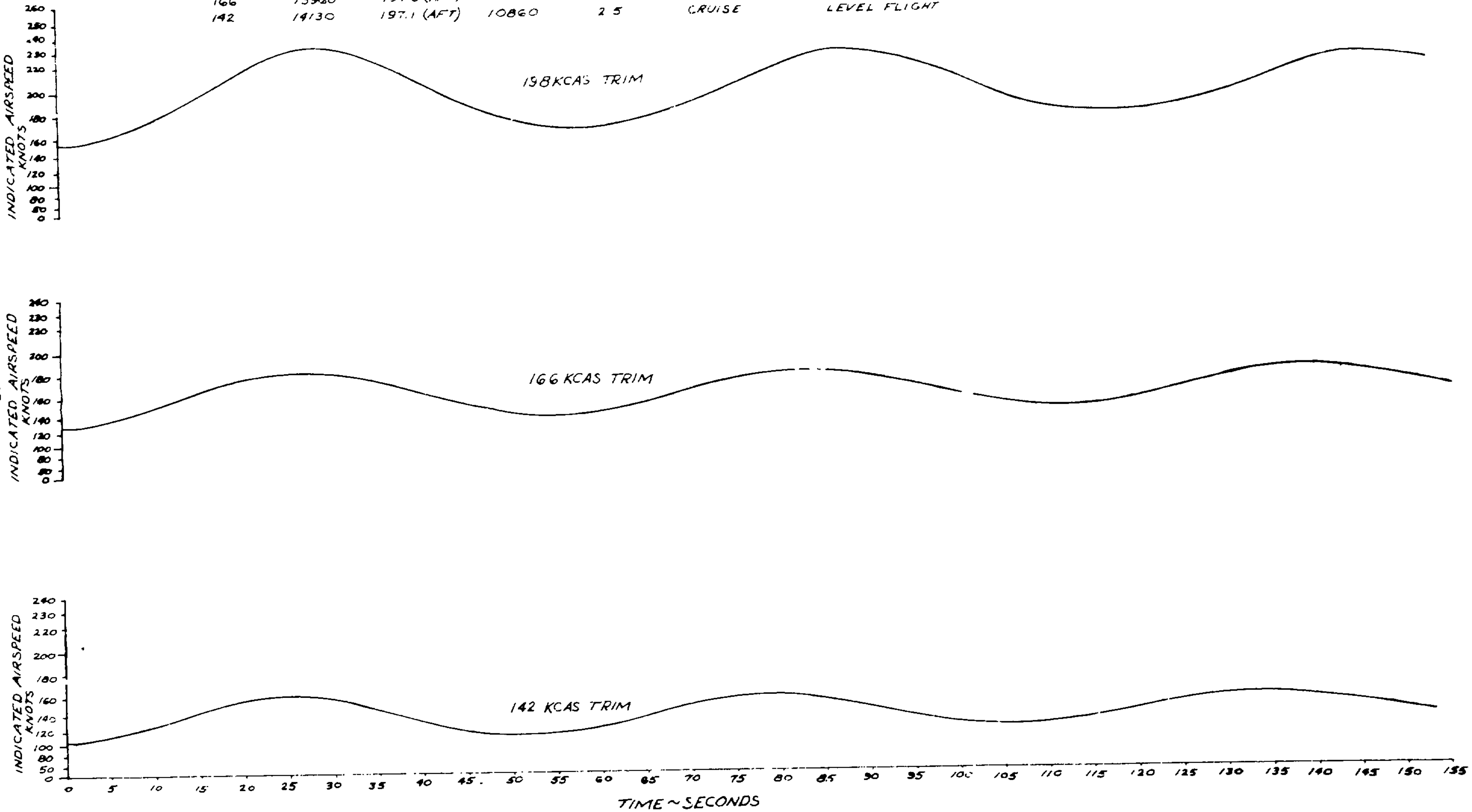


FIGURE 43
AIRCRAFT RESPONSE FOLLOWING A RUDDER DOUBLET
BAC MODEL 200 USA S/N 71-21058

GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14220	197.0(AFT)	11040	3.5	140	CRUISE	LEVEL FLIGHT

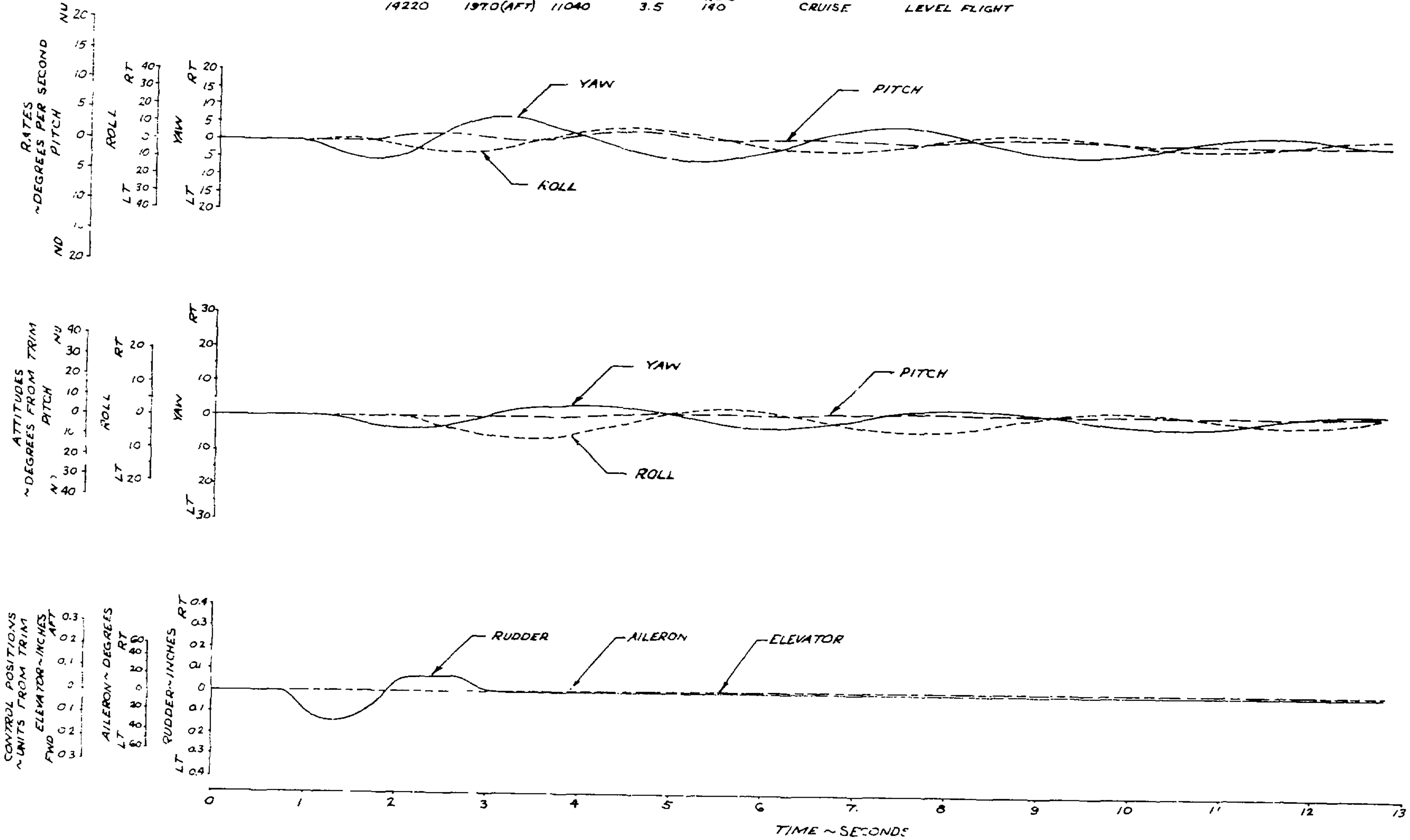


FIGURE 44
AIRCRAFT RESPONSE FOLLOWING A RUDDER DOUBLET
BAC MODEL 200 USA S/N 71-21058

GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14480	197.1 (AFT)	10960	3.5	168	CRUISE	LEVEL FLIGHT

