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PRELIMINARY AIRWOR'HINESS EVALUATION OF THE RC-12K

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AEFA

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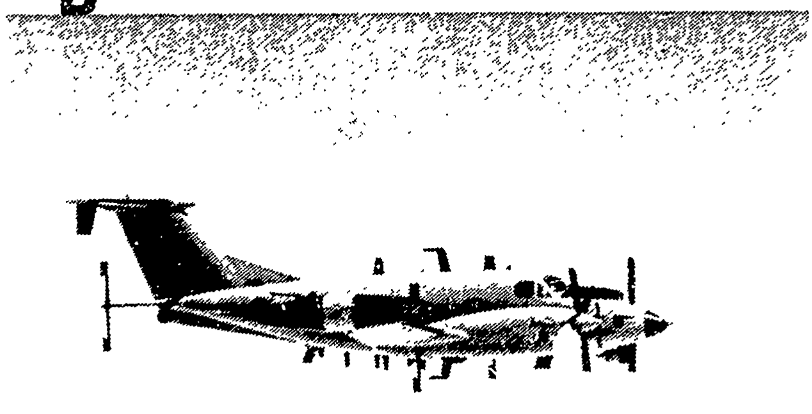

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The U.S. Army Aviation Engineering Flight Activity conducted a Preliminary Airworthiness Evaluation of the RC-12K aircraft from 12 through 22 June 1989 at Wichita, Kansas. During the test program, ten flights were conducted for a total of 22.8 hours, of which 14.6 hours were productive. Tests were performed in the Guardrail/Common Sensor System mission configuration ballasted to an average gross weight of 16,000 lb and centers of gravity at fuselage station 188.1 (forward) and 195.1 (aft). Test results were evaluated against military specification MIL-F-8785C for handling qualities. Performance data presented in the operator's manual were confirmed, as was the Beech Aircraft Corporation provided drag polar. The RC-12K test aircraft had good takeoff, accelerate-stop, landing, and climb performance. At 16,000 lb maximum gross weight, the standard day service ceiling is above the certified ceiling of 35,000 ft. The handling qualities of the RC-12K were satisfactory. Minimum control speeds dictated that the flaps down approach speed was 106 knots indicated airspeed regardless of weight. The heavy-duty main gear wheel and brake system contributed to good accelerate-stop and landing performance. The lack of adequate stall warning and the poor engine cowl securing device were identified as deficiencies. One shortcoming was identified. Keywords: Army reconnaissance aircraft handling, Takeoff/Climbing, Aircraft landings; Aircraft engine cowlings; Wheels/Braking, Flight testing. (EDC)

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INTRODUCTION

BACKGROUND

1. The RC-12K is similar to the RC-12H aircraft with the exceptions listed in appendix B. The aircraft is designed for electronic warfare/electronic intelligence operations. Special equipment includes a series of external antennas. A quantitative and qualitative evaluation of aircraft performance and handling qualities was required. The U.S. Army Aviation Engineering Flight Activity (AEFA) was tasked by the U.S. Army Aviation Systems Command (ref 1, app A) to conduct a Preliminary Airworthiness Evaluation (PAE) of the RC-12K airplane.

TEST OBJECTIVE

2. The objective of the PAE of the RC-12K was to verify handbook performance data and document aircraft handling qualities.

DESCRIPTION

3. The RC-12K is a 16,000 lb maximum gross weight Beech Aircraft Corporation (BAC) King Air Model A 200 CT aircraft modified by BAC with sensor pods and antennas. The test aircraft, USA Serial Number 85-0149, is a pressurized, all-weather transport with all metal construction. The aircraft is powered by two PT6A-67 engines, flat rated at 1100 shaft horsepower and equipped with four-bladed full-feathering propellers. The pilot and copilot are seated side by side with dual flight controls. The retractable tricycle landing gear is hydraulically actuated. The flight control system is fully reversible. A single-engine, asymmetrical thrust, rudder boost and yaw-damper system is provided to improve directional stability. The wing structure incorporates a three element spar. The empennage has nonmovable stabilons attached. A more detailed description of the RC-12K aircraft is contained in the operator's manual (ref 2) and Beech Specification BS-24123 (ref 3). Appendix B contains a more detailed description of the RC-12K.

TEST SCOPE

4. A PAE was conducted on RC-12K (Guardrail/Common Sensor), USA SN 85-0149, at Wichita, Kansas from 12 to 22 June 1989. A total of 14.6 productive flight hours were flown (22.8 total hours). Mission antennas depicted in figures B-6 through B-9 were installed and the wing tip mission pods were ballasted to mission weight. The test aircraft had an 86 in. boom installed on the nose at fuselage station 14 0 to accommodate test instrumentation (sideslip, angle of attack, and pitot/static pressure). The test aircraft was ballasted to a takeoff gross weight of approximately 16,700 lb to permit testing at the maximum gross weight of 16,000 lb and longitudinal center of gravity of 188.1 (forward) for the performance tests and fuselage station 195.1 (aft) for the handling qualities tests. The test aircraft handling qualities were compared to the requirements of military specification MIL-F-8785C (ref 4). Performance was

compared with the Type Inspection Report (ref 5) and drag polars provided by BAC. Flight restrictions and operational limits contained in the operator's manual and the airworthiness release (ref 6) were observed. The aircraft configurations are presented in table 1 and the test conditions are shown in tables 2 and 3.

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used during this test program (refs 7 and 8). The test methods are described briefly in the Results and Discussion section of this report. Flight test data were recorded on magnetic tape and logged from calibrated cockpit instruments. A test airspeed boom system was mounted on the nose at fuselage station 14.0. A 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, and data reduction techniques are described in appendix D. Control system rigging check, fuel cell calibration, and aircraft weight and balance were performed by BAC and monitored by AEFA personnel. A pitot-static system calibration was provided to AEFA personnel by BAC. An airspeed calibration check was conducted using the pace method with AEFA's T-34 calibrated airspeed system. Deficiencies and shortcomings are in accordance with the definitions presented in appendix D.

Table 1. Aircraft Configurations

Configuration	Landing Gear Position	Flap Setting (%)	Power Settings	Propeller Speed (rpm)
Takeoff (TO)	Down	0 40	Takeoff	1700
Cruise (CR)	Up	0	As Required	As Required
Landing (L)	Down	100 0	Idle	1700
Power Approach (PA)	Down	100	Power to maintain 5 deg descent angle	1700
Glide (GL)	Up	0	Power off, propellers feathered	0
Go-Around (GA)	Down	100	Takeoff	1700
Climb (CL)	Up	0	Maximum Continuous	1700

Table 2. Performance Flight Test Conditions[1]

Test	Indicated Airspeed (kts)	Pressure Altitude (ft)	Average Outside Air Temperature (°C)	Configuration
Dual-Engine Takeoff Perfomance	Handbook Recommended Speeds	1030	+24	TO[2]
Single-Engine Takeoff Performance (Accelerate-Go)			+24	
Accelerate-Stop Performance (Rejected Takeoff)			+26	
Dual-Engine Landing Performance[3]			+27	L[4]
Single-Engine Landing Performance[3]				
Level Flight Performance	117 to 165	31,000	−44.5	CR
Glide Performance	110 to 170	15,500 to 13,500	−7.5	GL
Climb Performance	140 to 31,000 ft H_p[5] 127 to 35,000 ft H_p	1332 to 35,000	-17	CL

NOTES:

[1]Tests were conducted ball-centered at 16,000 lb average gross weight and center of gravity (cg) at fuselage station (FS) 188.1 (fwd) without IR suppressors installed.
[2]Takeoff performance was conducted at 0% and 40% flaps.
[3]Dual and single-engine landing performance was conducted using maximum ground fine and maximum braking.
[4]Landing performance was conducted at 0% and 100% flaps.
[5]H_P: Pressure altitude.

Table 3. Handling Qualities Test Conditions[1]

Test	Trim Calibrated Airspeed (kt)	Pressure Altitude (ft)	Configuration[2]
Control Positions	120 to 170	31,000	CR
Static Longitudinal Stability	110	5000	PA
Static Lateral-Directional Stability	131, 161	31,000	CR
Dynamic Longitudinal Stability	109	5000	PA
	109, 131	24,000	CR
Dynamic Lateral-Directional Stability	131	31.000	CR
Roll Performance[3]	132, 164	31,000	CR
Dual-Engine Stall Characteristics[4]	110	14,000	TO, CR, L, PA
Single-Engine Stall Characteristics[5]	98		TO, PA
Single-Engine Characteristics[6]	98	8000 12,000 16,000	TO, GA
		31,000	CR

NOTES:

[1]Tests were conducted ball-centered, at 16,000 lb gross weight and center of gravity (cg) at fuselage station (FS) 195.1 (aft) without IR suppressors installed.
[2]CR: Cruise, PA: Power Approach, TO: Takeoff, L: Landing, GA: Go-Around.
[3]These tests were conducted with both DF/ELINT pods ballasted to mission weights.
[4]Unaccelerated stalls were conducted with cg at FS 188.1 (fwd) and spot checked at FS 195.1 (aft). Accelerated stalls were also conducted using 2g windup turns with 2 knots per second or less deceleration.
[5]Single-engine stalls were conducted with takeoff power and power for single-engine approach with inoperative engine propeller feathered. Single-engine stalls were conducted with a fuel imbalance of 200 lb on the side with the inoperative engine.
[6]Static and dynamic V_{mc}. V_{mc} (minimum airspeed for which control can be maintained) were determined at takeoff power condition. Single-engine characteristics were evaluated with a fuel imbalance of 200 lb.

RESULTS AND DISCUSSION

GENERAL

6. Limited performance and handling qualities tests of the RC-12K aircraft were conducted at Wichita, Kansas, in the mission configuration. The aircraft and wing pods were ballasted to mission weight. The RC-12K's takeoff, accelerate-stop, landing, climb performance and rudder boost system enhance safe mission accomplishment. Two deficiencies (poor engine cowl securing device and inadequate stall warning), and one shortcoming (illumination of the master caution light with propeller blade angle changes) were identified.

PERFORMANCE

General

7. The performance characteristics of the RC-12K aircraft were evaluated in the normal mission configuration near the maximum takeoff gross weight (16,000 lb) and forward longitudinal center of gravity (FS 188.1). Takeoff, accelerate-stop, accelerate-go, and landing performance was conducted at Salina, Kansas Municipal Airfield on a dry, hard-surfaced runway. The RC-12K met or exceeded these operator's manual performance numbers. A previously reported deficiency of main landing gear wheel lockup during heavy braking on the RC-12D and RC-12H models was less of a problem with the increased capacity wheel and brake system, however, one tire was blown due to heavy braking during a maximum performance landing test. Propeller feathered glide tests essentially confirmed the baseline drag polar developed by Beech Aircraft Corporation (BAC) for the RC-12K. Table 4 provides a listing and definitions of the speeds used for the purposes of this test.

Takeoff Performance

8. Dual-engine takeoff performance was evaluated at the conditions shown in table 2. The BAC recommended procedure was used. Dual-engine takeoffs were conducted as follows: After completing the line-up check, the copilot applied power to the setting determined from the "minimum static takeoff power at 1700 rpm" chart furnished by BAC. Brakes were released and directional control maintained with nosewheel steering and rudder, while maintaining wings level with ailerons. The pilot retained a light hold on the power levers until the copilot called takeoff decision speed ("V_1") so as to be ready to initiate abort procedures if required. The copilot ensured that the autofeather advisory lights were illuminated and monitored engine torque during the takeoff roll. As the aircraft accelerated, engine torque increased but the engine torque and turbine gas temperature limits were not exceeded. As the copilot called "V_1", the pilot removed his hand from the power levers. When the copilot called "rotate" at rotation speed (V_R) the pilot commenced a smooth, positive rotation to an indicated pitch attitude of approximately 10 degrees. As soon as a positive rate of climb was verified, the landing gear was retracted by the pilot. The aircraft accelerated rapidly through takeoff safety speed (V_2) to V_y or cruise climb airspeed. The flaps were retracted when safely airborne. The two-engine takeoff distance test results and

Table 4. Speed Definitions

Test Speed	Definition
V_1	Takeoff Decision Speed That speed below which if an engine failure occurs during takeoff the takeoff is aborted. Above this speed the takeoff can be continued.
V_R	Rotation Speed That speed during takeoff at which back elevator pressure is applied rotating the aircraft to a specified pitch attitude causing the aircraft to become airborne.
V_2	Takeoff Safety Speed That speed which is attained at or before 35 feet above the takeoff surface after an engine failure at which the required one-engine inoperative climb performance can be achieved.
V_{ENR}	Enroute Climb Speed This speed is V_{yse} or single-engine best rate-of-climb speed.
V_{REF}	Reference Speed The airspeed on the desired approach angle at 50 feet above the runway threshold.
V_{EF}	Engine Failure Speed That speed during the takeoff ground roll at which an engine was shutdown and the takeoff continued or aborted after a two second delay. This speed was V_1 - 3 KIAS and is used for flight test purposes only.
V_y	Best rate of climb airspeed.

operator's manual chart values are compared in table E-1. The RC-12K test aircraft exceeded the performance shown in the operator's manual for all two-engine takeoff tests. The RC-12K two-engine takeoff performance contributes to safe mission accomplishment by providing rapid acceleration though critical, low airspeeds and altitudes.

9. Single-engine takeoff (accelerate-go) performance was evaluated at the conditions shown in table 2. The BAC recommended procedure as discussed for dual-engine takeoffs was followed up to engine failure speed (V_{EF}). At V_{EF} (3 knots indicated airspeed (KIAS) slower than V_1) the copilot placed the critical (left) engine condition lever to the fuel cut-off position, causing the engine to shut down, the propeller to autofeather, and rudder boost system to activate. Directional control was easily maintained with very little (less than five ft) deviation from center line. Single-engine acceleration was continued to V_R where the pilot smoothly increased the pitch attitude to approximately 10°. After takeoff, a positive rate of climb was verified and the gear retracted. The pilot continued to accelerate the aircraft so as to achieve V_2 by 35 ft, using up to 5° of bank into the operating engine to enhance climb performance and lateral-directional control. V_2 speed was maintained until 500 ft where a level acceleration to enroute climb speed (V_{ENR}) was accomplished. At V_{ENR}, the flaps were retracted, if used, and the climb continued at V_{ENR} to 1500 ft. Test result single-engine takeoff (accelerate-go) data are compared with operator's manual accelerate-go data in table E-1. Test results showed less distance required for single-engine takeoffs than presented in the operator's manual. The RC-12K single-engine takeoff performance presented in the operator's manual is conservative. The RC-12K accelerate-go performance contributes to safe operation in the event of engine failure during takeoff.

Accelerate-Stop Performance

10. Accelerate-stop (rejected takeoffs (RTO's)) were conducted at the conditions shown in table 2. The BAC recommended procedure for dual-engine takeoffs was followed up to V_{EF}, where the copilot placed the critical (left) engine condition lever to the fuel cut-off position, failing the engine and starting the autofeather operation. The pilot continued to accelerate single-engine for two seconds, with no control inputs, simulating pilot reaction time, then placed both power levers to the maximum ground fine position and commenced maximum wheel braking to bring the aircraft to a stop. Test RTO distances are compared with the distances presented in the operator's manual in table E-1. The operator's manual depicted RTO distances are conservative. The RC-12K's RTO performance contributes to safe operation in event of engine failure during takeoff. Although the RTO performance is good, braking action could be improved for all conditions, especially for wet or slippery runways by incorporating an anti-skid device. Incorporation of an anti-skid capability would result in maximum performance RTO's with minimum chance of aircraft damage.

Landing Performance

11. The RC-12K's landing performance was evaluated at the conditions shown in table 2. Landings were conducted by stabilizing at the reference airspeed (V_{ref}), at

approximately 500 ft per minute rate of descent. At a radar altitude of 50 ft, the throttles were retarded to flight idle and a normal flare performed. At touchdown, the nosewheel was lowered to the runway, the throttle placed in ground fine for disking drag and maximum braking used to bring the aircraft to a stop. One landing was done with maximum reverse although there are no landing performance charts for maximum reverse. The test results and operator's manual presented distances are shown in table E-1. The test results for all landing distances show that the operator's manual distances are conservative. During one landing, the left outboard tire was blown due to sliding the wheel. Even though the stopping distances for all accelerate–stop and landing performance was excellent and the pilot had a relatively good feel for the braking action (there appeared to be a relatively linear increase in brake pedal displacement with increasing pressure and wheel brake effectiveness), there were no good cues that a wheel was actually sliding (the engine noise precluded hearing squalling tires). The RC-12K should incorporate an anti–skid system to improve wet/slippery runway stopping performance and aircraft control, and to preclude inadvertent wheel lockup during conduct of maximum performance landing and RTO procedures. The maximum reverse landing showed some reduction in landing distance but maximum reverse would only be needed for emergency landing runways. The RC-12K landing performance is satisfactory.

Climb Performance

12. A climb to the certified ceiling of 35,000 ft was conducted using handbook recommended endurance airspeeds of 140 KIAS to 31,000 ft and 127 KIAS to 35,000 ft. Takeoff gross weight was 16,697 lb. The time from brake release (field elevation of 1272 ft) to 35,000 ft was 35 minutes with the following interim altitudes and times:

Pressure Altitude (ft)	Time to Climb
12,000	6 min, 20 sec
22,800	13 min
26,000	16 min
30,000	21 min, 35 sec
33,000	28 min
35,000	35 min

The weight at 35,000 ft was approximately 16,000 lb with a climb rate of 200 – 300 feet per minute. Climbs using V_y speeds should provide better climb rates than the operator's manual published cruise climb schedule. The operator's manual should include a time, fuel, and distance to climb chart for V_y speeds as it does for a cruise climb schedule. This will allow mission pilots to expedite climbs when required for mission operations. The RC-12K's climb performance and two engine service ceiling is significantly better than the operator's manual predicted climb performance. The RC-12K's climb performance contributes to mission on–station capability.

Level Flight Performance

13. Level flight performance was evaluated at the conditions shown in table 2 to verify BAC's performance results. The propeller feathered/stopped glide test method was used to obtain the baseline drag polar for the aircraft (fig. E-1). These tests were conducted with both engines shut down and the propellers feathered and stopped. The aircraft was stabilized in a descent at incremental airspeeds at a wings level, ball-centered condition through a pressure altitude band of 15,500 to 13,500 ft. Descents were repeated on reciprocal aircraft headings to account for wind gradients. The constant pressure altitude test method was used to determine dual-engine power required as a function of airspeed (figs. E-2 and E-3). The climb and level flight drag polar coefficients determined for the RC-12K are presented in table 5. Coefficients provided by BAC are also presented. Engine characteristics data are presented in figures D-1, D-2 and E-4 through E-6 and are included for future engineering analysis.

14. The propeller feathered glide drag polar was determined to be approximately the same as that provided by BAC. The zero thrust baseline drag polar furnished by BAC indicates higher drag than was determined during this evaluation. The drag polar coefficients provided by BAC indicate approximately 25 to 40 thrust horsepower per engine more power required for level flight than the data obtained during this evaluation. Based on the limited test scope of this evaluation, the BAC Type Inspection Report (ref 5) for level flight performance of the RC-12K appears conservative and is satisfactory.

Stall Performance

15. Stall performance was evaluated at the conditions listed in table 3. Unaccelerated stalls were conducted wings level with approximately 1 knot/second deceleration. Accelerated stalls were conducted using wind-up turns at constant load factor with a deceleration of approximately 2 knots/second. The stall airspeed as defined in MIL-F-8785C, paragraph 6.2.2 was the speed at which uncommanded pitching, rolling, or yawing occurred. Stall speeds, stall warning, and stall buffet speeds are shown in table E-2). Aerodynamic buffet provided inadequate stall warning. Artificial stall warning was provided by a stall warning horn. The airspeed margin between activation of the stall warning horn and unaccelerated stalls (as defined by MIL-F-8785C) was 9 knots above the stall in the power-approach (PA) configuration with flaps 100%, and one knot above the stall in the cruise (CR) configuration with power for level flight, and one knot above the stall in the landing configuration with maximum power and 100% flaps. Stalls conducted in all cruise configurations resulted in inadequate stall warning at 15000 ft H_p Stalls conducted at mission altitude (31,000 ft) in the cruise configuration were not preceded by an artificial or airframe warning. Stall warning margin for the takeoff (TO) and landing (L) configurations were also inadequate. The inadequate artificial stall warning in the CR, TO,and L configurations is a deficiency. The artificial stall warning system failed to meet the requirements of paragraph 3.4.2.1.1 of MIL-F-8785C in that the minimum stall warning is less than 5 knots for the CR, TO and L configurations. Accelerated and

Table 5. Climb and Level Flight Drag Polar Coefficients[1]

<table>
<tr><th>Number of Engines Operating</th><th>Flight Condition</th><th>C_{D_o}</th><th>$\frac{\Delta C_D}{\Delta C_L^2}$</th><th>A</th><th>B</th><th>C</th></tr>
<tr><td>0</td><td>Glide</td><td rowspan="2">0.0415</td><td rowspan="2">0.0409</td><td>Zero</td><td>Zero</td><td>Zero</td></tr>
<tr><td>2</td><td>Level</td><td>Zero</td><td>0.00836</td><td>0.0028</td></tr>
</table>

The following coefficients were provided by BAC

Number of Engines Operating	Flight Condition	C_{D_o}	$\frac{\Delta C_D}{\Delta C_L^2}$	A	B	C
0	Glide	0.0416	0.0418	Zero	Zero	Zero
2	Level	0.0463	0.0460	Zero	0.0676	-0.00496

NOTES:

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

Where:

C_D = Coefficient of drag

C_{D_o} = Baseline zero lift drag coefficient

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

C_L = Coefficient of lift

$T_{C'}$ = Coefficient of thrust

A, B, C = Drag polar constants

single-engine stall warning margins were satisfactory. The RC-12K should incorporate an angle-of-attack system that provides a constant indication to the pilot of his operational margin above the stall and provides information for operation at optimum performance and angle-of-attack.

HANDLING QUALITIES

General

16. A limited handling qualities and pilot workload evaluation of the RC-12K aircraft was conducted to determine stability and control characteristics at the test conditions listed in table 3. Emphasis was placed on operation at the maximum mission gross weight of 16,000 lb and aft mission cg (FS 195.1). All maneuvers were flown using ball-centered flight as a trim reference.

Control System Characteristics

17. Control system characteristics were measured on the ground in static conditions. Freeplay and breakout plus friction forces were measured and are presented in table E-3. Control surface travels and measured cable tensions are presented in table D-1. The RC-12K control system characteristics are satisfactory.

Control Positions in Trimmed Flight

18. The capability to trim the aircraft to a given airspeed and zero control force was evaluated concurrently with other testing. The control positions in trimmed forward flight are presented in figure E-7. Manual trim of all controls was satisfactory and easily accomplished for all configurations tested. The RC-12K control position characteristics in trimmed flight are satisfactory.

Static Longitudinal Stability

19. The RC-12K static longitudinal stability was evaluated at the conditions listed in table 3. The aircraft was trimmed in steady-heading, ball-centered level flight at V_{ref} (106 KIAS), then stabilized at incremental airspeeds greater than and less than the trim airspeed. Static longitudinal test data are presented in figure E-8. The stick-free static longitudinal stability, as indicated by the variation in longitudinal control force with airspeed, was positive for both airspeeds above and below the trim airspeed. The stick-fixed stability, as indicated by the variation in longitudinal control position with airspeed, was weak, but positive. The static longitudinal stability characteristics of the RC-12K airplane are satisfactory and meet the requirements of MIL-F-8785C.

Static Lateral-Directional Stability

20. Static lateral-directional stability tests were performed at the conditions listed in table 3. Tests were conducted by trimming the aircraft (ball-centered) in level flight and then stabilizing at various sideslip angles both left and right in approximate 1/2

ball width increments while maintaining a constant airspeed, power lever position, and zero turn rate. Test data are presented in figures E-9 and E-10. Apparent dihedral (variation of lateral control position with sideslip) and apparent directional stability (variation of directional control position with sideslip) were both positive. The directional control force variation with sideslip angle indicated positive stability and was essentially linear over the sideslip range tested. The RC-12K airplane had a sideslip to pitch coupling, as indicated by the requirement for increasing aft elevator control displacement and pull force with increasing sideslip angles in both directions. The side-force cues (variation of bank angle with sideslip) provided adequate indications of out-of-trim conditions. The static lateral-directional stability characteristics of the RC-12K airplane are satisfactory and met the requirements of MIL-F-8785C.

Dynamic Longitudinal Stability

21. The dynamic longitudinal stability characteristics were evaluated at the conditions shown in table 3. The long-term (phugoid) dynamic characteristics were evaluated by varying airspeed approximately 10 knots above or below the trim airspeed, then returning the longitudinal control to the trim position. The control fixed and control free long-term responses were evaluated during level flight with the autopilot system off. A time history of a representative response is presented in figures E-11. With both controls fixed and free, the long-term response was lightly damped with the autopilot system off. The short period response was excited by using approximately one in. elevator control doublets. A representative short period time history is presented in figure E-12. The short period was heavily damped for all conditions tested, with approximately one overshoot. The dynamic longitudinal stability characteristics of the RC-12K are satisfactory and meet the requirements of MIL-F-8785C.

Dynamic Lateral-Directional Stability

Dutch Roll Characteristics:

22. The dynamic lateral-directional stability characteristics (lateral-directional damping and dutch roll characteristics) were evaluated at the condition shown in table 3. These tests were conducted by exciting the aircraft from a coordinated level flight trim condition with rudder doublets at the aircraft natural frequency and releases from sideslips. Tests were conducted with yaw damper off and on and with controls fixed and free. Representative time histories are presented at figures E-13 and E-14. The lateral-directional oscillations (dutch roll mode) with the yaw damper and autopilot off were lightly damped. With the yaw damper engaged and the autopilot ON or OFF, the dutch roll mode was heavily damped and not easily excited. The dutch roll characteristics of the RC-12K aircraft are satisfactory and meet the requirements of MIL-F-8785C.

Spiral Stability:

23. The spiral stability characteristics of the RC-12K aircraft were evaluated at the conditions shown in table 3. These tests were conducted by establishing 10, 20, and 30 degree bank angles (both left and right) from trim conditions, using aileron only, and after stabilizing at the prescribed bank angle, the control was slowly returned to the trim position. Spiral stability (as indicated by change in bank angle with elapsed time) was essentially neutral for both left and right turns. The spiral stability characteristics of the RC-12K aircraft are satisfactory and meet the requirements of MIL-F-8785C.

Stall Characteristics

General:

24. The dual and single-engine stall characteristics of the RC-12K aircraft were evaluated in conjunction with stall performance testing (para 12) and stall handling qualities at the conditions listed in table 3. Stall warning, stall, and stall recovery characteristics were evaluated.

Unaccelerated Stalls:

25. The RC-12K unaccelerated dual-engine stalls were characterized by: (1) very light buffet onset; (2) artificial stall warning; (3) slight pitch oscillations; and (4) mild wing rock. Maximum power stalls with 100% flaps typically had a left roll simultaneous with the pitch break. Power on stalls resulted in minimal altitude loss. At heavy gross weight conditions (16,000 lb) the power off stall recovery required nose down pitch of approximately 20 degrees for several seconds to gain sufficient airspeed to avoid secondary stalls. This resulted in approximately 1500 ft altitude loss for stalls with gear and flaps retracted and approximately 1000 ft altitude loss with gear and flaps down. A significant portion of the altitude loss during power-off stalls may be attributed to turbine engine lag. As airspeed was decreased, the uncommanded nose down pitch could be controlled by applying additional aft elevator control. The aircraft generally could be controlled into deep stall to full aft elevator control by quick pilot reaction with aileron and rudder control to counter rolling and yawing motions. The ailerons and rudder were effective in controlling the aircraft laterally and directionally, even with full aft elevator control and 200 lb fuel imbalance. The good handling qualities in deep stall, with flaps extended may be attributed to the Wheeler Vortex Generators installed on the left outboard flap and repositioned stall strip on the right wing. Stalls conducted at an aft cg did not result in uncommanded pitch ups as they did with the RC-12H. This was probably due to the increased cg envelope made possible by the stabilons. The RC-12K handling qualities in the stall and during stall recoveries were satisfactory. Stalls induced with the autopilot engaged along with an altitude hold mode at 31,000 ft resulted in entering deep stall. The aircraft could not be powered out of the stall at mission altitude if power was applied at first warning (usually nose down pitch followed by artificial warning). As a result of the inadequate stall warning

previously discussed in paragraph 12, the autopilot had to be disconnected and stall recovery procedures used to return to normal flight.

26. Unaccelerated single-engine stall characteristics were evaluated at the conditions listed in table 3. The single-engine stall characteristics were essentially the same as the dual-engine stall characteristics. The single-engine unaccelerated stall characteristics of the RC-12K are satisfactory.

Accelerated Stalls:

27. Dual-engine accelerated (2g) stalls were evaluated at the conditions listed in table 3 using windup turns to the left. At the stall, the aircraft exhibited a tendency to roll right out of the turn. The dual-engine accelerated stall characteristics of the RC-12K are satisfactory.

Stall Recovery:

28. The RC-12K aircraft was recovered from all dual-engine stalls by relaxing aft longitudinal control force, reducing angle of attack and adding power to minimize altitude loss. At heavy gross weight, secondary stall tendency (recurrence of buffet) was encountered if sufficient airspeed was not attained prior to leveling the pitch attitude. Accelerated stall recovery was easily accomplished by reducing bank or pitch.

29. Single-engine stall recovery was best achieved by slightly reducing power on the operating engine at the pitch break, lowering the nose of the aircraft to the horizon, accelerating to the best single-engine rate of climb airspeed, and applying maximum controllable power to minimize altitude loss. Altitude loss during single-engine stall was 800 to 1200 ft.

Single Engine Characteristics

Static V_{mc}:

30. Static single-engine V_{mc} tests were conducted at the conditions in table 3, with the left (critical) engine inoperative and propeller feathered, by decelerating at 1 knot/sec while banking 5 degrees into the operating engine in constant heading flight. The operating engine was set at takeoff power with a propeller speed of 1700 rpm. Static V_{mc} was defined as the airspeed at which directional or roll control could not be maintained or single-engine stall (whichever occurred first).