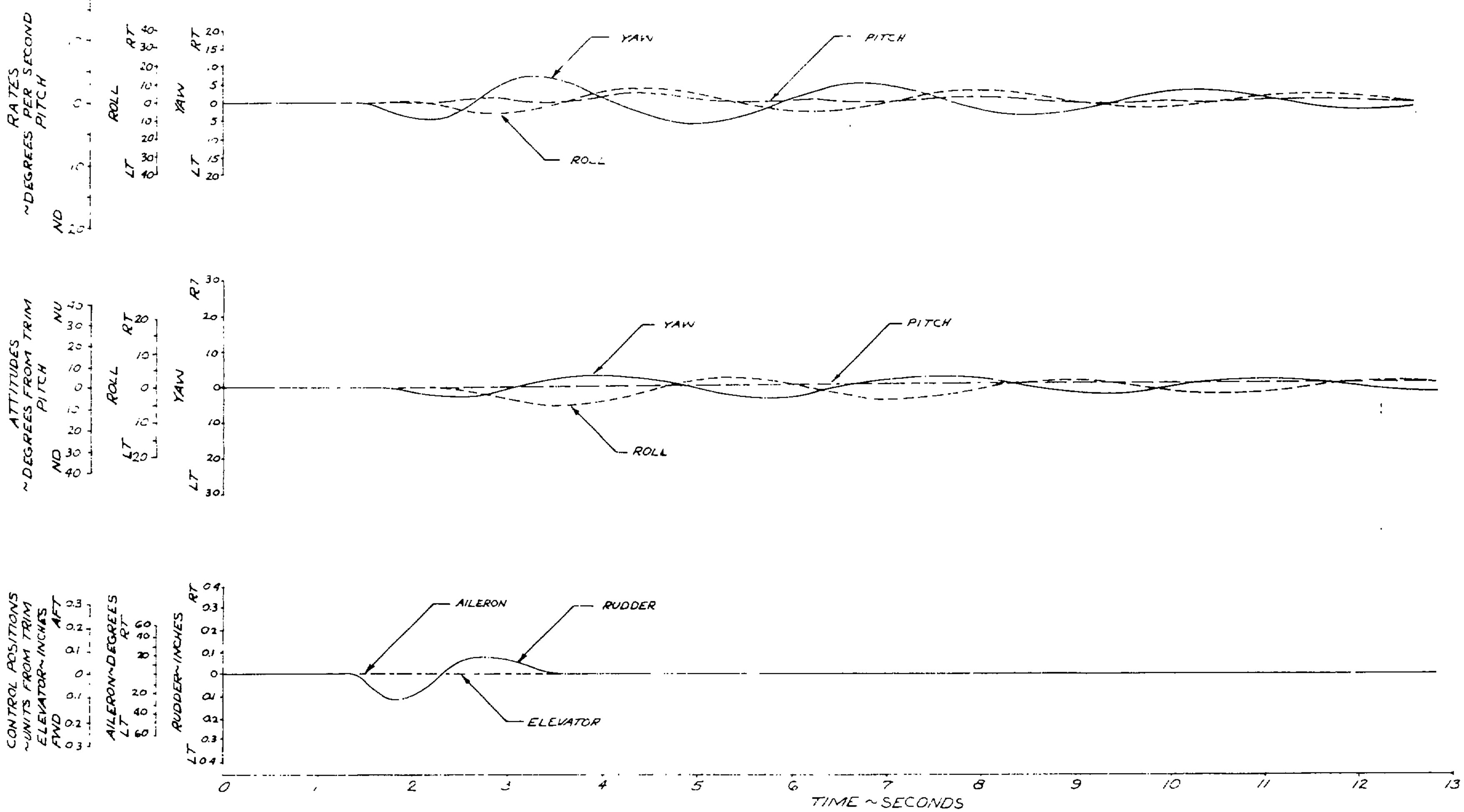


FIGURE 45

AIRCRAFT RESPONSE FOLLOWING A RUDDER DOUBLET
BAC MODLE 200 USA S/N 71-21058

GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14800	197.2(AFT)	11000	4.5	199	CRUISE	LEVEL FLIGHT

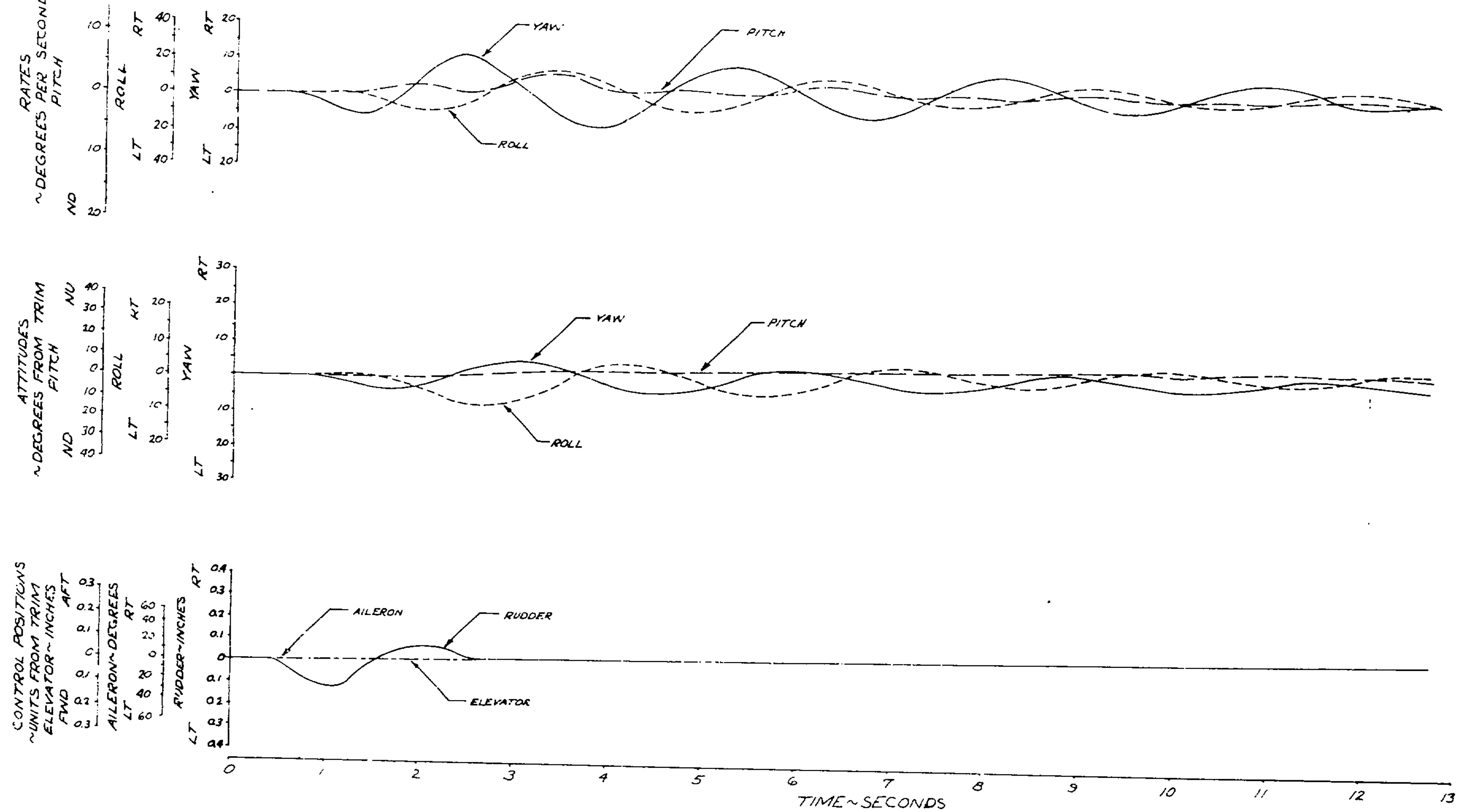


FIGURE 46
AIRCRAFT RESPONSE FOLLOWING AN AILERON DOUBLET
BAC MODEL 200 USA S/N 71-21058

GROSS WEIGHT ~LB	CG LOCATION ~IN	DENSITY ALTITUDE ~FT	OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
14460	197.1 (AFT)	10900	3.0	168	CRUISE	LEVEL FLIGHT

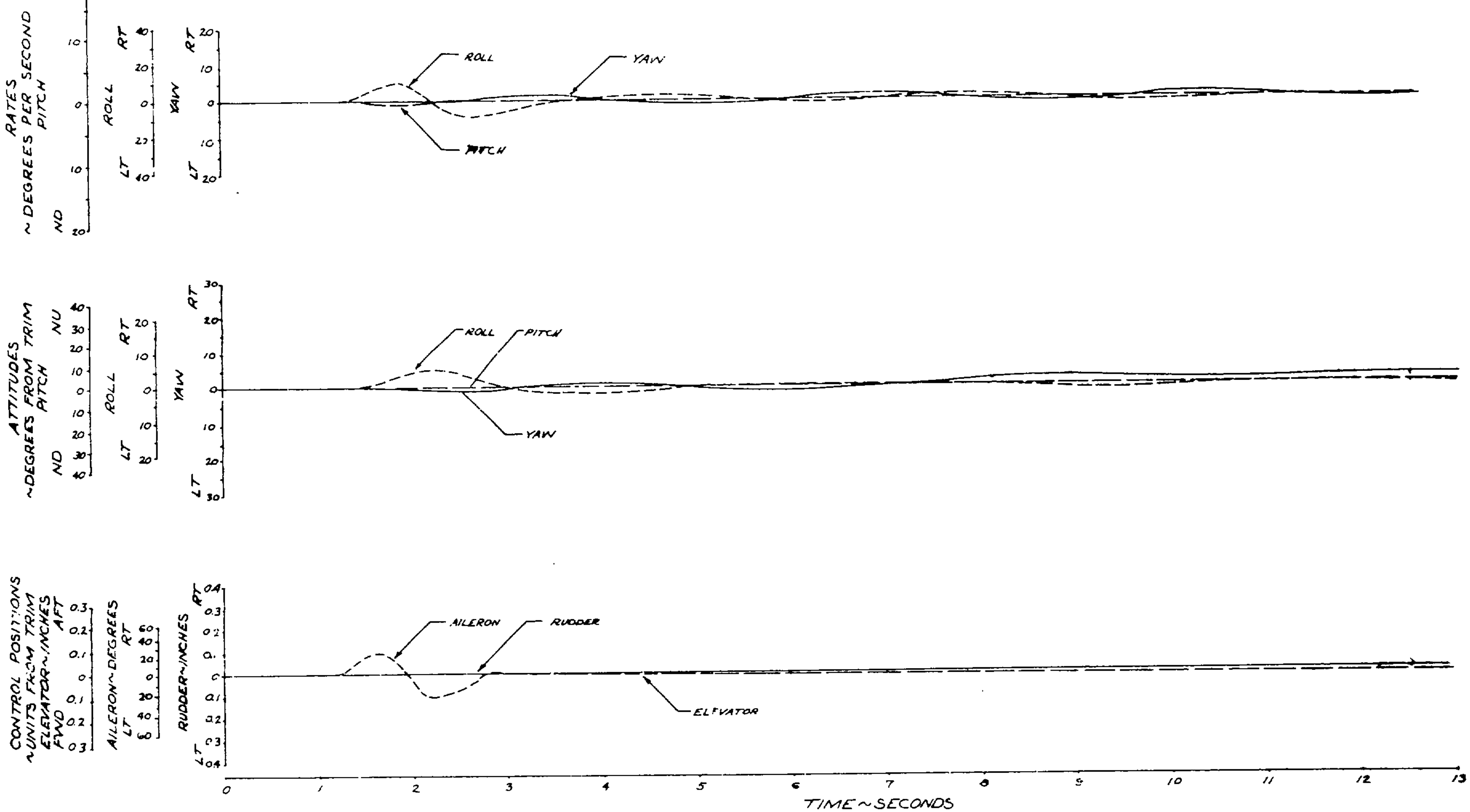


FIGURE 47
MANEUVERING STABILITY
BAC MODEL 200 USA S/N 71-21058

SYM	TRIM AIRSPEED ~KCAS	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	MANEUVER
○	169	14420	197.1 (AFT)	10340	-2.0	CRUISE	LEFT WIND-UP TURN
□	169	14320	197.1 (AFT)	10340	-2.0	CRUISE	RIGHT WIND-UP TURN
△	169	13900	197.1 (AFT)	10720	4.0	CRUISE	SYMMETRICAL PULL-UP
◇	172	14280	197.1 (AFT)	10340	-2.0	CRUISE	SYMMETRICAL PUSH-OVER

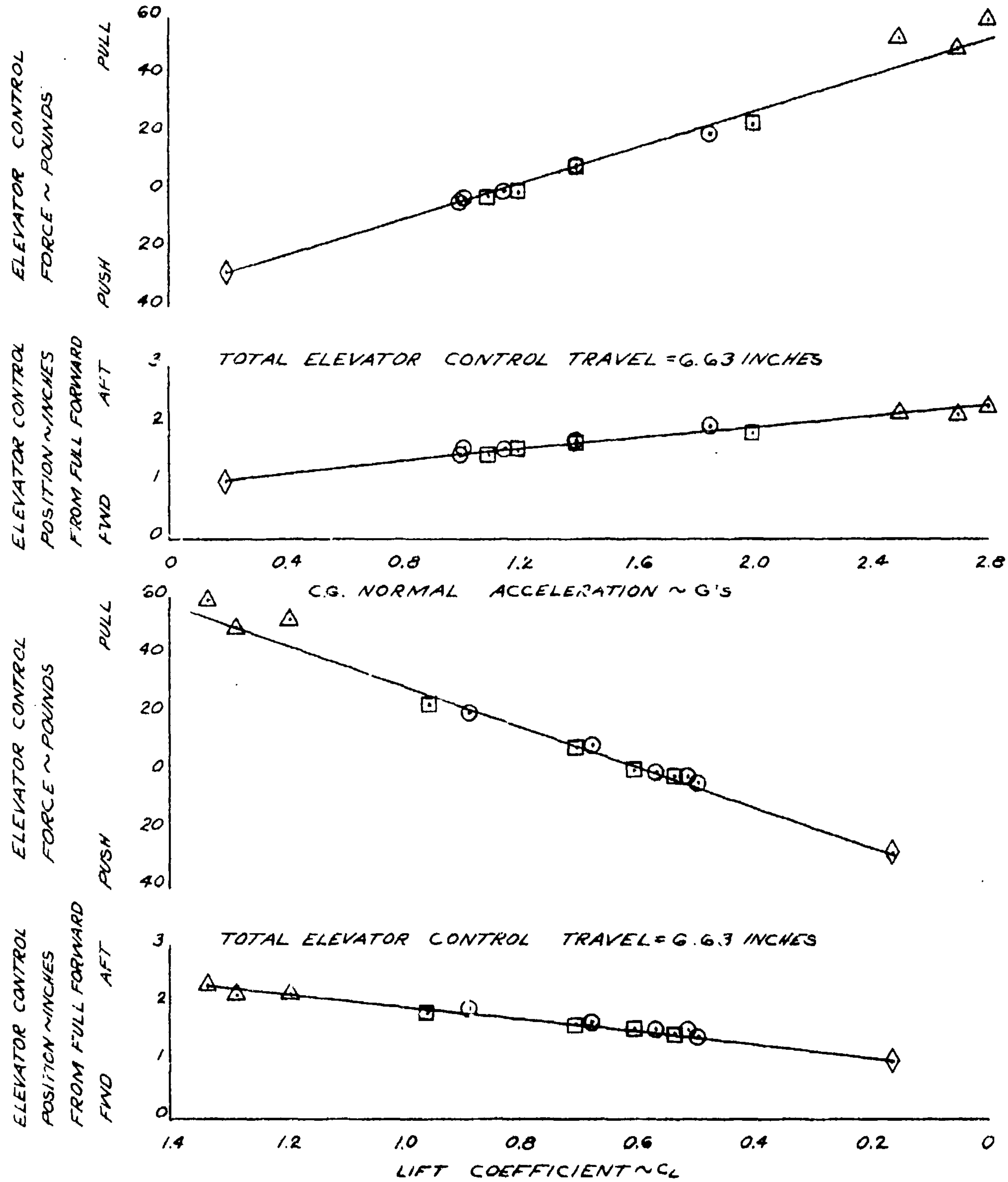


FIGURE 48
MANEUVERING STABILITY
BAC MODEL 200 USA S/N 71-21058

SYM	TRIM AIRSPEED ~KCAS	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	MANEUVER
○	198	14720	197.2 (AFT)	10340	-2.0	CRUISE	LEFT WIND-UP TURN
□	198	14620	197.2 (AFT)	10340	-2.0	CRUISE	RIGHT WIND-UP TURN
△	202	14850	197.3 (AFT)	10900	4.0	CRUISE	SYMMETRICAL PULL-UP
◇	198	14550	197.1 (AFT)	10340	-2.0	CRUISE	SYMMETRICAL PUSH-OVER

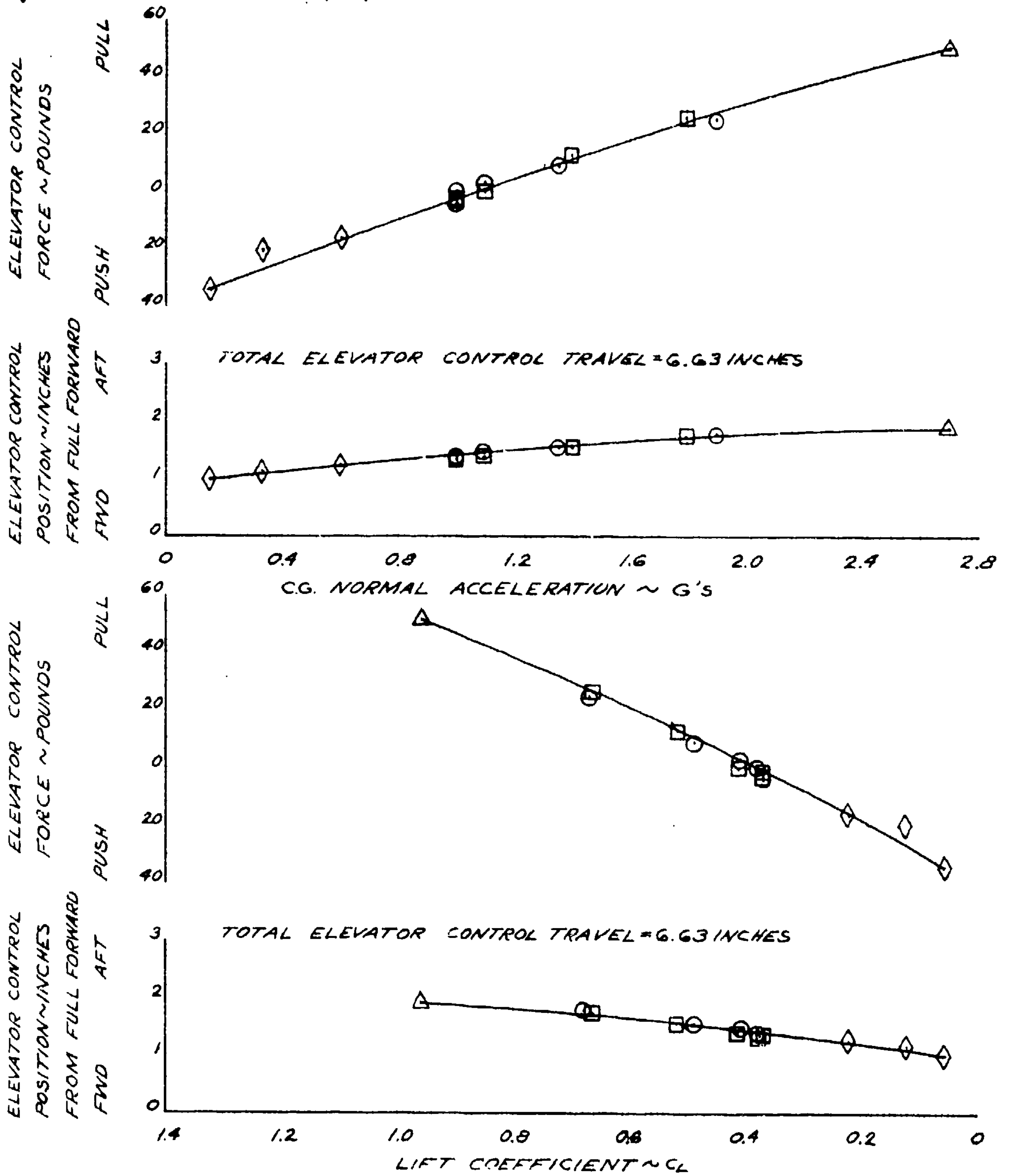


FIGURE 49
MANEUVERING STABILITY
BAC MODEL 200 USA S/N 71-21058

SYM	TRIM AIRSPEED ~KCAS	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	MANEUVER
○	142	14120	197.0 (AFT)	10340	-2.0	CRUISE	LEFT WIND-UP TURN
□	140	14020	197.1 (AFT)	10340	-2.0	CRUISE	RIGHT WIND-UP TURN
◇	140	13980	197.1 (AFT)	10340	-2.0	CRUISE	SYMMETRICAL PUSH-OVER