31. V_{mc} was the single-engine stall speed for all conditions tested. At 8000 ft pressure altitude, loss of control and stall occurred simultaneously. V_{mc}/stall recovery was easily achieved by reducing pitch attitude simultaneously with a slight power reduction. Altitude loss was minimized by increasing the operating engine power as airspeed increased, compatible with the capability to control the aircraft directionally. The single-engine static V_{mc} characteristics are satisfactory.

32. The rudder boost system actuated when a difference of 60% torque was sensed between the engines. A typical time history of a single-engine V_{mc} is shown in

figure E-15. Rudder forces required during single-engine operations were significantly reduced by the rudder boost system with 75 lb required during the stall/V_{mc} tests using 100% torque on the right engine. No reversals of rudder force occurred (i.e., no excessive inputs of the rudder boost system) and system inputs appeared smooth and linear with torque difference greater than 60%. The RC-12K rudder boost system contributes to safe single-engine operation.

Dynamic V_{mc}:

33. Dynamic V_{mc} tests were conducted at conditions presented in table 3 in symmetrical flight using takeoff power. At 12,000 ft pressure altitude (H_P), two test methods were used to determine dynamic V_{mc}. One method was to simulate a left (critical) engine failure with the autofeather system armed (power to flight idle, propeller to minimum rpm) and the other method consisted of an actual engine shutdown with the autofeather system feathering the propeller. No significant differences were observed between the methods. The controls were held fixed for two seconds simulating pilot reaction time. All flight controls were then used to return the aircraft to stabilized flight at the trim airspeed without reducing power on the operating engine or adding power from the simulated failed engine. Dynamic V_{mc} was 2 to 3 knots above static V_{mc} (single-engine stalls) at all conditions tested. A representative time history is presented in figure E-16. The RC-12K dynamic V_{mc} characteristics are satisfactory.

Roll Performance

34. Roll performance of the RC-12K was evaluated at the conditions presented in table 3 with the yaw damper on. These tests were initiated by stabilizing in a 45° banked turn, making the desired control input and noting the time for 60° of roll (30° each side of wings level). One-half and full lateral control step inputs (in 0.2 seconds) both left and right were conducted. Test results are presented in figure E-17. Representative time histories are presented in figures E-18 and E-19. The aircraft was responsive in roll and the lateral control forces were satisfactory. The slight amount of adverse yaw was not objectionable. Time required to roll 60° either left or right with full control deflection was approximately 1.5 seconds at 158 KIAS. The roll performance of the RC-12K is satisfactory.

MISCELLANEOUS

Speed Command

35. The speed command display (fast-slow bug) did not give appropriate on-speed commands since its computer had been programmed for V_{ref} of 1.3 V_s. Due to V_{mc} considerations, the flaps down (100%) V_{refs} were 106 KIAS. V_{refs} for flaps up were based on 1.3 V_s. These speeds are given in the operator's manual in the flaps up landing distance charts. The speed command computer should be programmed to provide the correct speed command.

Engine Cowling

36. During conduct of the first group of stall evaluations, the left cowl on the number two engine came open with the bottom portion of the cowl being tucked inside of the engine compartment. The cowling remained attached to the nacelle during the return-to-field flight and did not present any further problems. Pictures of the cowl and latching mechanism are shown in figure 1. The forward latch apparently became unsecured. The cowl latches were taped over for the remainder of the stall evaluations. The poor engine cowl latching device is a deficiency. The engine cowl latching device should be redesigned to provide positive cowl security.

Taxiing Characteristics

37. Ground taxi operations were more difficult with the RC-12K than previous C-12 aircraft due to the unpredictability of the 7° propeller blade angle change. This change would occur while adjusting the power levers to effect an aircraft direction change, or reduce taxi speed. When the blade angle changed without the pilot knowing precisely when it was going to change, the aircraft would yaw significantly requiring full rudder in the opposite direction. This problem would be worse during taxi operations on slippery surfaces such as ice, snow or a wet runway. As more taxi experience was attained, the pilot was better able to predict when the blade angle change would occur. He could avoid that area unless he wanted to use it for turning or increasing/decreasing taxi speed. The change in power lever operational characteristics may require some pilot compensation until the characteristics become familiar to the pilot.

Windshield Anti-Ice

38. The windshield anti-ice switches are lever locked to prevent inadvertent activation to the "HIGH" position. However, it is easy to inadvertently bump them to the "NORMAL-ON" position resulting in a distorted view through the windshield as the pilot lands the aircraft. The windshield anti-ice switches should be lever-locked to the "NORMAL" as well as the "HIGH" position to prevent inadvertent activation of the windshield heating system and the resultant visual distortion.

Master Caution Light

39. The master caution light and the left/right prop pitch segmented caution advisory light illuminates when the power levers are placed to the ground fine position and the propeller blade pitch angle changes to 12.5°. The frequent illumination of the master caution light during ground taxi is distracting and will lead the flight crew to disregard the warning system. The propeller blade angle change should be an advisory only light and not cause the master caution light to illuminate. The illumination of the master caution light with propeller blade angle changes during ground taxi is a shortcoming.

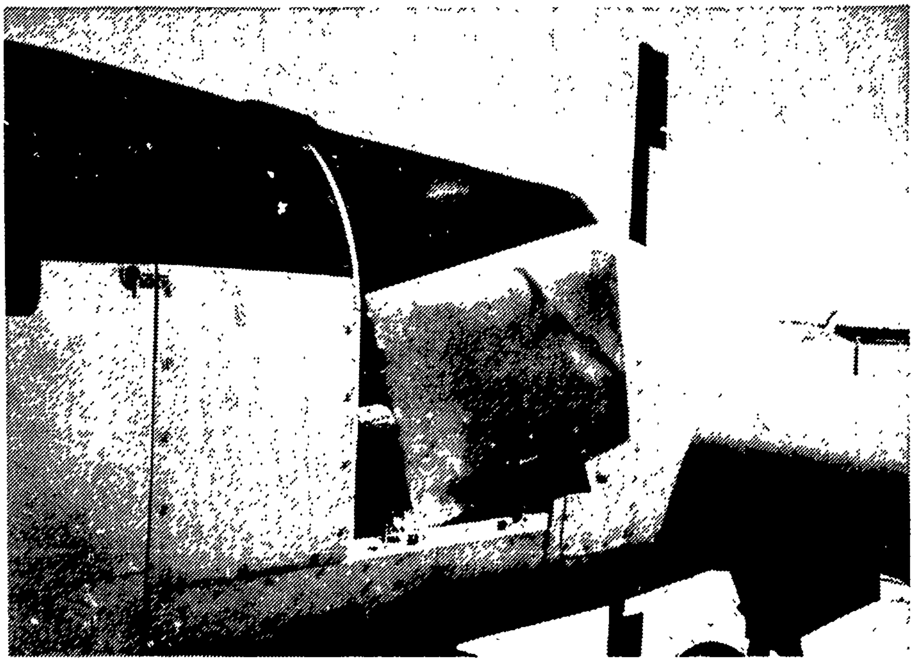


Figure 1. Engine Cowling

Speed Control

40. The airspeed difference between maximum speed in level flight with maximum continuous power (V_H) and maximum operating limit speed (M_{mo}) at mission altitude is only about 20 knots. M_{mo} was determined by stabilon flutter requirements. Exceeding M_{mo} could result in structural failure and loss of airframe integrity. It is easy to exceed M_{mo}, especially when descending out of altitude. The following **CAUTION** should be placed in chapter 8 of the operator's manual alerting the pilots to frequently crosscheck the airspeed and Mach limit indicator to avoid exceeding M_{mo} during descent.

CAUTION

> M_{mo} may be easily exceeded when descending from high altitude. The pilot should frequently cross check the airspeed and Mach limit indicators to avoid exceeding M_{mo}. Exceeding M_{mo} could result in structural failure and loss of airframe integrity.

Control Oscillations

41. Occasionally, with the autopilot engaged, self–exciting, small magnitude roll control oscillations were observed. Control yoke lateral displacements of approximately ±1/4-in. and aileron control surface displacements of approximately ±1/8-in. were noted. The oscillations just barely effected a roll attitude change. Time histories of roll control oscillations are presented in figures E-20 and E-21.

42. Rudder pedal oscillations of ±1/4-in. were also occasionally observed with flaps extended full down. The rudder pedal oscillations were not associated with the aileron control oscillation and occurred with autopilot ON and OFF. A time history of the rudder oscillations is presented in figure E-22. The autopilot induced lateral and directional control oscillations should be evaluated to determine if the oscillations occur on all RC-12K aircraft or just the test aircraft. The autopilot should be adjusted/repaired to eliminate the intermittent oscillations.

Airspeed Calibration

43. Prior to the start of the PAE, an airspeed calibration was conducted on the RC-12K to verify the standard ship's airspeed position errors provided by BAC. A calibrated T-34C aircraft, which was furnished by AEFA, was used to pace the test aircraft. The test results are presented in figure E-23. In the cruise configuration (gear up, flap 0%), BAC's furnished curve is acceptable. However, in the other configurations, a difference of approximately 4 knots greater position error at the lower airspeeds can be seen. According to BAC Type Inspection Report (ref 5), their airspeed calibration was performed with gear up in all configurations while AEFA's airspeed calibration was conducted with the test aircraft in the proper gear/flap configuration. The difference in gear position may account for the different results.

Additional testing to determine the proper position error of the standard ship's airspeed system should be conducted with the correct gear/flap configuration and without a nose boom system installed.

CONCLUSIONS

GENERAL

44. Operator's manual takeoff, accelerate-go accelerate-stop, and landing performance data is conservative (paras 8, 9, 10 and 11).

45. The RC-12K's climb performance and two engine service ceiling is significantly better than presented in the operator's manual (para 12).

46. The RC-12K's takeoff, accelerate-stop, landing, climb performance, and rudder boost system enhances safe mission accomplishment (paras 8, 9, 10, 12 and 32).

DEFICIENCIES

47. The poor engine cowl securing device (para 36).

48. The inadequate artificial stall warning in the cruise (CR), takeoff (TO) and landing (L) configurations (para 15).

SHORTCOMING

49. The illumination of the master caution light with propeller blade angle changes during ground taxi (para 39).

SPECIFICATION COMPLIANCE

50. The artificial stall warning system failed to meet the requirements of paragraph 3.4.2.1.1 of MIL-F-8785C in that the minimum stall warning is less than 5 knots for the CR, TO and L configurations (para 15).

RECOMMENDATIONS

51. The deficiencies identified in paragraphs 47 and 48 should be corrected prior to conduct of operational missions.

52. The shortcoming identified in paragraph 49 should be corrected as soon as practical.

53. The RC-12K should incorporate an anti-skid system to improve wet/slippery runway stopping performance and aircraft control, and to preclude inadvertent wheel lockup during conduct of maximum performance landing and rejected takeoff procedures (paras 10 and 11).

54. The operator's manual should contain a time, fuel, and distance to climb chart for best rate of climb speeds as it does for the cruise climb schedule (para 12).

55. The RC-12K should incorporate an angle-of-attack system that provides a constant indication to the pilot of his operational margin above the stall and provides information for operation at optimum performance angle-of-attack (para 15).

56. The speed command computer should be programmed to provide correct approach speed commands (para 35).

57. The engine cowling latching devices should be redesigned to provide more positive latching security (para 36).

58. The windshield anti-ice switches should be lever locked to the "Normal" as well as the "High" position to prevent inadvertent activation of the windshield heating system (para 38).

59. The propeller blade angle change should be an advisory only light and not cause the Master Caution light to illuminate (para 39).

60. The operator's manual should contain the following **CAUTION** in chapter 8 (para 40).

CAUTION

> M_{mo} may be easily exceeded when descending from high altitude. The pilot should frequently cross check the airspeed and Mach limit indicators to avoid exceeding M_{mo}. Exceeding M_{mo} could result in structural failure and loss of airframe integrity.

61. The autopilot induced lateral and directional control oscillations should be evaluated to determine if the oscillations occur on all RC-12K aircraft or just the test aircraft (para 42)

62. The autopilot should be adjusted/repaired to eliminate the intermittent oscillations (para 42).

63. Additional testing to determine the proper postion error of the standard ship's airspeed system should be conducted with the correct gear/flap configuration and without a nose boom system installed (para 43).

APPENDIX A. REFERENCES

1. Letter, AVSCOM, AMSAV-8, 20 December 1988, subject: Preliminary Airworthiness Evaluation of the RC-12K aircraft. (Test Request)

2. Technical Manual, TM 55-1510-222-10, *Operator's Manual for Army RC-12K Aircraft*, Preliminary Draft, unpublished.

3. Beech Specification, BS-24123, Revision A, *Model Specification Model RC-12K Guardrail/Common Sensor*, 19 August 1987.

4. Military Specification, MIL-F-8785C, *Flying Qualities of Piloted Airplanes*, 5 November 1980.

5. Department of Transportation, Federal Aviation Administration Type Inspection Report, Beech A200CT (RC-12K), 1 March 1989.