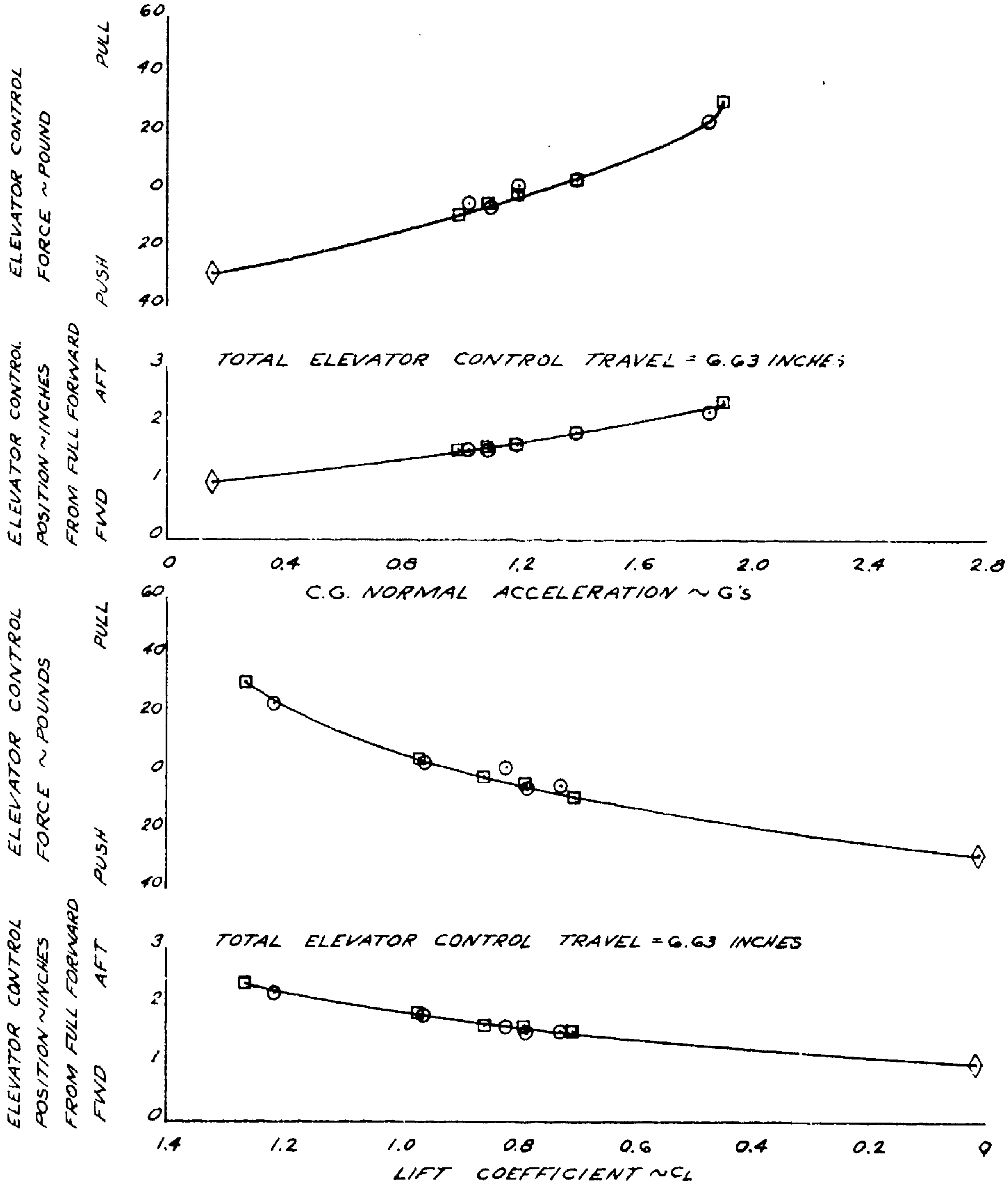


FIGURE 50
ROLL PERFORMANCE
BAC MODEL 200 USA S/N 71-21058

SYM	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	ROLL DIRECTION	CONFIGURATION	FLIGHT CONDITION
○	14560	197.4 (AFT)	11020	4.5	LEFT	CRUISE	LEVEL FLIGHT
□	14960	197.4 (AFT)	11020	4.5	RIGHT	CRUISE	LEVEL FLIGHT

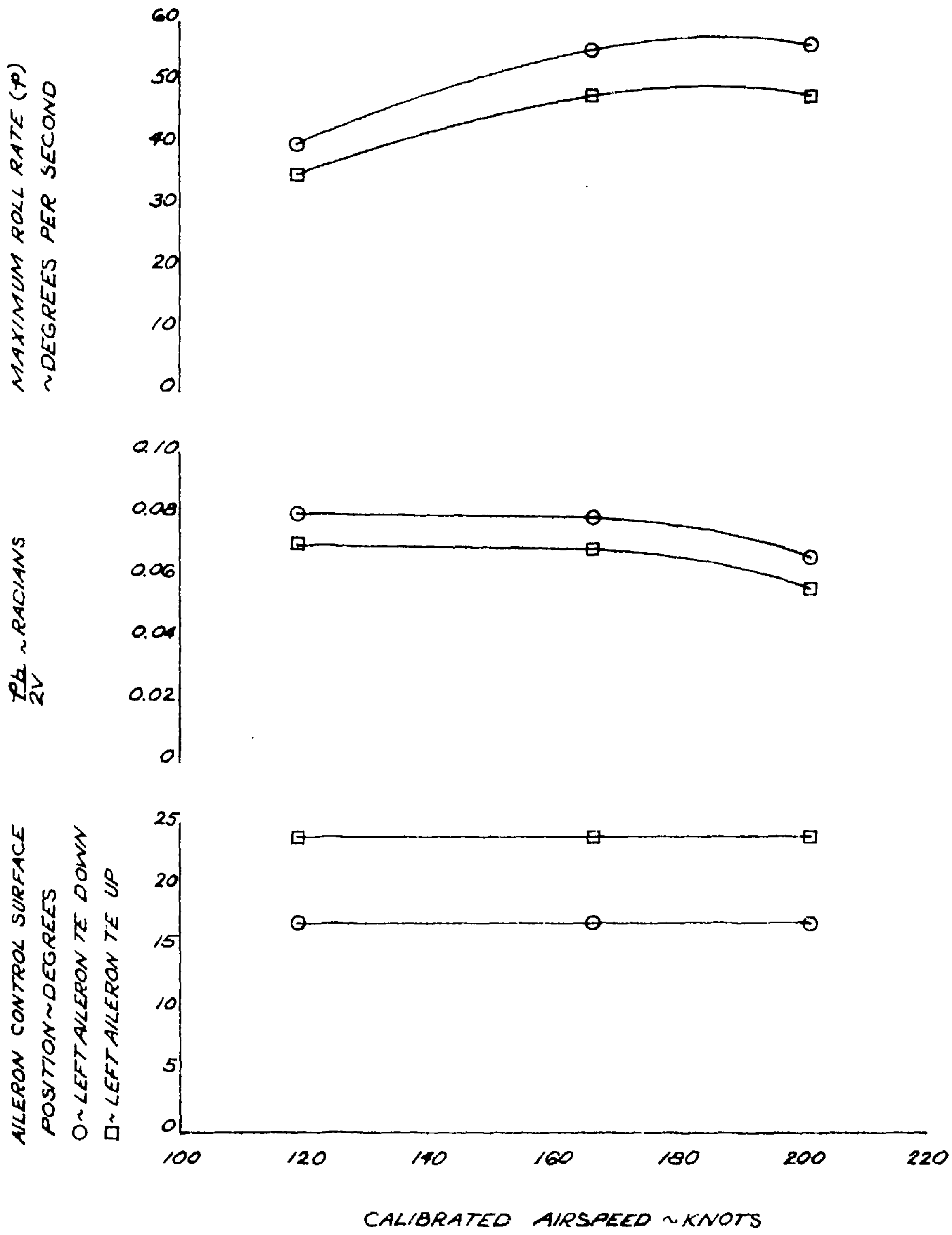


FIGURE 51

ROLL PERFORMANCE

BAC MODEL 20C USA S/N 71-21058

SYM	AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	ROLL DIRECTION	CONFIGURATION	FLIGHT CONDITION
○	13820	197.2 (AFT)	10680	1.0	LEFT	POWER APPROACH	LEVEL FLIGHT
□	13820	197.2 (AFT)	10580	1.0	RIGHT	POWER APPROACH	LEVEL FLIGHT

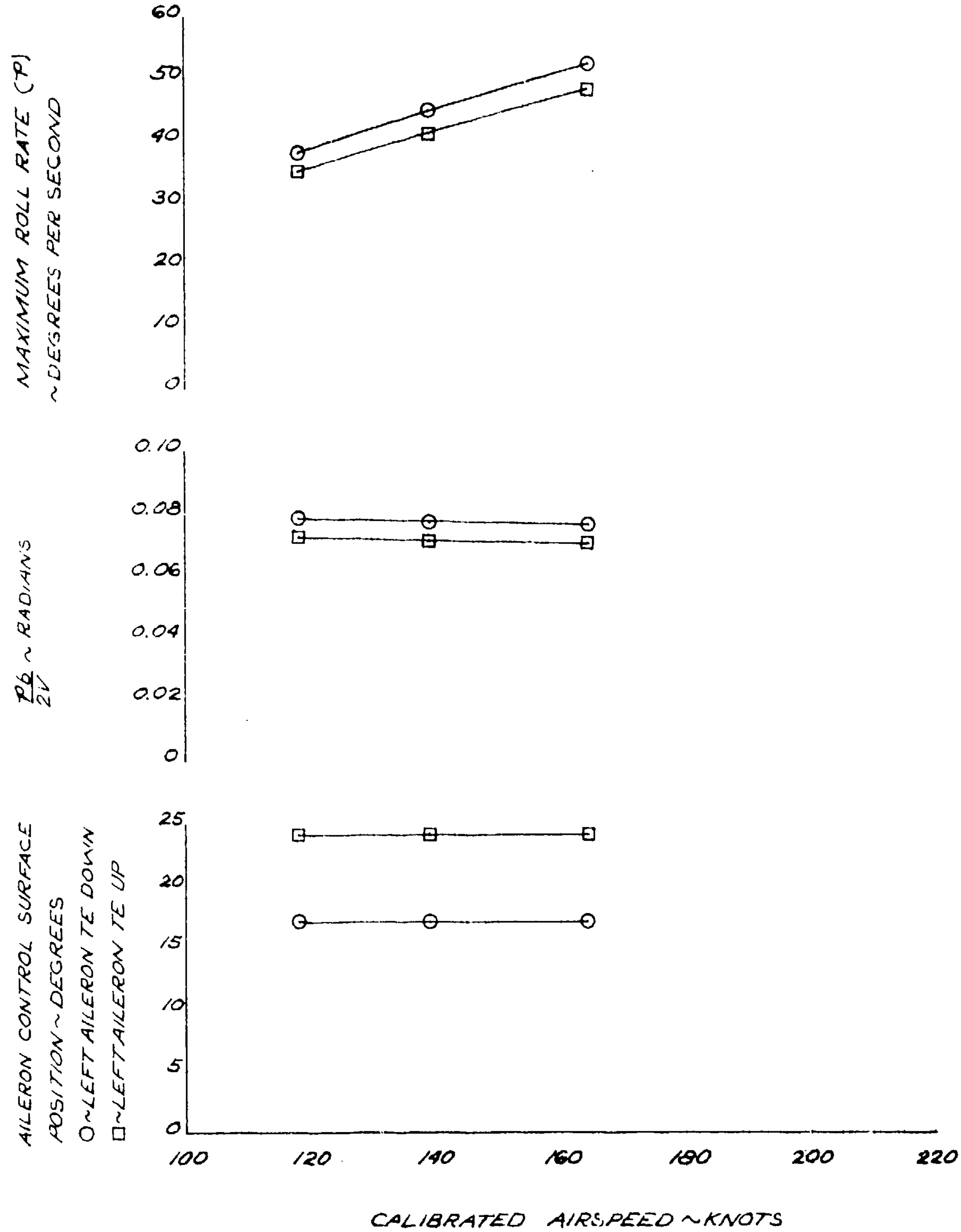


FIGURE 52

ENGINE CHARACTERISTICS

PT6A-41

MAXIMUM CRUISE AND CRUISE CLIMB POWER AVAILABLE
SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
SPECIFICATION ENGINE IS A MINIMUM ENGINE WITH
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE:
1. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
2. ACCESSORY LOSSES: 14.25 SHP BELOW 10°C & 19.29 SHP ABOVE 10°C
3. BLEED AIR: 6.1 LB/MIN. BELOW -17°C AND 4.9 LB/MIN. ABOVE -17°C
4. ENGINE INLET EFFICIENCY CURVE FIGURES 56 AND 57
5. PROPELLER SPEED = 2000 RPM

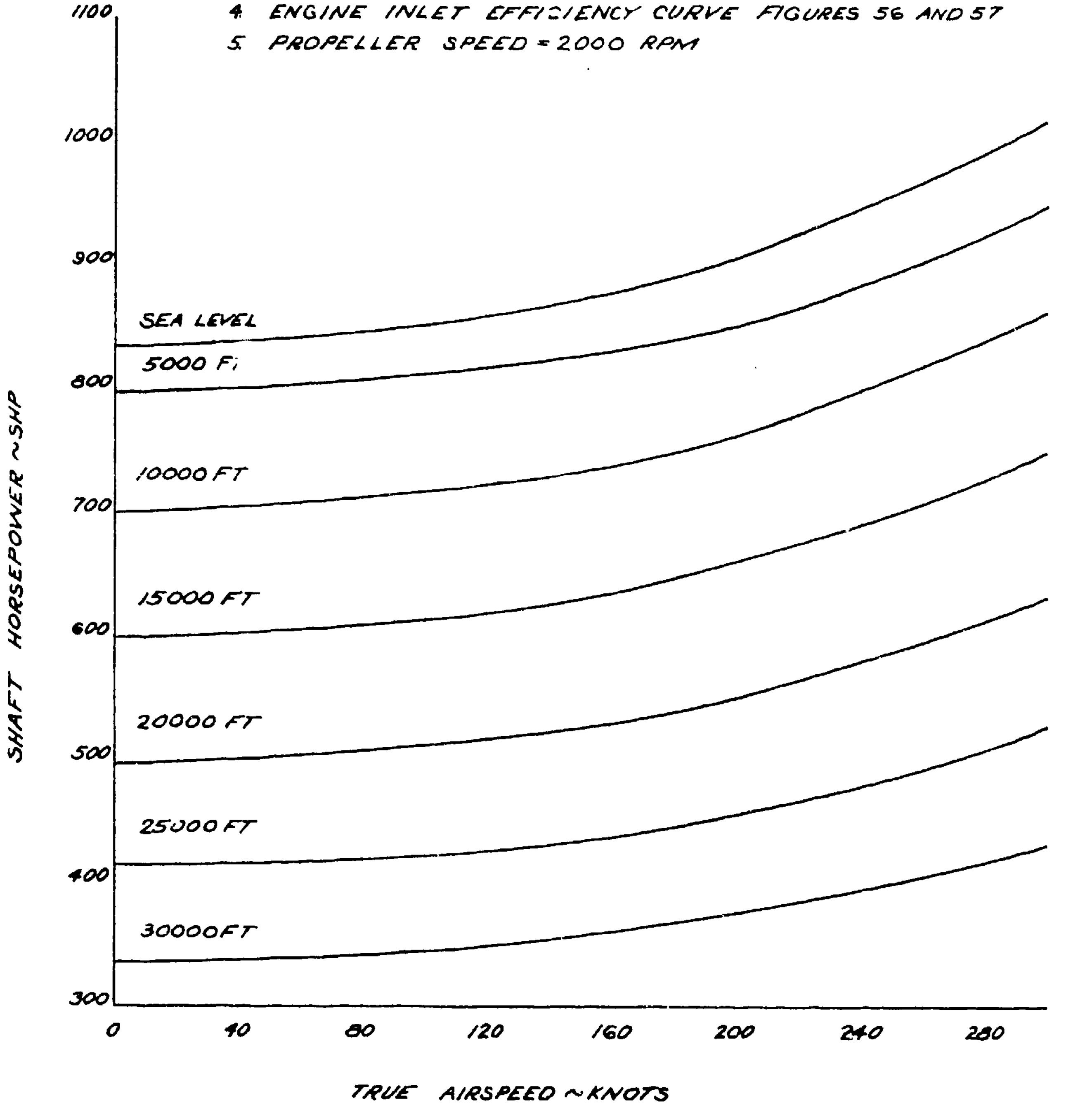


FIGURE 53

ENGINE CHARACTERISTICS

PT6A-41

JET THRUST FOR MAXIMUM CRUISE POWER AVAILABLE
SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
SPECIFICATION ENGINE IS A MINIMUM ENGINE WITH
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE: 1. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
2. ACCESSORY LOSSES: 14.25 SHP BELOW 10°C & 19.29 SHP ABOVE 10°C
3. BLEED AIR: 6.1 LB/MIN. BELOW -17°C AND 4.9 LB/MIN. ABOVE -17°C
4. ENGINE INLET EFFICIENCY CURVE FIGURES 56 AND 57
5. PROPELLER SPEED = 2000 RPM