6. Letter, AVSCOM, AMSAV-E, subject: Airworthiness Release, RC-12K (SN 85-0149) Preliminary Airworthiness Evaluation, AEFA Project No. 88-03

7. Flight Test Manual, Naval Air Test Center, FTM No. 104, *Fixed Wing Performance*, July 1977.

8. Flight Test Manual, Naval Air Test Center, FTM No. 103, *Fixed Wing Stability and Control*, 1 January 1975.

9. Final Report, AEFA Project No. 87-11, *Preliminary Airworthiness Evaluation of the RC-12H Airplane*, June 1988.

APPENDIX B. DESCRIPTION

GENERAL

1. The RC-12K (Beech A200CT), specified in Beech Model Specification BS 24123 Revision A (ref 3, app A) is modified specifically for the Guardrail/Common Sensor System 4 mission. The RC-12K is physically and functionally identical to the RC-12D aircraft specified in Beech Model Specification BS 23525 and modified to the RC-12H aircraft specified in Beech Model Specification BS 23938 with the following exceptions:

a. 16,000 lb to takeoff gross weight, RC-12H is 15,000 lb.

b. 12,700 lb zero fuel weight, RC-12H is 11,500 lb.

c. PT6A-67 engines flat rated at 1100 shaft horsepower, RC-12H has PT6A-41 engines rated at 850 shaft horsepower at sea level standard day conditions.

d. McCauley 4-blade, 105 in. diameter propellers, RC-12H has Hartzell 3-blade, 98.5 in. diameter propellers.

e. Pitot engine cowls, RC-12H has standard chin inlet cowls.

f. Pneumatically operated engine fire detection system, RC-12H had infrared detectors.

g. Engine bleed air flow control units are electrically controlled, RC-12H bleed air flow control units are pneumatically controlled.

h. D-4 exhaust stacks, RC-12H has standard stacks.

i. NACA engine oil cooler air inlet, RC-12H uses engine inlet air for oil cooling.

j. Certified service ceiling is 35,000 ft, RC-12H is 30,000 ft.

k. Cabin pressurization has 6.5 pounds per square inch (psi) differential, RC-12H has 6.0 psi.

l. Heavy gross weight wheel brakes similar to the Beech Model 1900.

m. Hydraulically actuated landing gear system, RC-12H has electrically driven landing gear system.

n. Stabilon installation on the aft fuselage.

o. Vortex generators on the left outboard flap.

p. Right wing stall strip moved up 0.5 in.

q. Stall warning system lift computer changed to meet FAR Part 23 requirements.

r. Rudder boost system is TORQUE sensing and boost is provided by the Sperry Autopilot rudder servo, RC-12H has pneumatic rudder boost system.

s. Elevator trim tab trailing edge travel is 15° down, RC-12H is 13° down.

t. Rudder trim tab anti-servo gearing is 7°, RC-12H is 10°.

u. Aileron mass weight is increased.

v. Shear fitted three element wing spar, RC-12H has tension bolt fitting style wing attachment.

w. Strengthened fuselage aft section.

x. Strengthened fuselage nose section.

y. Increased number of structural rivets in outboard wing spar web splice.

z. Beefed up jack points.

aa. Eliminate fuselage cabin windows, RC-12H has painted cabin windows.

bb. Different left avionics nose door, RC-12H incorporates TSEC access door.

cc. Uses Model 300 type horizontal stabilizer deice boots, RC-12H uses standard Model 200 style horizontal stabilizer deice boot.

dd. .Increased cockpit cooling with added floor outlets and larger overhead eyeball air outlets.

ee. Lighter weight sound deadener material.

ff. All digital electrical meters in the cockpit overhead switch panel, RC-12H has analog meters.

gg. 42 channel caution annunciator panel, RC-12H has 36 channel caution annunciator.

hh. Cockpit infrared floodlighting.

ii. Chemical toilet, RC-12H has electric flush toilet.

jj. Automatic direction finding (ADF) sense antenna removed from dorsal fin.

kk. Incorporates ARC-201 radio, RC-12H incorporates ARC-186 radio.

ll. Incorporates (2) VHF-22B radios and (2) VIR-32 NAV radios, RC-12H incorporates (1) VHF-20 and (2) VIR-30 NAV radios.

mm. Incorporates a Collins ADF-60, RC-12H incorporates a Collins DF-203 ADF radio.

nn. Incorporates an ARN-136A TACAN system, RC-12H incorporates an ARN-136 system.

oo. Incorporates (3) KY-58 radios.

The RC-12K is a pressurized, low wing, all metal aircraft, powered by two PT6A-67 turboprop engines and has day and night all weather capability. The overall aircraft dimensions are shown in figure B-1. Four views of the test aircraft are shown in figures B-2 through B-5. Specific dimensions and general data are presented in table B-1. Locations of aircraft antenna on the test aircraft are shown in figures B-6 through B-9. A detailed description of the RC-12K is contained in the Beech Model Specification BS 24123 (ref 3) and TM 55-1510-22-10, Operator's Manual for Army RC-12K Aircraft, Preliminary Draft (ref 2).

AIRFRAME

2. The RC-12K airframe is an all metal design conventional semi-monocoque structure. It incorporates a shear fitted three element wing spar and strengthened nose and tail sections allowing increased operational gross weights. The maximum zero fuel weight is 12,700 lb with a maximum ramp weight of 16,090 lb. The maximum takeoff and landing gross weights are 16,000 and 15,200 lb, respectively. The maximum operating altitude is 35,000 ft. The crew and passenger compartments are designed for an internal maximum operating differential pressure of 6.5 $\pm$ 0.10 psi. This provides an 8,000 foot cabin pressure altitude at an operating altitude of 29,700 ft, 10,000 ft cabin pressure altitude when operating at 34,000 ft and 10,400 ft cabin pressure altitude at 35,000 ft. Maximum airspeed (V_{NE}) is 246 knots indicated airspeed (KIAS) or Mach (M_{MO}) .47. The maximum design maneuvering speed is 166 KIAS. Maximum flap speeds are 40% flaps (14° trailing edge down) at 197 KIAS and 100 % flaps full down (35° trailing edge down) at 154 KIAS. Maneuvering load factors are +3.02 g's to -1.21 g's with wing flaps up or at approach and +2.0 g's to 0.0 g with wing flaps full down.

3. The painted cabin windows forward of the aft cabin "porthole" windows on the RC-12H have been replaced on the RC-12K with metal fuselage skin. The total interior space has a volume of 393 cubic ft with 299 cubic ft available for mission equipment. The main cabin entrance "airstair" door is located on the left side of the aircraft and is part of the cargo door. The "airstair" door is hinged from the bottom and provides an opening of 21.5 in. wide by 49 in. high. The cargo door is hinged from the top and is 52 in. wide by 52 in. high. Emergency egress is provided by a 19.75 in. wide by 26 in. high hatch located on the right side of the aircraft above the wing.

4. The empennage consists of a T-tail with vertical taillets attached to the outboard, lower surface of the fixed horizontal tail. Nonmovable stabilons are attached on each side of the lower empennage below the vertical fin at fuselage station 403.141, waterline 118.961. The stabilons are a modified NACA 0012-64 airfoil installed at 3° angle of incidence to the fuselage waterline. The stabilons total area is 11.41 square ft

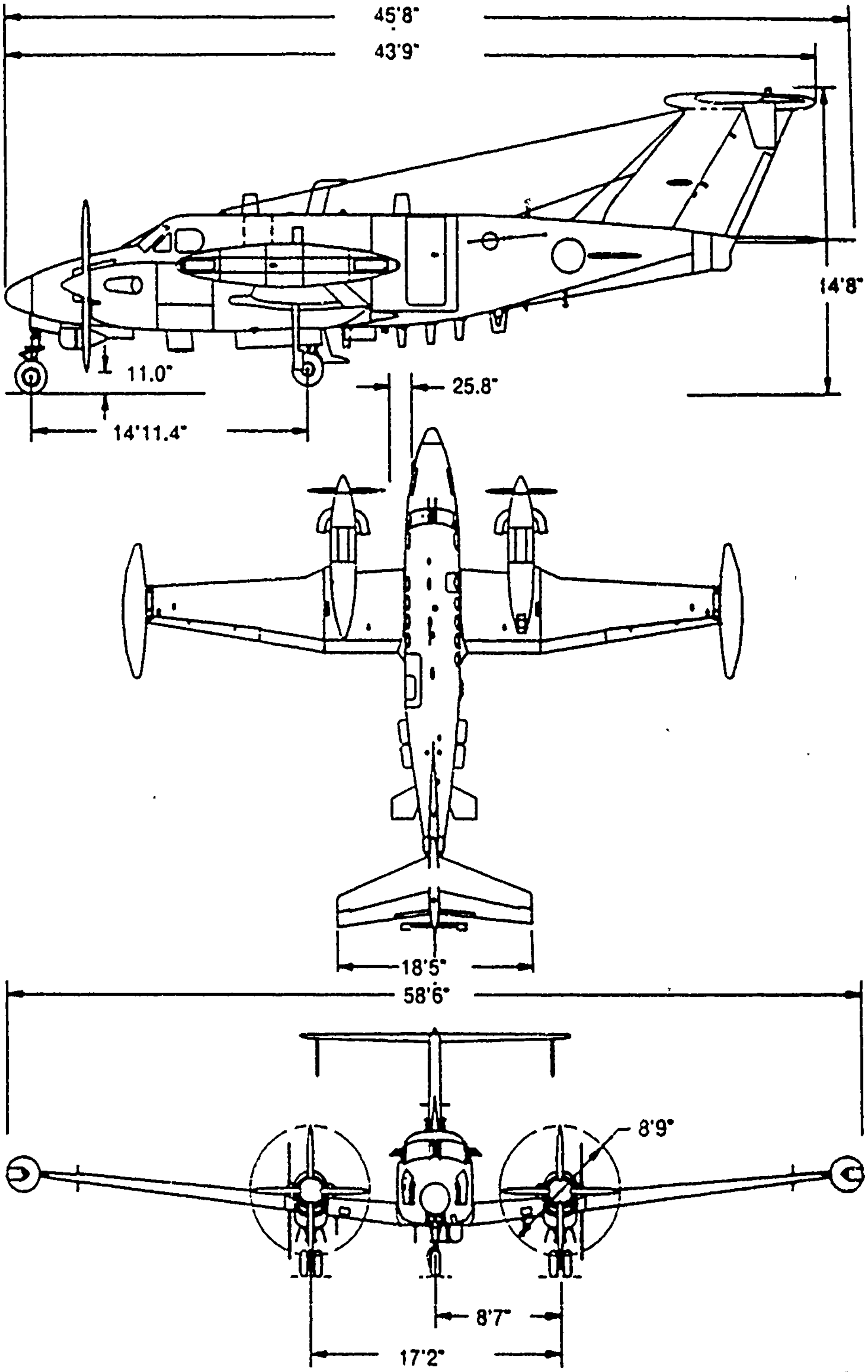


Figure B-1. Aircraft Dimensions

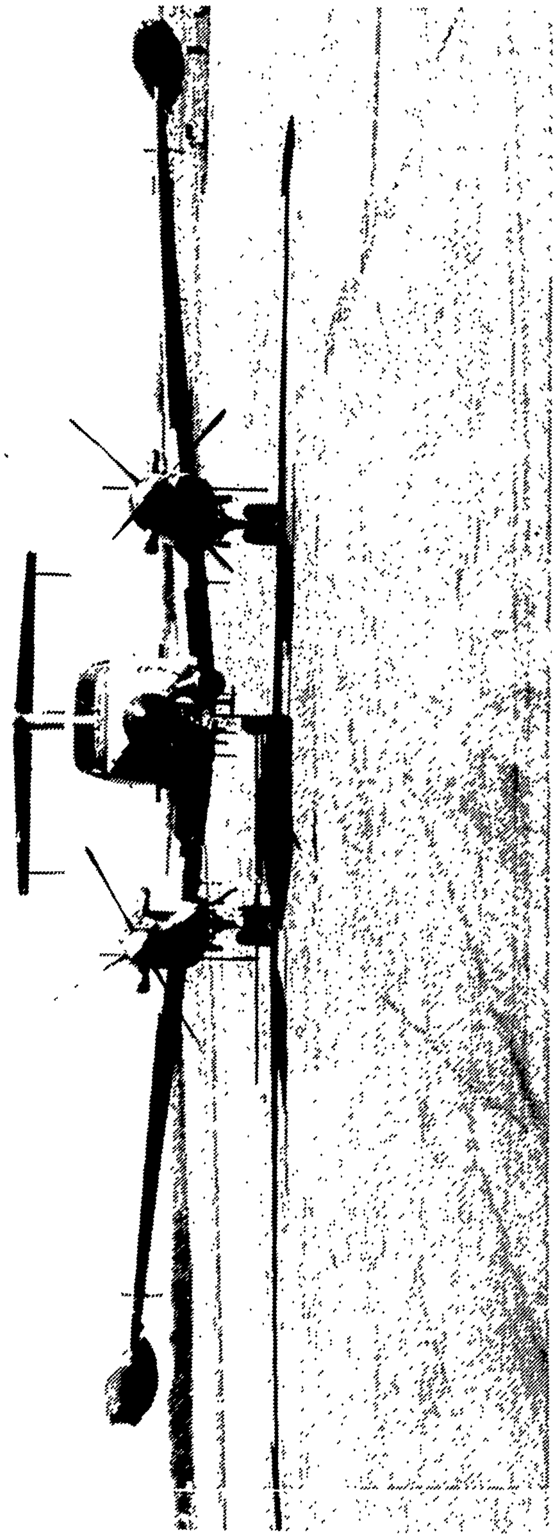


Figure B-2. RC-12K, Front View

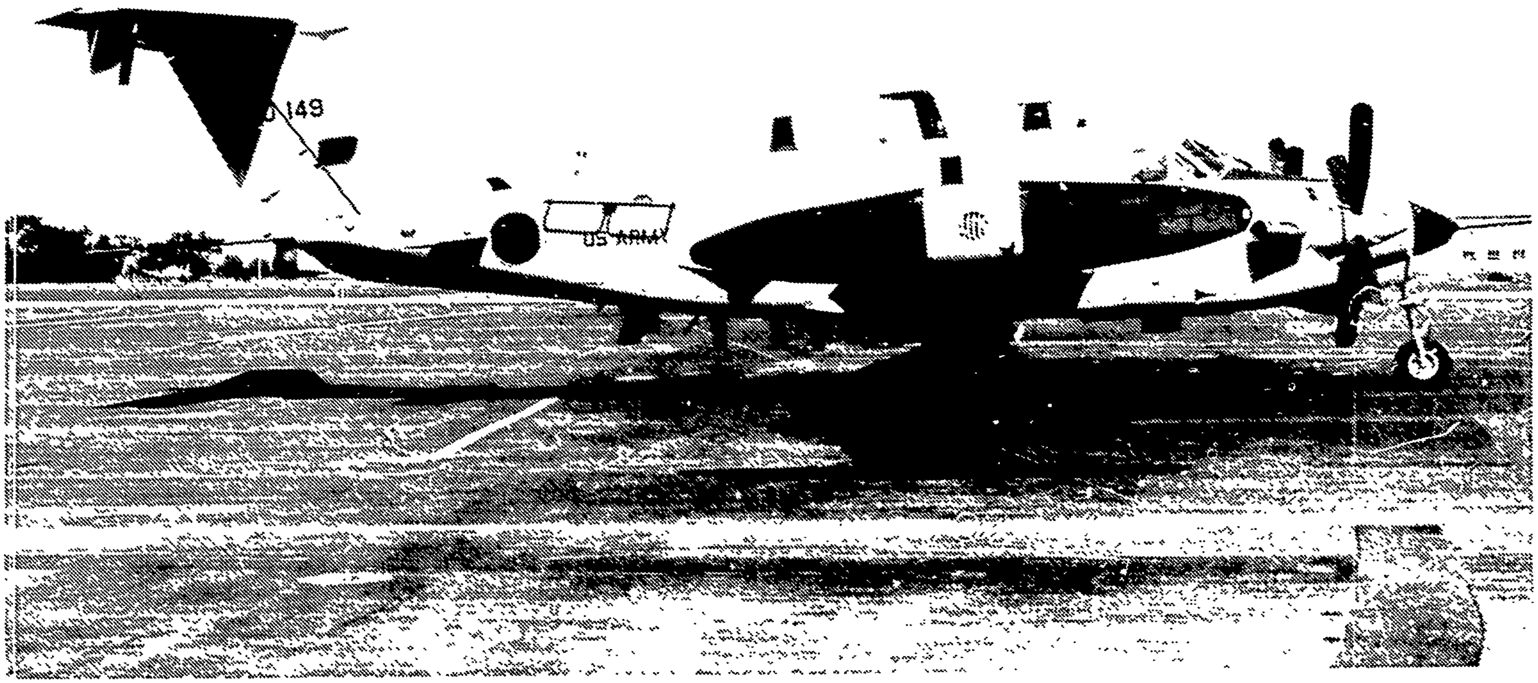


Figure B-3. RC-12K. Right Side View

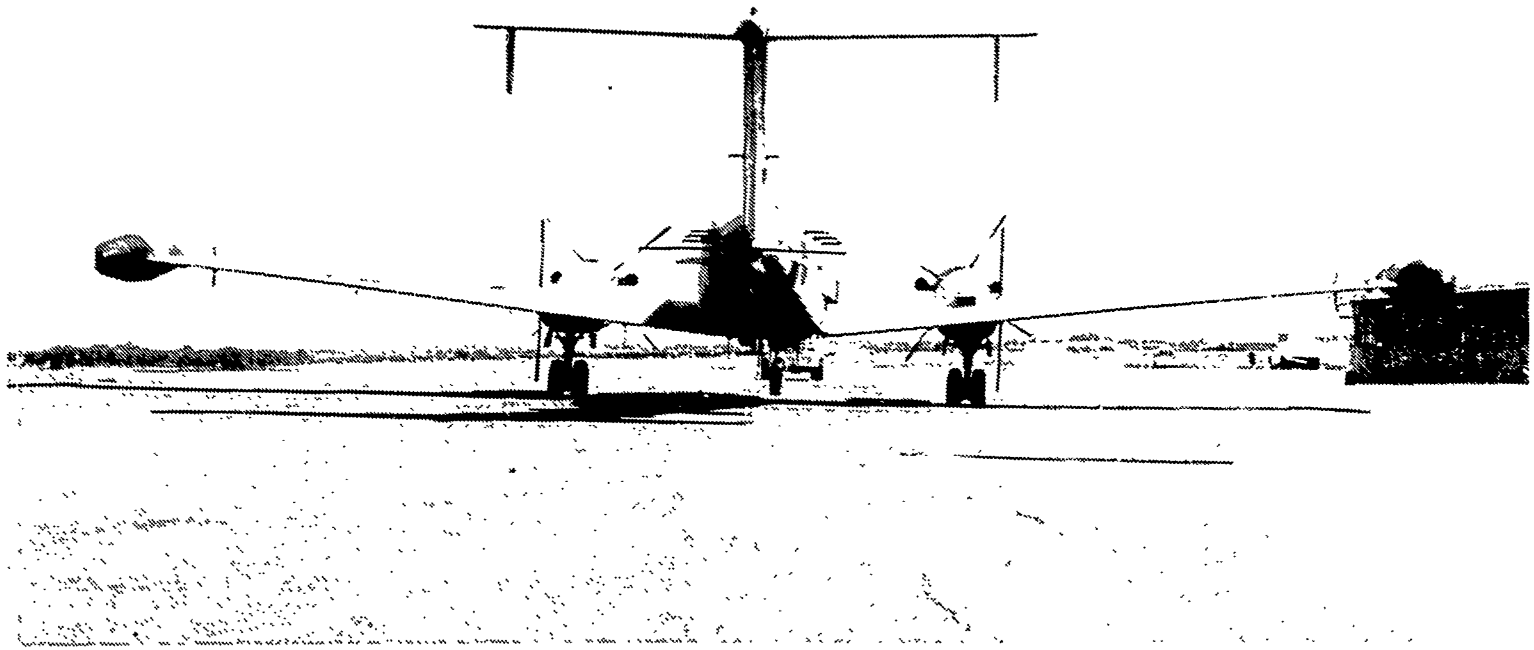


Figure B-4. RC-12K. Rear View

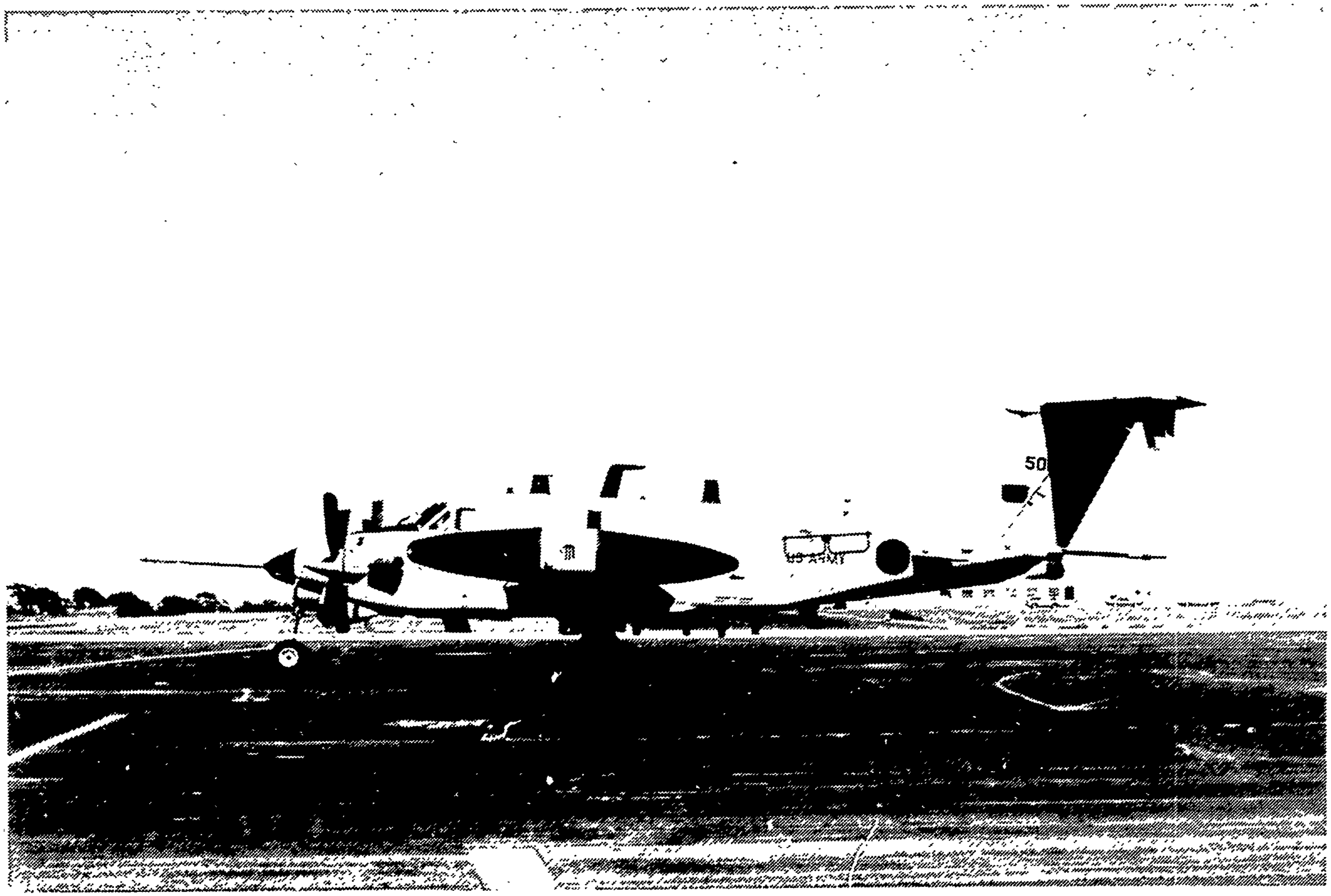


Figure B-5. RC-12K, Left Side View

Table B-1. Dimensions and General Data

1. The following dimensions and data are for descriptive purposes and are not to be used as inspection criteria.

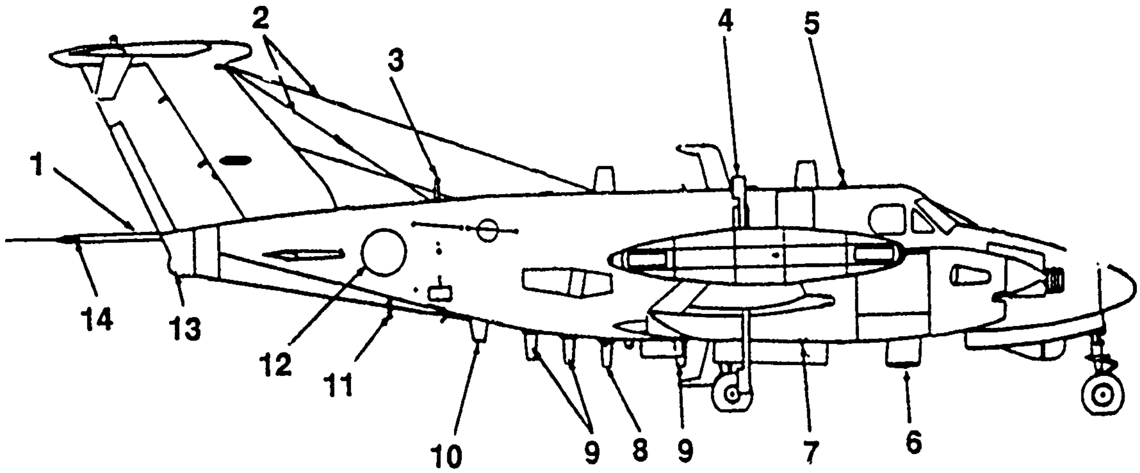
Wing	
Span, maximum	58.5 ft
Chord	
At root (centerline of fuselage)	85.75 in.
At root station 123.99 (disregarding leading edge edge extension)	79.07 in.
At root station 328.74	35.64 in.
Mean Aerodynamic	70.41 in.
Leading edge of mean aerodynamic chord	Fus Sta 171.23
Airfoil section designation:	
At station 25	NACA 23018 (Modified)
At station 298.74	NACA 23012
Incidence (degrees)	
At root (theoretical centerline of fuselage)	3.48°
At Station 328.74	-1.07°
Sweepback:	
Outer panel at 25 percent chord	0°
Center section at 100 percent chord	0°
Dihedral, degrees	6.0°
Aspect ratio	9.8
Height over highest fixed part of aircraft (tail) (airplane in normal ground attitude)	14.67 ft
Length, maximum (normal ground attitude)	45.67 ft
Distance from wing MAC quarter chord point to horizontal tail MAC quarter chord point	25.19 ft
Distance from wing MAC quarter chord point to vertical tail MAC quarter chord point	20.96 ft
Angle between reference line and wing zero–lift line	-2°
Ground angle, degrees (static condition)	2.55°
Propeller clearance, (normal design) loading condition reference line level	14.04 in.
Propeller diameter	105 in.
Wheel size	
Main Wheels	6.50 x 10
Nose Wheels	6.50 x 10
Tire size	
Main Wheels	22 x 6.75 x 10
Nose Wheels	22 x 6.75 x 10
Tread of Main Wheels	17.2 ft
Wheel Base	14.9 ft

Vertical travel of axle from extended to fully compressed position
Main Wheels .... 17.95 in.
Nose Wheel .... 10.11 in.
Distance from main wheel contact point to center of gravity
Horizontal distance parallel to ground
At most forward cg at gross weight .... 19.0 in.
At most aft cg at gross weight .... 11.4 in.

2. The following control surfaces and control movements information are for descriptive purposes and are not to be used as inspection criteria..

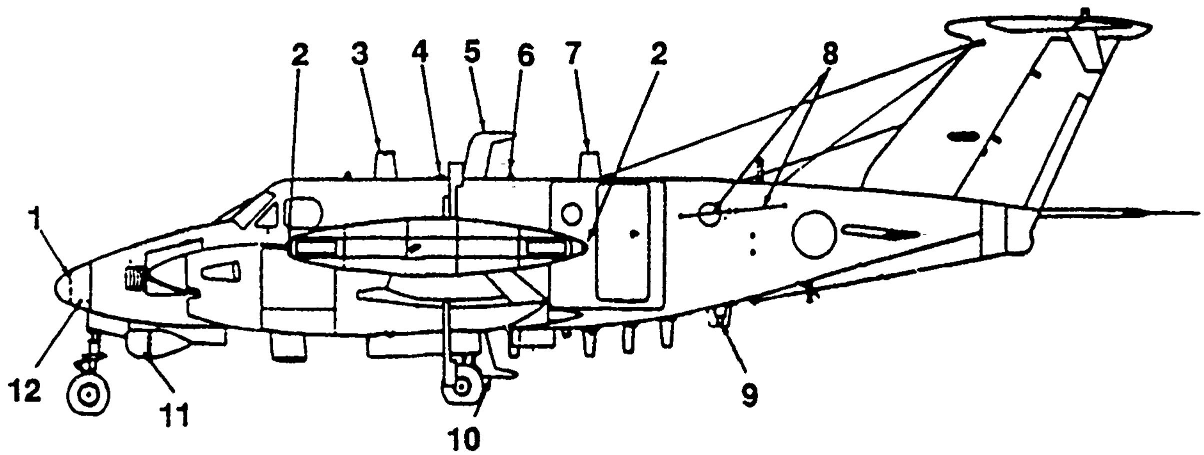
Control and control surface movements on each side of neutral position for full movement, as limited by stops.

Rudder	25° right, 25° left
Rudder Pedal	3.82 in. forward 3.46 in. aft
Rudder tab or trim surface	15° right, 15° left
Rudder tab or trim surface control	4 turns for 30° of tab or trim surface movement
Elevators	20° above, 14° below
Elevator control	4.35 in. aft, 2.0 in. forward
Elevator tab	3.5° above, 13° below
Elevator tab control .... of time to trim through full range,	2.75 turns for 47 seconds 16.5° movement
Ailerons	24° trailing edge up 16° trailing edge down
Aileron control wheel	70° right, 70° left
Aileron tab control	4 turns for 3° tab movement
Wing flap (maximum)	35°
Aileron tab or trim surface	15° trailing edge up 15° trailing edge down



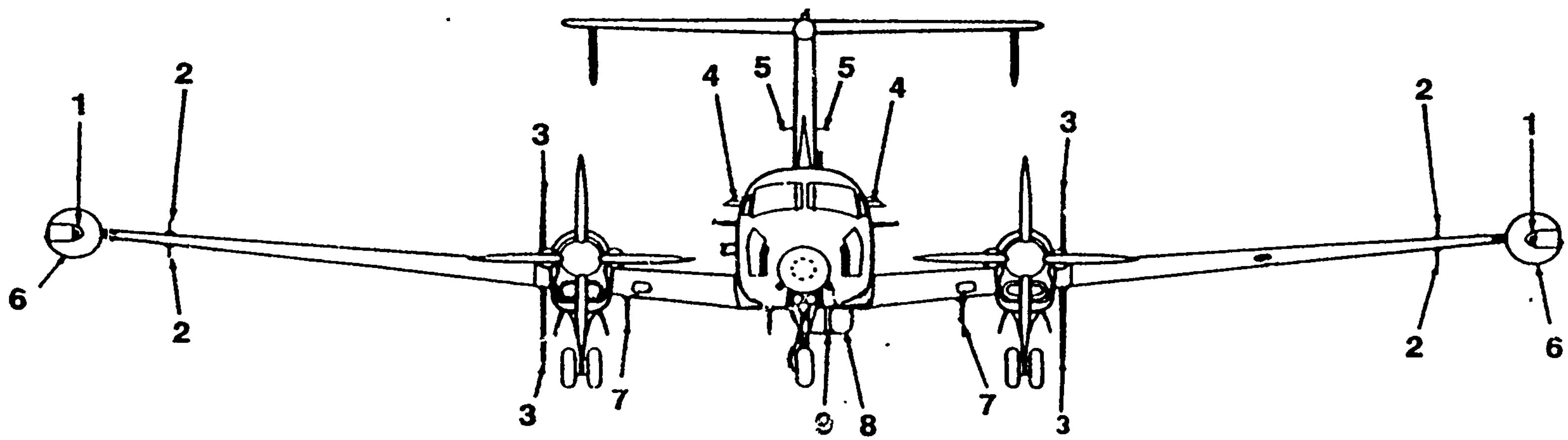
1. Aft Rotating Boom Antenna
2. HF Long Wire Antenna
3. AN/APR-44 Antenna
4. Low Band Dipole Antenna
5. TACAN Antenna
6. High Band Vert & Horiz Antenna
7. AN/APR-39 Blade Antenna
8. INS/TACAN Antenna
9. High Band Monopoles
10. VHF Comm Antenna
11. AN/APR-44 Antenna
12. "P" Band Antenna
13. Wide Band Data Link Aft Antenna
14. Mid Band Dipole Antenna