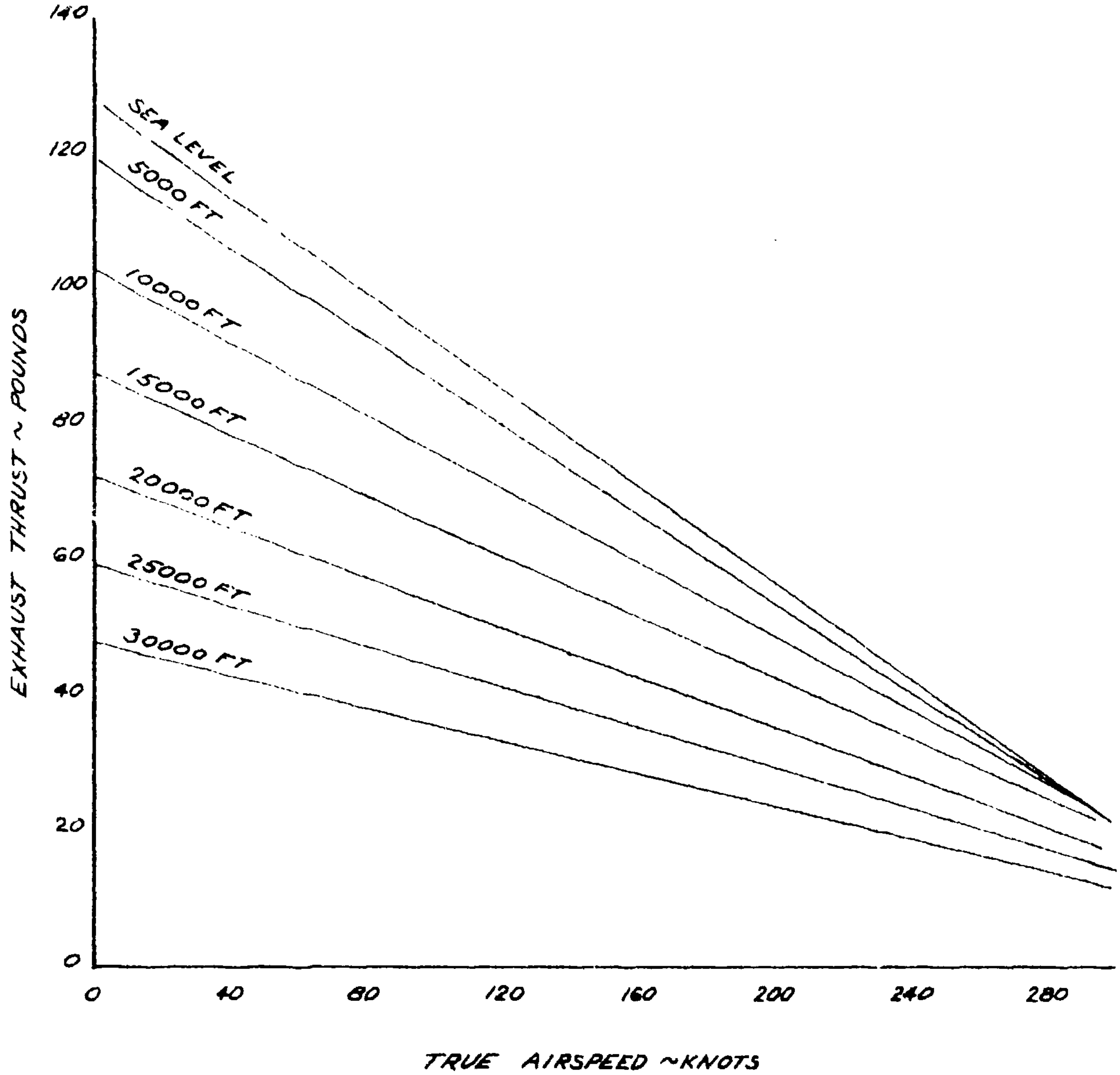


FIGURE 54

ENGINE CHARACTERISTICS

PT6A-41

MAXIMUM CRUISE AND CRUISE CLIMB POWER AVAILABLE
SPECIFICATION ENGINE BASED ON UACL ENGINE DATA
SPECIFICATION ENGINE IS A MINIMUM ENGINE WITH
MAXIMUM TIME BEFORE OVERHAUL
STATIC CONDITION

NOTE 1. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
2. ACCESSORY LOSSES: 14.25 SHP BELOW 10°C & 19.29 SHP ABOVE 10°C
3. BLEED AIR: 6.1 LB/MIN. BELOW -17°C AND 49 LB/MIN ABOVE -17°C
4. HOT-DAY DEFINITION OBTAINED FROM MIL-C-8678 (AER)
5. ENGINE INLET EFFICIENCY CURVE FIGURES 56 AND 57
6. PROPELLER SPEED = 2000 RPM

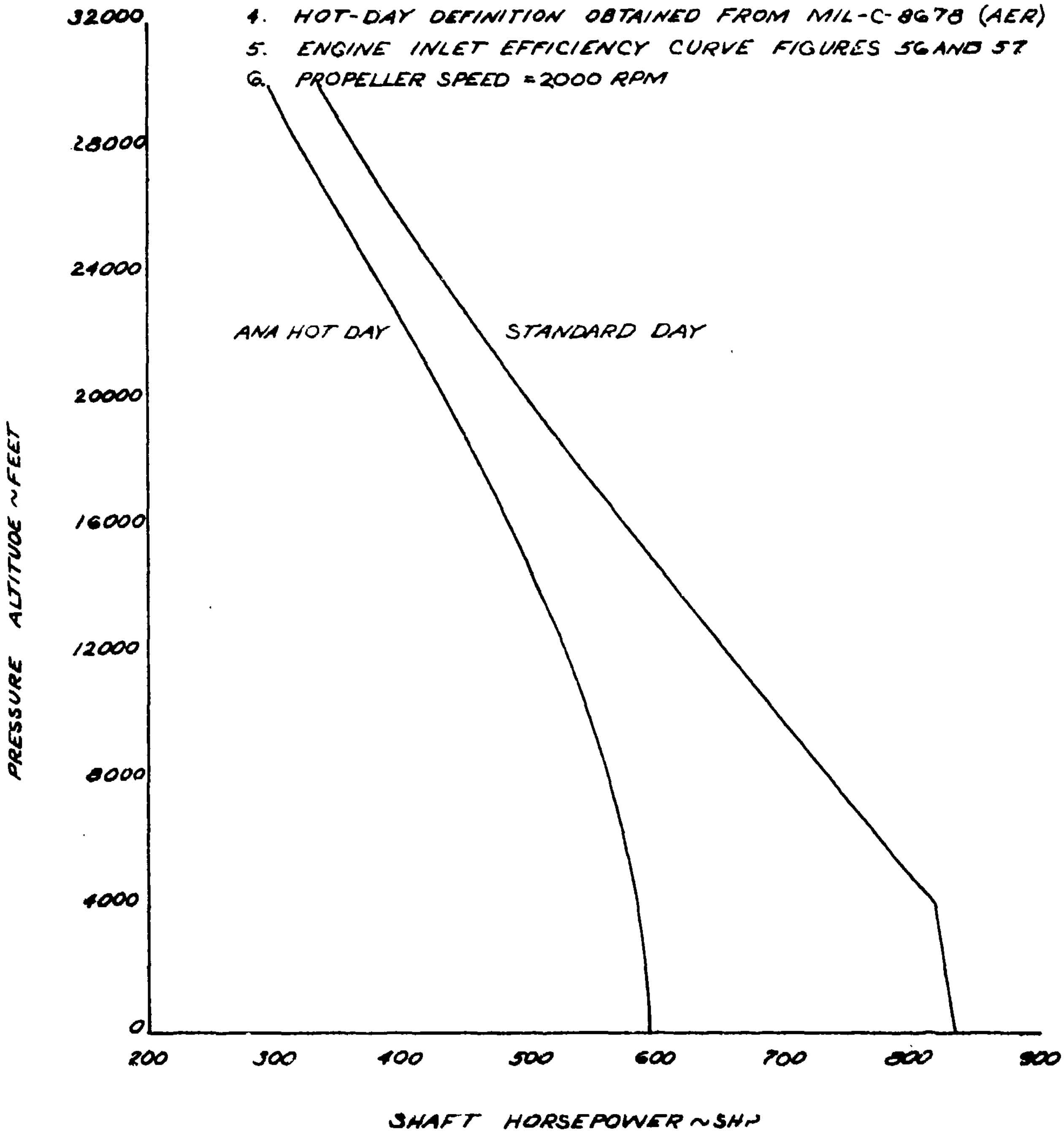


FIGURE 55

ENGINE CHARACTERISTICS

PT6A-41

MAXIMUM CRUISE AND CRUISE CLIMB POWER AVAILABLE
SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
SPECIFICATION ENGINE IS A MINIMUM ENGINE WITH
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE
1. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
2. ACCESSORY LOSSES: 14.25 SHP BELOW 10°C & 19.29 SHP ABOVE 10°C
3. BLEED AIR: 6.1 LB/MIN BELOW -17°C AND 4.9 LB/MIN ABOVE -17°C
4. ENGINE INLET EFFICIENCY CURVE FIGURES 56 AND 57
5. PROPELLER SPEED = 2000 RPM

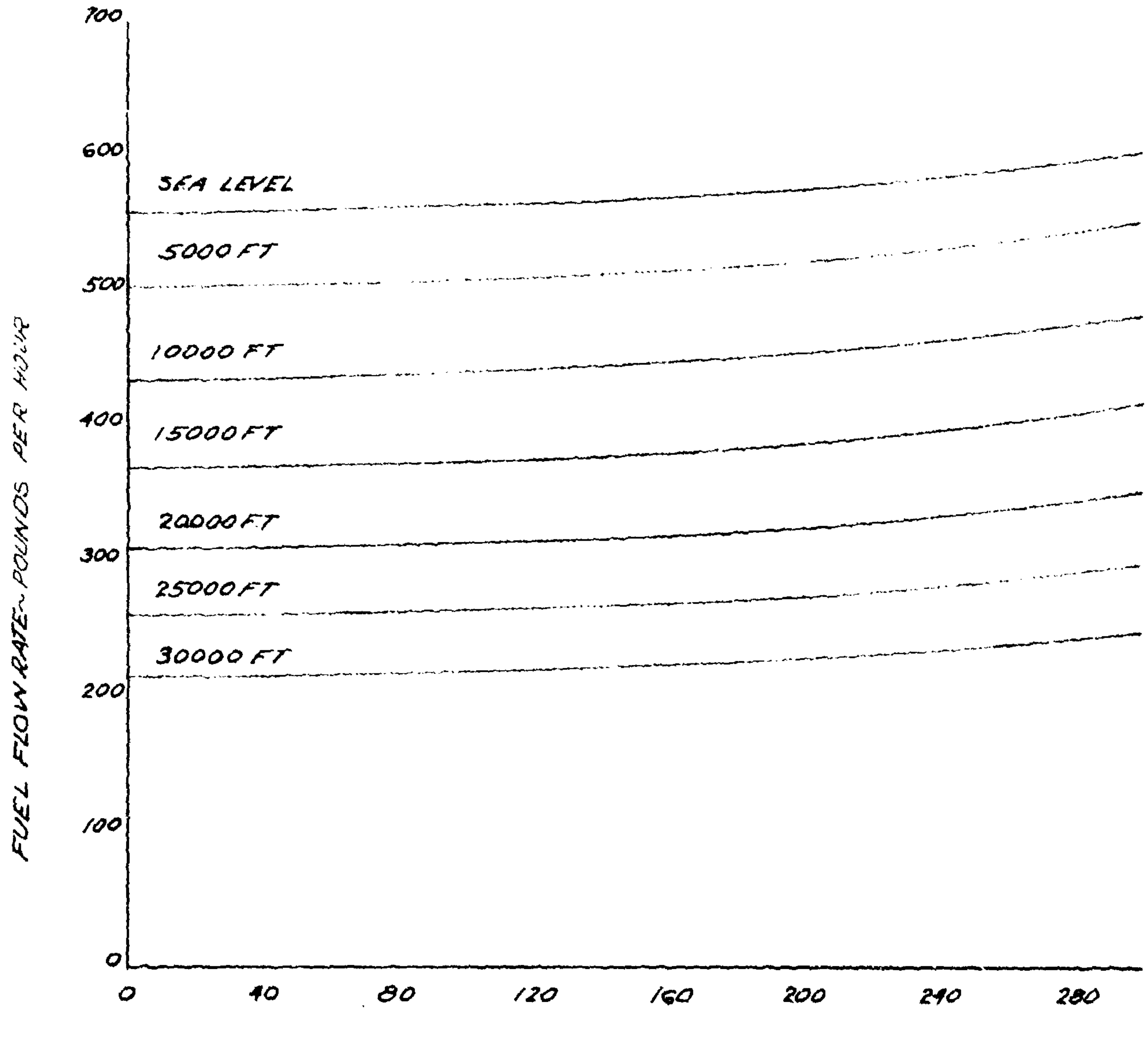


FIGURE 56
ENGINE CHARACTERISTICS
PTGA-41

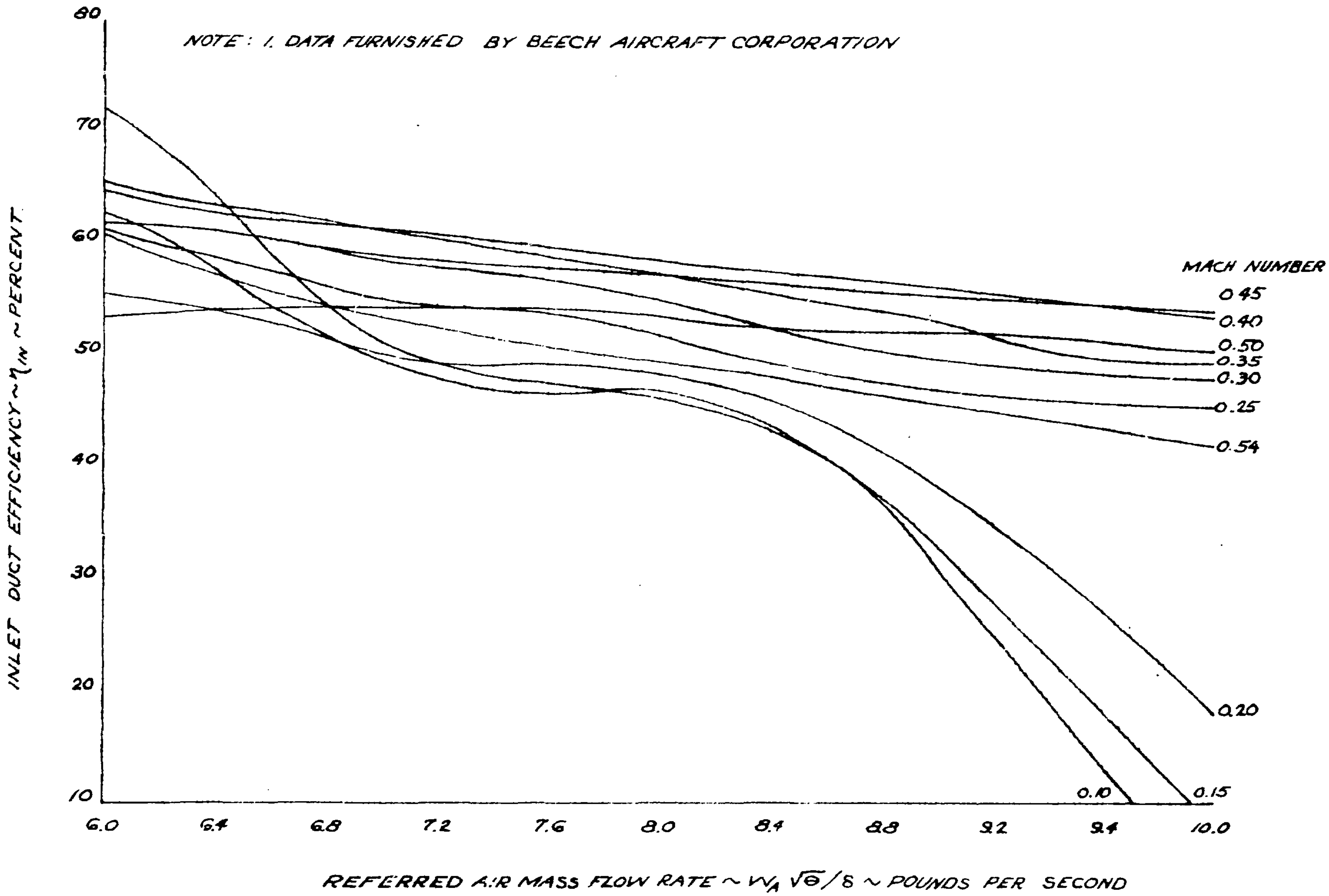


FIGURE 57

ENGINE CHARACTERISTICS

PT6A-41

NOTE: 1. DATA FURNISHED BY BEECH AIRCRAFT CORPORATION

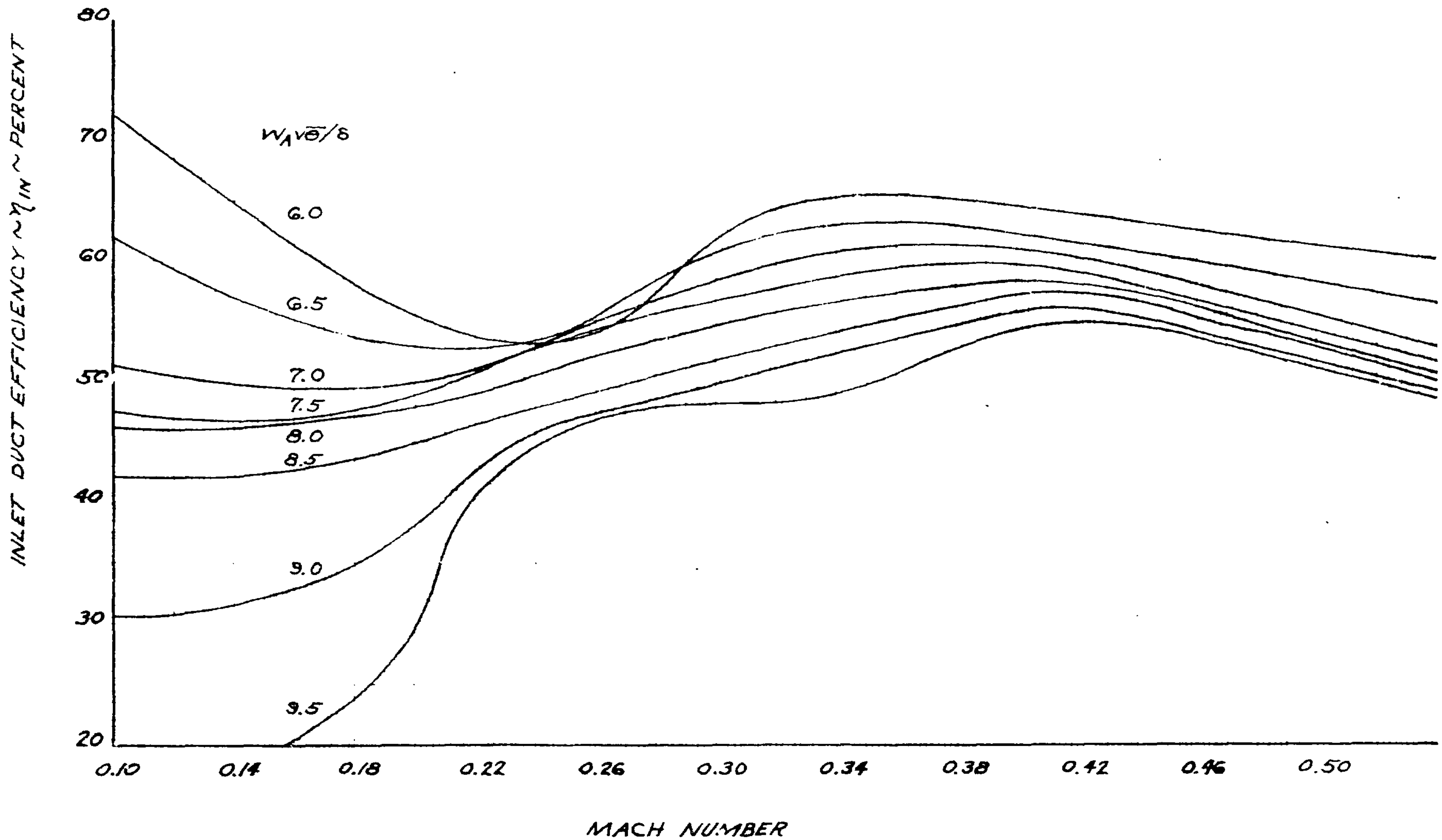
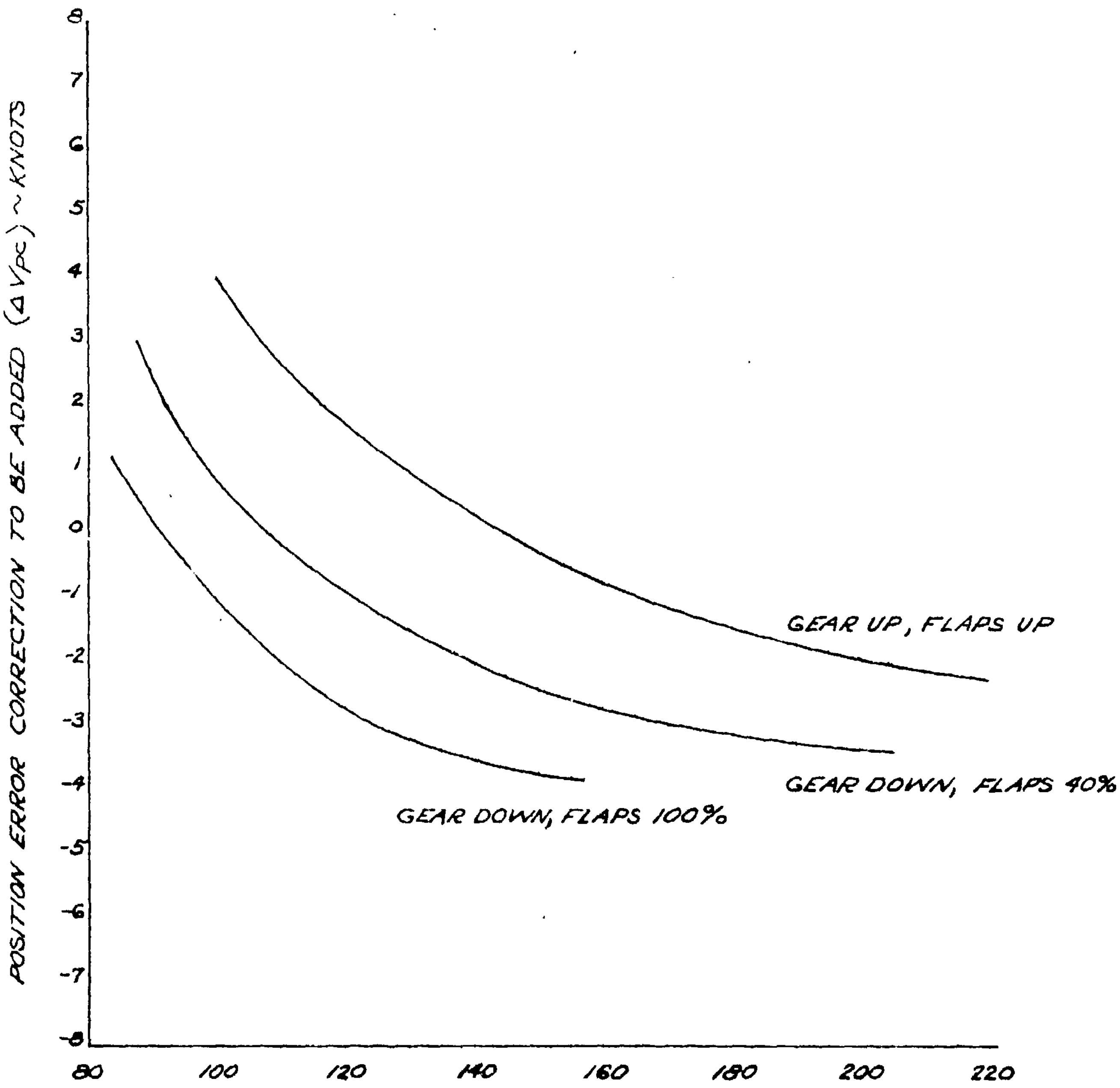


FIGURE 58
PITOT STATIC SYSTEM CALIBRATION
BAC MODEL 200 S/N 71-21058
PILOT'S STANDARD SYSTEM

NOTE: DATA FURNISHED BY BAC



APPENDIX F. DEFINITIONS ABBREVIATIONS, AND SYMBOLS

This list includes most of the symbols used in this report. However, certain portions of the report use special or unusual abbreviations and symbols. The meaning of these is made clear in the text of the report and, when that is the case, the abbreviation or symbol will not be found in this list. Also, certain symbols have more than one meaning; however, the context should make the meaning clear.

Symbols and Abbreviations	Definition	Unit
ANA	Air Force Navy Aeronautical	–
AC	Alternating current	–
b	Wing span	feet
C_{D_O}	Minimum coefficient of drag of the propeller feathered drag polar	–
C_D	Coefficient of drag	–
$C_{D_{BL}}$	Base-line coefficient of drag	–
$C_{D_{PF}}$	Powered flight coefficient of drag	–
C_P	Coefficient of power	–
C_L	Coefficient of lift	–
Cont	Continuous	–
D	Drag	–
De	Degree	°C
e	Oswald's span efficiency factor	–
f	Equivalent flat plate area	ft^2
F_N	Jet thrust	pounds
g	Acceleration of gravity	ft/sec^2
H_D	Density altitude	feet

Hp_i	Indicated pressure altitude	feet
Hp	Pressure altitude	feet
Hp_{ic}	Instrument corrected pressure altitude	feet
J	Advance ratio	–
L	Lift	pounds
MAC	Mean aerodynamic chord	–
Max	Maximum	–
MCP	Maximum continuous power	–
Min	Minimum, minute	–
N_p	Propeller s[illegible]d	rpm
N_1	Gas producer speed	percent