Figure B-6. Antenna Locations (Right Side View)



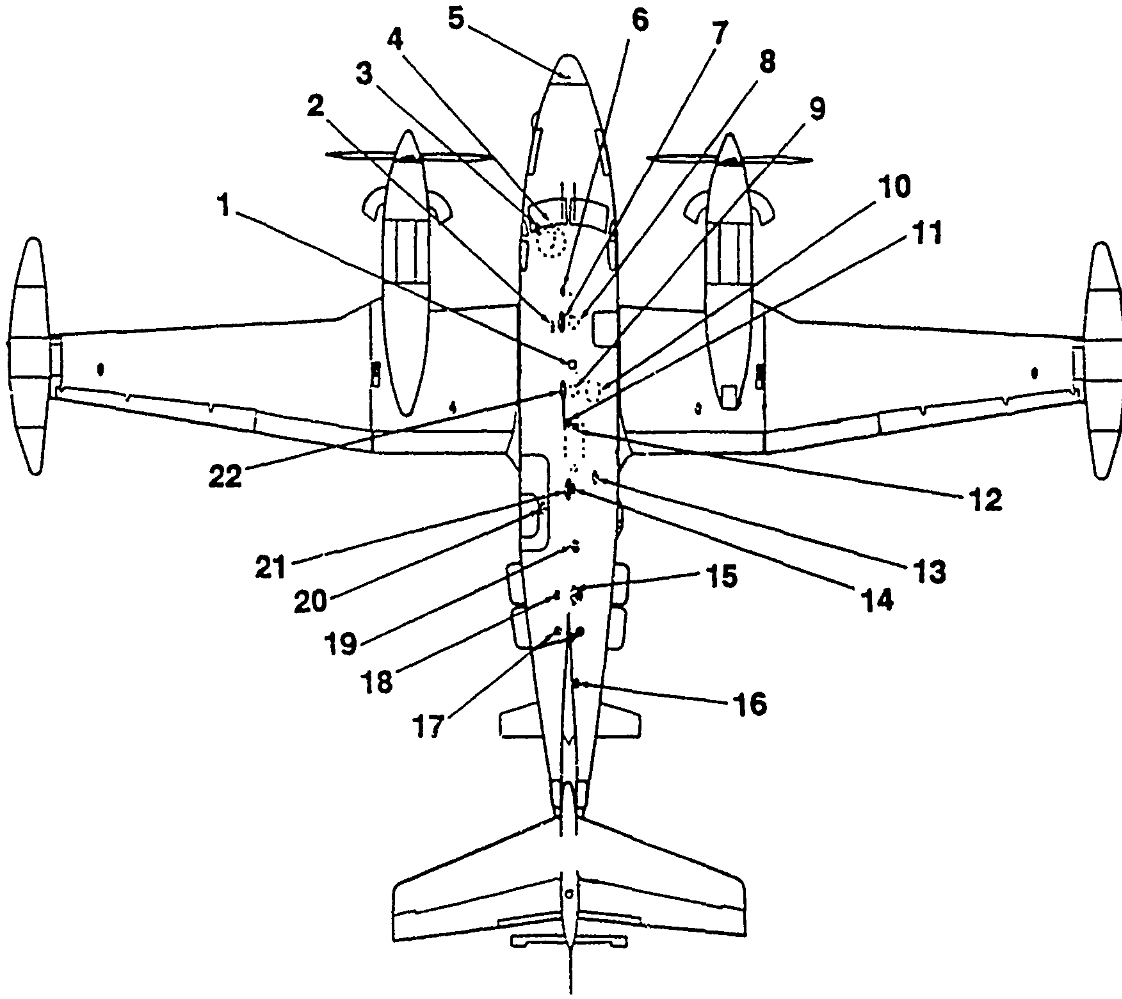
1. Weather Radar Antenna
2. AN/APR-39 Spiral Antenna
3. SINCGARS Antenna
4. Global Positioning System Antenna
5. Low Band Vertical Bent Blade
   (upper) Antenna
6. Transponder Antenna
7. VHF COMM Antenna
8. Low Band Horiz Towel Bar Antenna
9. Transponder
10. Low Band Vert Bent Blade
11. (lower) Antenna
   Wide Band Data Link Fwd Antenna
12. Glideslope Antenna

Figure B-7. Antenna Locations (Left Side View)



1. AN/APR-39 Spiral Antenna
2. UHF COMM & Intercept Antenna
3. Mid Band Dipole Antenna
4. Low Band Horiz Towel Bar Antenna
5. VOR NAV/LOC Antenna
6. ELINT & DF Antenna Pod
7. High Band Monopole Antenna
8. Wide Band Data Link Fwd Antenna
9. High Band Vert & Horiz Antenna

Figure B-8. Antenna Locations (Front View)



Note: Dashed lines indicate antenna located on bottom of aircraft.

1. Global Positioning System Antenna
2. AN/APR-39 Blade Antenna
3. High Band Vert & Horiz Antenna
4. Marker Beacon Antenna
5. Glideslope Antenna
6. TACAN Antenna
7. ~~SINCGARS Antenna~~
8. Radio Altimeter Antenna
9. Radio Altimeter Antenna
10. ADF Loop Antenna
11. Transponder Antenna
12. High Band Monople Antenna
13. High Band Monople Antenna
14. INS/TACAN Antenna
15. VHF COMM Antenna
16. ELT Antenna
17. AN/APR-44 Antenna
18. ~~UHF/Transponder~~
19. High Band Monopole Antenna
20. High Band Monopole Antenna
21. VHF COMM Antenna
22. Low Band Vert Bent Blade (upper) Antenna

Figure 3-9. Antenna Locations (Top View) and are used to enhance the center of gravity envelope and to meet Federal Aviation Regulations static longitudinal stability requirements.

LANDING GEAR

5. The aircraft is equipped with a retractable tricycle landing gear suitable for unimproved field operations. The single nose gear tire and dual main gear tires are 22 x 6.75 – 10 tubeless with 8-ply rating with normal inflation pressures of 55 to 60 psi for the nose and 73 to 77 psi for the main tires. The main wheels are equipped with multiple-disc, non-boosted hydraulic brakes with metallic lining. The wheel components of the RC-12K are the same as those used on the modified Beech Model 1900 which is certified for a maximum gross weight of 18,700 lb while the brake components are those used on the unmodified Beech Model 1900 certified for a maximum gross weight of 16,600 lb. One brake assembly is provided at each main gear wheel and the two assemblies are interchangeable. The brakes are designed to provide a minimum of 500 stops within 500 flight hours. The brake system is sized to provide maximum braking capabilities with a reasonable pedal travel and force. A force of 75-100 lb at the top of the rudder pedal will produce a locked brake on either hot or cold brakes at speeds from 0 to 40 miles per hour. Braking is permitted from either set of rudder pedals. The toe brake sections of the rudder pedals are connected to the master cylinders which actuate the system for the corresponding wheels. The parking brake handle is located below and on the right side of the left subpanel and is accessible from either cockpit position. The brakes are set by pressing both brake pedals then pulling out the parking brake handle. The parking brake may be set from either cockpit station and is released when the handle is pushed in.

6. The landing gear is electrically controlled and hydraulically actuated. The landing gear assemblies are extended and retracted by a hydraulic power pack, located in the left wing center section forward of the main spar designed to operate at 3,000 psi. The power pack consists primarily of a hydraulic pump, a 28 volt direct current motor, a gear selector valve and solenoid, a two section fluid reservoir, filter screens, a gear-up pressure switch, and a low fluid level sensor. Engine bleed air, regulated to 18 to 20 psi, is plumbed into the power pack reservoir and the system fill reservoir to prevent cavitation of the pump. A hydraulic hand pump is installed for manual extension operation with the pump extension handle located on the floor at the right side of the pilot's seat. The fluid level sensor activates a yellow caution light on the caution/advisory annunciator panel when the fluid level in the power pack is low. The main landing gear retract forward with the wheels slightly exposed below the gear well through cutouts in the landing gear doors when fully retracted. The nose gear retracts aft and is completely enclosed in the nose wheel well when the doors are closed. The nose landing gear incorporates two landing lights and a taxi light. If these lights are inadvertently left ON after takeoff, they will extinguish when the nose gear retracts. The landing gear system utilizes folding braces called drag legs, that lock in place when the gear is fully extended. The nose landing gear actuator incorporates an internal down-lock to hold the gear in the fully extended position. The two main landing gear

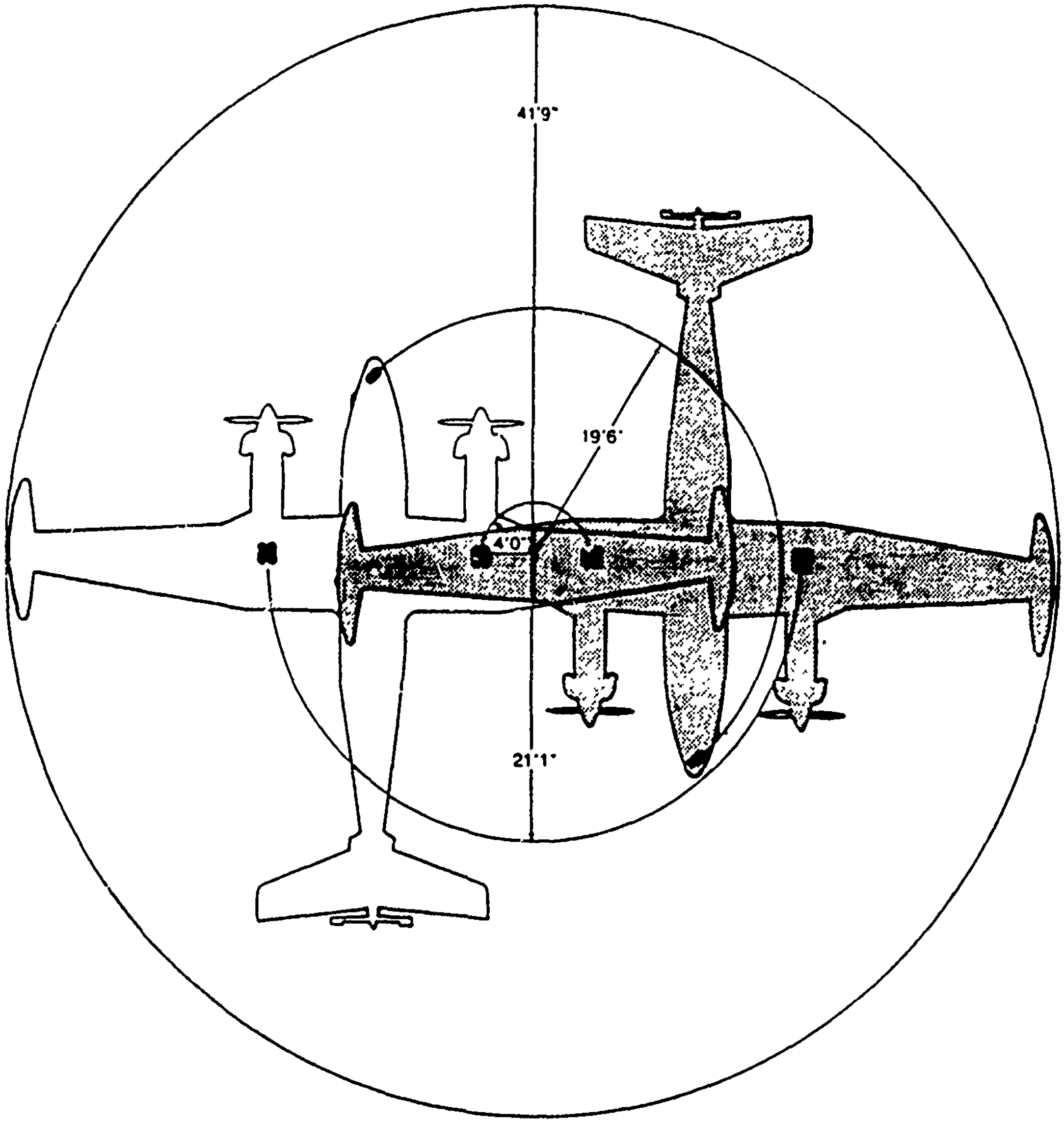
are held in the fully extended position by mechanical hook and pin locks. All three landing gear are held in the up position by hydraulic pressure. The pressure is controlled by the power pack pressure switch and an accumulator that is precharged with nitrogen to 800 ±50 psi. If the pressure switch senses hydraulic pressure less than 2250 ±75 psi, with the landing gear handle up, the electric motor will activate the hydraulic pump increasing the pressure to 2775 ±55 psi, thereby keeping the gear fully retracted. Landing gear extension or retraction is normally accomplished in 6 to 7 seconds. Voltage to the power pack is terminated after the fully extended or retracted position is reached. The power pack is protected by a time delay module which senses operation voltage through a 5-ampere circuit breaker located beneath the aisleway floor forward of the main spar. If electrical power has not terminated within 14 seconds, a relay beneath the aisleway floor, and the 2-ampere landing circuit breaker, on the overhead circuit breaker panel, will open interrupting electrical power to the system power pack. The maximum gear extension speed is 178 KIAS with a maximum gear retraction speed of 160 KIAS. The aircraft can be maneuvered on the ground by the steerable nose wheel system. Direct linkage from the rudder pedals to the nose wheel steering linkage allows the nose wheel to be turned 12° to left of center and 14° to the right. When rudder pedal steering is augmented by main wheel braking, the nose wheel can be deflected up to 48° either side of center. Minimum ground turning radius is depicted in figure B-10. Shock loads which would normally be transmitted to the rudder pedals are absorbed by a spring mechanism in the steering linkage. Retraction of the landing gear automatically centers the nose wheel and disengages the steering linkage from the rudder pedals.

FLIGHT CONTROLS

7. The aircraft's primary flight control system consists of conventional rudder, elevator and aileron control surfaces. These surfaces are manually operated from the cockpit pilot or copilot station through mechanical linkage using a control wheel for the ailerons and elevators and two position adjustable rudder/brake pedals for the rudder. Trim control for the rudder, elevator and ailerons is accomplished through a manually actuated cable-drum system for each set of control surfaces.

8. The control wheels incorporate switches on the outboard grip to electrically operate the elevator trim tabs. An intercom/microphone switch, chaff dispense switch, and an autopilot-yaw damp/electric trim disconnect switch are also installed on the outboard grip of each control wheel. In addition, a transponder ident switch is installed on top of the inboard grip of each control wheel. Installed in the center of each control wheel is a digital electric clock, a map light switch and a touch control steering switch.

9. The rudder is hinged to the trailing edge of the vertical stabilizer and incorporates a 56.75 in. span adjustable trim tab. The trailing edge of the rudder above the trim tab has a 22.75 in. span symmetrical bulge which helps prevent rudder lock. The trim tab is manually activated by a rudder trim wheel located on the control pedestal and mechanically controlled by a cable-drum and jackscrew actuator system. The trim tab



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..41 ft 9 in

Turning radii are predicated on the use of partial braking action and differential power.

Figure B-10. Ground Turning Radius

incorporates anti-servo action, i.e., as the rudder is displaced from the neutral position the trim tab moves in the same direction as the control surface. This action increases control pressure as the rudder is deflected from the neutral position. The rudder control system incorporates an electrically powered servo in the rudder cable system to improve directional stability. The system must be deactivated for takeoffs and landing and may be used at any altitude but is required for flight above 17,000 ft. The system is controlled by a YAW DAMP switch located on the autopilot control panel. A green advisory light located above the pilot's vertical situation indicator illuminates when the yaw damper is engaged. A rudder boost system is provided to reduce rudder forces required to maintain directional control resulting from an engine failure or large differences in power between the engines. The system automatically makes rudder inputs through the yaw damper servo when a torque difference of approximately 60% between the two engines is detected. If the yaw damper is engaged when this occurs, it will automatically disengage. Delta engine torque and airspeed signals are input to the system. Forward rudder displacement toward the high torque engine increases as delta torques between the two engines increase and the forward rudder decreases displacement as airspeed increases. The rudder boost system is armed by a RUDDER BOOST switch located on the pedestal extension. A yellow caution light located above the pilot's vertical situation indicator illuminates when the rudder boost system is deactivated.

10. A two-piece elevator is hinged to the trailing edge of the fixed horizontal stabilizer mounted on top of the vertical stabilizer in a "T" tail shaped arrangement. Adjustable trim tabs are installed on each elevator. The elevator trim tabs incorporate neutral, non-servo action, i.e., as the elevators are displaced from the neutral position, the trim tab maintains in an - as adjusted - position. The elevator trim tab control wheel is located on the left side of the control pedestal and controls the trim tab on each elevator. Electric elevator trim is controlled by moving the elevator trim tabs through an ELEV TRIM switch located on the pedestal extension adjacent to the RUDDER BOOST switch. With this switch ON, the elevator may be trimmed electrically by actuating the pitch trim switches on the pilot's or copilot's control wheel. This activates an elevator trim servo installed in the elevator trim system. If the electric trim is turned off by using the switch on the pedestal extension or depressing the autopilot/yaw damp/trim button on the control wheel through the second detent, the green ELEC TRIM OFF advisory light on the caution/advisory panel will illuminate. To reengage the electric trim system, the ELEV TRIM switch on the pedestal extension must be cycled from OFF/RESET to ON.

11. The ailerons are symmetrical sections except in the wing tip area. They are attached to the wing at three hinge points and have overhanging aerodynamic balance. The aileron mass weight is increased as compared to that of the RC-12H. Each aileron incorporates a symmetrical bulge at the trailing edge located near the outboard span which aids in aerodynamic centering. An adjustable trim tab is located on the left aileron, positioned spanwise at the inboard end of the aileron. The aileron system also

incorporates an electrically actuated roll servo which operates when the autopilot is engaged.

12. The slot-type wing flaps are electrically operated and consist of two sections for each wing. These sections extend from the inboard end of each aileron to the junction of the wing and fuselage. During extension or retraction, the flaps are operated as a single unit, each section being actuated by a separate jackscrew actuator. The actuators are driven through flexible shafts by a single reversible electric motor. Wing flap movement is indicated in percent of travel by a flap position indicator on the forward control pedestal. Approach flaps (40%) equates to 14° trailing edge down. Full flaps (100%) equates to 35° trailing edge down. Full flap extension and retraction time is approximately 11 seconds. Maximum airspeed for approach flaps is 197 KIAS and 154 KIAS for flap setting greater than approach. In the event that two adjacent flap sections extend 3 to 5° out of phase, a safety mechanism is provided to discontinue power to the flap motor. The left outboard flap has Wheeler Vortex Generators (fig. B-11) attached to the leading edge of the flap (not visible with the flaps retracted). This modification along with moving the right wing stall strip up 0.5 in. (fig. B-12) was made to meet Federal Aviation Regulation Part 23 stall requirements.

ENGINES

13. Two United Aircraft of Canada PT6A-67 engines, flat rated at 1100 shaft horsepower, equipped with hydraulically controlled, reversible, constant speed, four-bladed, full-feathering propellers are installed on the RC-12K. The engines are reverse flow, free turbine employing a four stage axial compressor and a single-stage centrifugal compressor in combination, driven by a single stage reaction turbine. A two-stage free power turbine is connected to the flanged propeller shaft through planetary reduction gearing. All accessory pads are AND type. The oil tank, filler cap, dipstick and oil level sight glass are integral parts of the engine. A pneumatic fuel control system schedules fuel flow to maintain power set by the gas generator power lever. The accessory drive at the aft end of the engine provides power to drive the fuel pump, fuel control, oil pump, refrigerant compressor (right engine), starter/generator, and the tachometer generator. The engine oil system, which has a total capacity of approximately 3.5 gallons supplies oil for propeller operation and lubrication of the reduction gearbox and engine bearings. An external engine oil cooler unit (radiator) of fin-and-tube design is located in the lower aft nacelle below the engine air intake. The NACA design oil cooler air inlet is not part of the engine air inlet as it is in the RC-12H; thus, high oil temperature during ground operations due to the engine ice vanes extended has been eliminated. An oil-to-fuel heat exchanger, located on the engine accessory case, operates continuously to heat the fuel sufficiently to prevent ice from collecting in the fuel control unit. Additionally, the fuel control lines are automatically electrothermally heated during engine operation, i.e., there are no manually actuated fuel control heat switches.

14. The engine air inlet is automatically anti-iced by exhaust gases. A small duct facing into the exhaust flow of the engine's left exhaust stack diverts a small portion

Figure B-11. Wheeler Vortex Generators (Left Outboard Flap)

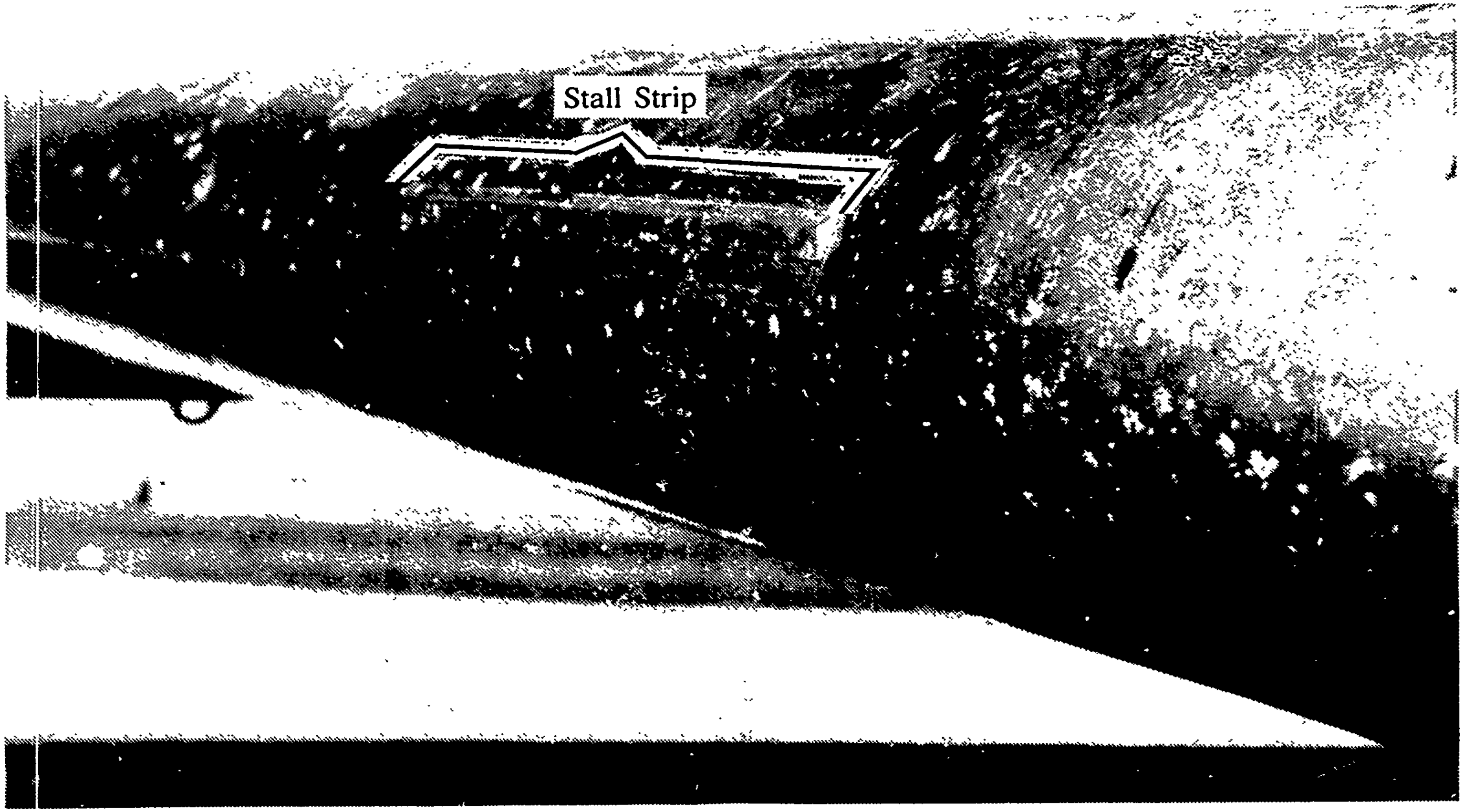


Figure B-12. Right Wing Stall Strip

of the engine exhaust gases to the engine air inlet anti-ice lip. The gases are circulated through the engine air inlet and exhausted through a duct in the engines right exhaust stack. Two ENG INLET LIP HEAT switches on the overhead cockpit control panel are part of an infrared (IR) suppression kit. Although this IR kit was not installed on the test aircraft, when installed these switches will enable the pilot to shut off the engine inlet lip anti-ice thereby reducing the IR signature.

15. The engine is protected from foreign object damage during ground operations and moisture and ice ingestion during flight by an inertial separation system built into the engine air inlet. The system consists of a movable vane and bypass door which are lowered into the engine inlet airstream during ground operations or flight in visible moisture at 5°C or colder. The vane deflects the ram airstream downward introducing a sudden turn in the airstream prior to going into the engine. This causes the heavier than air particles to continue undeflected and be discharged overboard. The vane and door are extended and retracted by an electric actuator controlled by the #1 and #2 VANE CONTROL cockpit overhead switches. There are two identical electric actuators for each engine ice vane system. In the event the MAIN actuator fails, the STBY actuator can be selected by the ICE VANE POWER SELECT switches on the overhead cockpit control panel. Green advisory lights on the caution/advisory panel illuminate when the ice vanes are extended. If the ice vanes do not attain the selected position within 33 seconds after activation, a yellow #1 or #2 ICE VANE FAIL caution light will illuminate. In this event, the appropriate #1 or #2 ICE VANE POWER SELECT switch should be placed in the STBY position. Once the vane is successfully positioned, the ICE VANE FAIL CAUTION LIGHT will extinguish and the appropriate #1 or #2 VANE EXTENDED advisory light will illuminate.

16. A pneumatic fire detection system is used to provide an immediate warning in the event of a fire or overtemperature condition in the engine compartment. The system operates on the principal of expanding inert and active gases within an inner and outer sealed tube. Fire warning lights are located in fire control T-handles on the instrument panel. Pulling the fire control T-handle will electrically arm the extinguisher system and close the fuel firewall shutoff valve for that particular engine. This will cause the light on the PUSH TO EXTINGUISH FIRE switch and the respective #1 or #2 fuel pressure light in the warning annunciator panel to illuminate. Pressing the lens of the PUSH TO EXTINGUISH FIRE switch will fire the squib on the fire extinguishing agent supply cylinder, expelling the agent into the respective engine compartment. The supply cylinder for each engine is located in the main gear wheel well aft of the engine. The 86 cubic in. cylinder is charged with 2.5 lb of CF_3Br and pressurized to 450 psi. Feeder lines extend from the cylinder through the firewall to the engine compartment where the agent is distributed through seven nozzles.

17. One starter-generator is mounted on each engine accessory drive section. In the starter function; 28 volts DC is required to power rotation. In the generator function each unit is capable of 400-amperes DC output.

18. The basic engine fuel system consists of an engine driven fuel pump, a fuel control unit, a fuel flow divider, a dual fuel manifold, fourteen fuel nozzles and a purge system.

The fuel purge system forces residual fuel from the manifolds into the combustion chamber where it is consumed during engine shutdown. The hydro-pneumatic fuel control unit is mounted on the accessory case of the engine. Two condition levers, one for each engine, are located on the cockpit control pedestal. Each lever starts and stops the fuel supply and controls the idle speed for its engine. In the FUEL CUTOFF position, the condition lever controls the cutoff function of its engine-mounted fuel control unit. From LOW IDLE to HIGH IDLE they control the governors of the fuel control units to establish minimum fuel flow levels. The condition lever is advanced from FUEL CUTOFF to LOW IDLE during the engine start cycle.The LOW IDLE N_1 speed is set at 60 to 62% at standard day sea level conditions. After the start cycle is complete, the condition lever is advanced to HIGH IDLE, 71 to 73% N_1 speed. The HIGH IDLE position is used for all ground and flight operations.

19. The two power levers located on the cockpit control pedestal are used to control engine power which is accomplished through adjustment of the N_1 speed governor in the fuel control. Power is increased when N_1 speed is increased. The fuel controls and throttles are adjusted such that with the throttle at the flight idle gate, the condition lever at LOW IDLE has set the minimum idle fuel flow to achieve 61 to 62% N_1. Initial movement of the power lever forward of the flight idle gate would increase the N_1 speed. However, since ground and flight operations are performed with the condition levers at HIGH IDLE (71 to 73% N_1 speed), initial forward movement of the power lever forward of the flight idle gate does not increase power. The power lever must be advanced to the point equivalent to the HIGH IDLE N_1 speed before further advance will increase power (N_1 speed). The power levers also control propeller reverse pitch. The power lever quadrant has two distinct gates aft of the forward power range. The first gate is the flight idle gate. The propeller is on the flight-idle low-pitch stop and the propeller blade angle is 19.5° with the power lever at the flight idle gate regardless of propeller lever position within the governing range. When the power lever is lifted upward and moved aft across this gate, the engine and propeller are operating in the "ground fine" range. Initial power lever movement aft of this gate decreases the propeller blade angle 7° below the flight-idle low-pitch stop. Further movement aft to maximum ground fine (i.e., the next gate) further decreases the blade angle 9.5°. Additionally, aft power lever movement from the flight idle gate to maximum ground fine decreases N_1 speed from 71 to 73% to approximately 63 to 67%. The second gate is the ground fine gate. The propeller blade angle is 3° with the power lever at the ground fine gate regardless of propeller lever position within the governing range. When the power lever is lifted upward and moved further aft across this gate, the engine and propeller are operating in the reverse range. Moving the power lever aft from the ground fine gate to the maximum reverse stop decreases the propeller blade angel from 3° (the maximum ground fine blade angle) to -9° at the maximum reverse stop. Additionally, aft power lever movement from the ground fine gate to the maximum reverse stop linearly increases N_1 speed from 63 to 67% to 86 to 88%.