N_2	Power turbine speed	rpm
NAMPP	Nautical air miles per pound of fuel	–
NU	Nose up	–
ND	Nose down	–
OAT	Outside air temperature	°C
p	Roll rate	radians/sec
P_a	Ambient pressure	in. of mercury
P_o	Standard-day, sea-level pressure	in. of mercury
psi	Pounds per square inch	$lb/in.^2$
q	Dynamic pressure	lb/ft^2
Q	Torque	ft-lb
ref	referred, reference	–
R/C	Rate of climb	ft/min

S	Wing area	ft^2
SE	Single engine	–
SHP	Shaft horsepower	–
SL	Sea level	–
S/N	Serial number	–
STD	Standard	–
T_a	Ambient air temperature	°C
T_C'	Coefficient of thrust	–
T_i	Indicated air temperature	°C
T	Thrust	lb
T_{ic}	Instrument corrected on temperature	°C
THP	Thrust horsepower	HP
T_o	Sea-level, standard-day static temperature	°K
UHF	Ultra high frequency	–
V_{cal}	Calibrated airspeed	knot
VHF	Very high frequency	–
V_i	Indicated airspeed	knot
V_{ic}	Instrument corrected airspeed	knot
V_T	True airspeed	knot
V_{MC}	Airspeed for minimum control	knot
V_S	Stall airspeed	knot
V_H	Maximum airspeed for level flight	knot
V_{MO}	Maximum operating airspeed	knot
V	True airspeed	ft/sec

Wa	Engine airflow	lb/hr
W	Weight	pounds
°C	Degrees Centigrade	degrees
°F	Degrees Fahrenheit	degrees
°K	Degrees Kelvin	degrees
Δ	Difference	–
$\Delta C_{D_{PF-BL}}$	Difference in coefficient of drag due to thrust effects	–
ΔV_{PC}	Airspeed position error correction	–
ζ	Damping ratio	–
θ	Temperature ratio, descent angle	–, degrees
δ	Pressure ratio	–
σ	Density ratio	–
ρ	Air mass density	slug/sec^3
ω_d	Damped natural frequency	radian/sec
ω_n	Undamped natural frequency	radian/sec
α	Angle of attack	degrees
Φ	Roll or bank angle	degrees
η_p	Propeller efficiency	–
Φ/β	Roll to yaw ratio	–
dh/dt	Tape-line rate of descent	ft/min
π	3.14159	–
$\eta_{in.}$	Inlet duct efficiency	percent