PROPELLERS

20. McCauley 4-bladed, 105 in. diameter aluminum propellers are installed on each engine. The propellers are full feathering, constant speed, variable pitch, counterweighted and reversible. They are controlled by engine oil pressure through single-action, engine driven propeller governors. The propeller is 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 reverse position. The propellers have no low rpm (high pitch) stops. Low pitch propeller position is determined by the low pitch stop which is a mechanically actuated hydraulic stop. Beta and reverse blade angles are controlled by the power levers in the ground fine and reverse range. Both manual and automatic propeller feathering systems are provided. Manual feathering is accomplished by pulling the corresponding propeller lever, located on the cockpit control pedestal, aft past a friction detent. To unfeather, the propeller lever is pushed forward into the governing range (1150 $\pm$ 50 to 1700 rpm).

21. The automatic feathering system is designed for use only during takeoffs and landings. It is armed by a switch on the cockpit overhead panel; however, the arming circuit is not complete until the power levers are advanced above 89% N_1 closing a microswitch contact in the throttle quadrant and illuminating the green #1 and #2 AUTOFEATHER advisory lights on the caution/advisory panel. The system will remain inoperative as long as either power lever is below 89% N_1 unless the TEST position of the AUTOFEATHER switch is selected to disable the power lever limit switches. With the power levers advanced above 89% N_1, should the torque for either engine drop to an indication between 14 and 20%, the autofeather system for the opposite engine will be disarmed. Disarming is confirmed when the AUTOFEATHER annunciator light of the opposite engine becomes extinguished. If torque drops further, to an indication between 7 and 13%, oil is dumped from the servo of the affected propeller allowing the feathering spring and counterweights to move the blades into the feathered position.

22. A constant speed governor and an overspeed governor are used to control propeller rpm. The constant speed governor, mounted on top of the reduction gearbox housing controls the propeller through its entire range and is operated by the propeller control lever. If the constant speed governor should malfunction and request more than 1700 rpm, the overspeed governor cuts in at 1802 rpm and dumps oil from the propeller to keep the rpm from exceeding approximately 1802 rpm. A solenoid, actuated by the GOVERNOR TEST switch located on the overhead control panel, is provided to reset the overspeed governor to approximately 1544 to 1584 rpm for test purposes. If the propeller sticks or moves too slowly during a transient overspeed condition causing the overspeed propeller governor to act too slowly to prevent an overspeed condition, the power turbine governor, contained within the constant speed governor housing, acts as a fuel topping governor. When the propeller reaches 106% of N_2 rpm (1802 propeller rpm) the fuel topping governor limits the fuel flow to the

gas generator, reducing N_1 rpm which in turn prevents the propeller from exceeding 1802 rpm. In practicality, the power topping governor would activate prior to the overspeed governor assuming no malfunction has occurred in either system. During operation in the reverse range, the power turbine governor is reset to approximately 95% of propeller rpm (1615 rpm) before the propeller reaches a negative pitch angle. This insures that the engine power is limited to maintain a propeller rpm of somewhat less than that of the constant speed governor setting. The constant speed governor therefore, will always sense an underspeed condition and direct oil pressure to the propeller servo piston to permit propeller operation in ground fine (beta) and reverse ranges with the propeller levers in the high rpm position.

23. The propellers on the RC-12K experience a natural vibratory condition known as the reactionless mode when operating on the ground at less than 1000 rpm and accumulate high stress cycles in this condition particularly with a quartering tail wind. In order to maintain propeller rpm above 1000 rpm during ground operation but still minimize forward thrust to control taxi speed, several procedural and mechanical changes to the propeller control systems were made. First, HIGH IDLE (71 to 73% N_1) is used during all ground operations after engine start. This change alone increases the propeller speed above 1000 rpm, but also increases forward thrust. To reduce forward thrust to manageable taxi speeds two systems are employed: a "beta shift" mechanism and an N_1 speed reduction. The "beta shift" mechanism is internal to the propeller and operates through the power lever linkage. When the power levers are moved aft of the flight idle gate, the propeller blade angle shifts 7 degrees below the flight idle low pitch stop. This results in increasing propeller rpm and decreasing torque. An N_1 speed reduction occurs (71–73% to 63–67%) as the power levers are moved further aft in the ground fine range; however, the propeller levers must be pulled aft to the minimum rpm position (not feathered) for this N_1 speed reduction to occur. With the propeller levers at the minimum rpm position, aft movement of the power levers in the ground fine range will cause the airbleed reset lever on the primary governor to bleed compressor discharge pressure (P_y bleed) at the fuel control unit thus lowering gas generator (N_1) speed. Propeller speed remains above 1000 rpm although N_1 speed is decreasing since propeller blade angle is decreasing as the power levers are moved aft in the ground fine range. Pulling the power lever up and aft across the ground fine gate into the reverse range will decrease the blade angle from 3° at the ground fine gate to -9° and the reverse stop. This will occur with the propeller levers at the minimum rpm or at maximum rpm position. However, at the minimum propeller rpm position, N_1 speed will not increase as the power levers are moved aft into the reverse range since compressor discharge air is being bled at the fuel control unit. With the propeller levers set at high rpm, movement of the power levers in the ground fine range will allow a minimum of 71 to 73% N_1 speed (HIGH IDLE). Power lever movement aft across the ground fine gate to the maximum reverse stop will increase N_1 speed from 71–73% to 86–88% with propeller blade angle decreasing to -9° as discussed above.

24. The propeller system incorporates a synchrophaser which matches left and right propeller rpm as well as propeller phase relationship. The maximum synchrophaser

range is approximately 20 rpm. The synchrophaser will not synchrophase the propellers unless they are set by the constant speed governor (propeller levers) within 20 rpm of each other. The synchrophaser will increase one engines propeller speed to match the other but will not decrease the speed set by the constant speed governor.

FUEL SYSTEM

25. The fuel supply system consists of a separate main fuel system and auxiliary fuel system for each side of the aircraft. The aircraft total fuel system capacity is 548 gallons with 542 gallons usable. The main fuel system provided for each engine consists of four interconnected bladder wing tanks and one integral (wet wing) cell fuel tank feeding into the nacelle bladder tank. The engine receives fuel from the nacelle tank. When the auxiliary tanks are filled, they are automatically used first. Fuel is transferred automatically from the auxiliary tank, into the nacelle tank by a transfer jet pump located in the auxiliary tank, keeping the nacelle tank full until the auxiliary tank is empty. The main fuel system wing tanks will automatically gravity feed into the nacelle tank maintaining fuel at approximately the same level. An engine-driven fuel boost pump located on the aft side of the engine accessory section provides positive fuel pressure to the engine-driven high pressure fuel pump when the engine is operating. An electric standby pump is located in each nacelle tank and is used in the event of engine-driven boost pump failure or for crossfeed of fuel in a single-engine situation. Fuel quantity is monitored by two fuel gages located on the overhead cockpit control panel. Each gage indicates its respective side main fuel system quantity. A FUEL QUANTITY switch, spring loaded to MAIN can be held in the AUXILIARY position to indicate fuel quantity in the auxiliary tanks. The fuel gaging system utilizes special shielded wiring similar to that used in the RC-12D for EMI purposes. The left and right main and auxiliary fuel systems are individually serviced by "over the wing" gravity fill.

ENVIRONMENTAL

26. Cabin pressurization, heating, cooling and ventilation are accomplished by mixing engine bleed air and ambient air and are controlled by electronic flow control units forward of the firewall in each engine nacelle. Cockpit cooling has been increased with added floor outlets and larger overhead eyeball air outlets as compared to the RC-12H. The cabin ducting is routed to exhaust on the mission equipment including the data link. These outlets distribute air from zero to a maximum of 550 $\pm$ 250 cubic ft per minute. The cabin air recirculating rate is in excess of one time per minute. Cold air conditioning utilizes a freon system having a rated capacity of 32,000 BTU/hour. The compressor is belt-driven by the right engine. Bleed air is extracted from both engines for heating. Engine bleed air provides a nominal heating capacity of 45,000 BTU/hour. The heating system is designed to maintain a cockpit/cabin temperature of +65°F with outside air temperatures down to -45°F while in flight or on the ground.

27. An oxygen system is provided primarily as an emergency system, and consists of two 70 cubic foot capacity oxygen supply cylinders located in the unpressurized portion of the aircraft behind the aft pressure bulkhead. The pilot and copilot positions are equipped with diluter demand type regulators and a first aid oxygen mask is provided in the cabin.

28. A side facing chemical toilet is installed in the aft cabin area. This non-flush system uses a dry chemical preparation to deodorize the stored waste. One relief tube is provided, located immediately aft of the cargo door on the left side of the fuselage.

ICE PROTECTION SYSTEM

29. The windshield panel in front of each pilot is electrically anti-iced and defogged by air from the cabin heating system. Aircraft surface deicing for the inboard and outboard wing leading edge, horizontal stabilizer, stabilons and taillets is performed by pneumatic deice boots. Aircraft surface deice boots are inflated through a single cycle manual or automatic mode. The manual mode inflates all aircraft surface deice boots simultaneously and keeps them inflated as long as the switch is held in the manual position. The automatic mode initiates a single cycle inflating the wing boots approximately 6 seconds, deflating the wing boots and inflating the horizontal stabilizer, stabilon and taillet boots for 4 seconds. A separate selector switch is used to pneumatically deice certain mission antennas through manual boot inflation or a single cycle timed inflation period. Data link antenna anti-ice is provided for the forward wideband data link radome and wheel brakes through the use of engine bleed air. Automatically cycled electrothermal anti-icing boots are installed on the propeller blades. Ice protection for the engines are provided by inertial separation and air inlet leading edge lip heating by engine exhaust bleed (NOTE: If IR suppressed exhaust stacks were installed, the engine air inlet anti-ice system would have provisions to deselect the engine air inlet anti-ice system.

ELECTRICAL

30. The RC-12K uses both direct current (DC) and alternating current (AC) electrical power. The primary DC power source consists of two engine-driven 28 volt, 400 ampere starter-generators. A 24 volt, 34 ampere/hour nickel-cadmium battery furnishes DC power when the engines are not operating. The output of each generator passes to a respective generator bus, then power is distributed to DC buses. When a generator is not operating, reverse current and over-voltage protection is automatically provided. Two inverters (750 volt-amperes, 115 volts and 26 volts, 400 hertz (Hz)) operating from DC power produce the aircraft required single phase AC power. The three phase mission AC (3000 volt-ampere 400 Hz) electrical power for inertial navigation and mission avionics is supplied by two DC powered inverters. Both generators' voltage output and percent load as well as the aircraft's single phase inverters' voltage output and frequency are independently, digitally displayed on the

overhead cockpit control panel. Battery voltage and amperage are digitally displayed on the independent meters located on the mission control panel right of the copilot's position on the cockpit wall. Both 3-phase mission inverters' voltage, frequency and percent load are also digitally displayed on the mission control panel.

AUTOMATIC FLIGHT CONTROL SYSTEM

31. The automatic flight control system (AFCS) is a completely integrated autopilot/flight director/air data system which has a full complement of horizontal and vertical flight guidance modes. These include all radio guidance modes and air data oriented vertical modes. The AFCS is a Sperry SPZ-4000 Digital Automatic Flight Control System. The autopilot may be coupled through the NAV mode of the flight director to a VOR or TACAN station or to the aircraft's Inertial Navigation System. Autopilot aileron servo torque has been tailored to accommodate the high roll inertia created by the heavy DF/ELINT antenna pods mounted at the wing tips. Horizontal modes include: heading hold (HDG); navigation tracking of VOR, localizer, TACAN, and Inertial Navigation System (INS) courses (NAV); approach tracking of VOR and localizer courses (APR); and approach tracking of backcourse localizers (BC). Vertical modes include altitude hold (ALT); altitude capture with automatic switch to altitude hold (ALT SEL); vertical velocity hold (VS); and indicated airspeed hold (IAS). In the APR mode, the AFCS will automatically capture and track the glide slope and localizer beam. When the autopilot is coupled to the flight director commands, the instruments act as a means to monitor the performance of the autopilot. When the autopilot is not engaged, the same modes of operation are available for the flight director only. The pilot maneuvers the aircraft to satisfy the flight director commands. One additional mode, which is available for uncoupled flight director commands only, is the go-around (GA) mode. When the GA mode is selected, by pressing a button on the left power lever, the autopilot will disengage and the flight director command cue will command a wings level, 7° pitch-up attitude.

32. A yaw damper automatically engages whenever the autopilot is engaged. When the autopilot is not engaged, the yaw damper may be utilized separately at altitudes below 17,000. The operator's manual requires use of the yaw damper at 17,000 ft and above. The yaw rate signal used for yaw damping is derived from the directional gyro. Yaw damping decreases as roll rate and bank attitudes are increased and is locked out if the aircraft roll rate exceeds 7 deg/sec or the bank attitude exceeds 45°, but the rudder pedal position will be held fixed by the rudder servo.

INTERIOR ARRANGEMENT

33. The interior arrangement consists of the crew compartment and the mission equipment area. The crew compartment is separated from the mission equipment area by a curtain which may be opened or closed. The total interior space available for mission equipment is 299 cubic ft. Provisions for the stowage of two chest parachutes is incorporated near the emergency exit door.

MISSION ANTENNAS

34. Mission antennas are provided as depicted in figures B-6 through B-9. A detailed description of mission equipment and operation is contained in the operator's manual (ref 2).

APPENDIX C. INSTRUMENTATION

1. Flight test data were recorded on magnetic tape using pulse code modulation and by hand from cockpit instruments located in the pilot's panel. Aileron, elevator, and rudder positions were measured using linear variable differential transducers. Control forces were measured using a strain gaged control yoke and pedals. A test boom pitot-static system was installed on the nose radome to measure airspeed. The position error of the boom airspeed system is presented in figure C-1.

2. Instrumented and related special equipment installed are presented below. Figures C-2 and C-3 show the cockpit instrument panel, instrumented control yokes, instrumented pedals, cabin instrumentation, and ballast locations.

Pilot/Copilot Panel

Center of gravity normal acceleration (g)
Airspeed (ship)
Airspeed (boom)
Outside air temperature
Pressure altitude (boom)
Torque (ship)
Propeller speed (left)
Propeller speed (right)
Gas generator speed (left)
Gas generator speed (right)
Turbine gas temperature (left)
Turbine gas temperature (right)
Engine torque (left)
Engine torque (right)

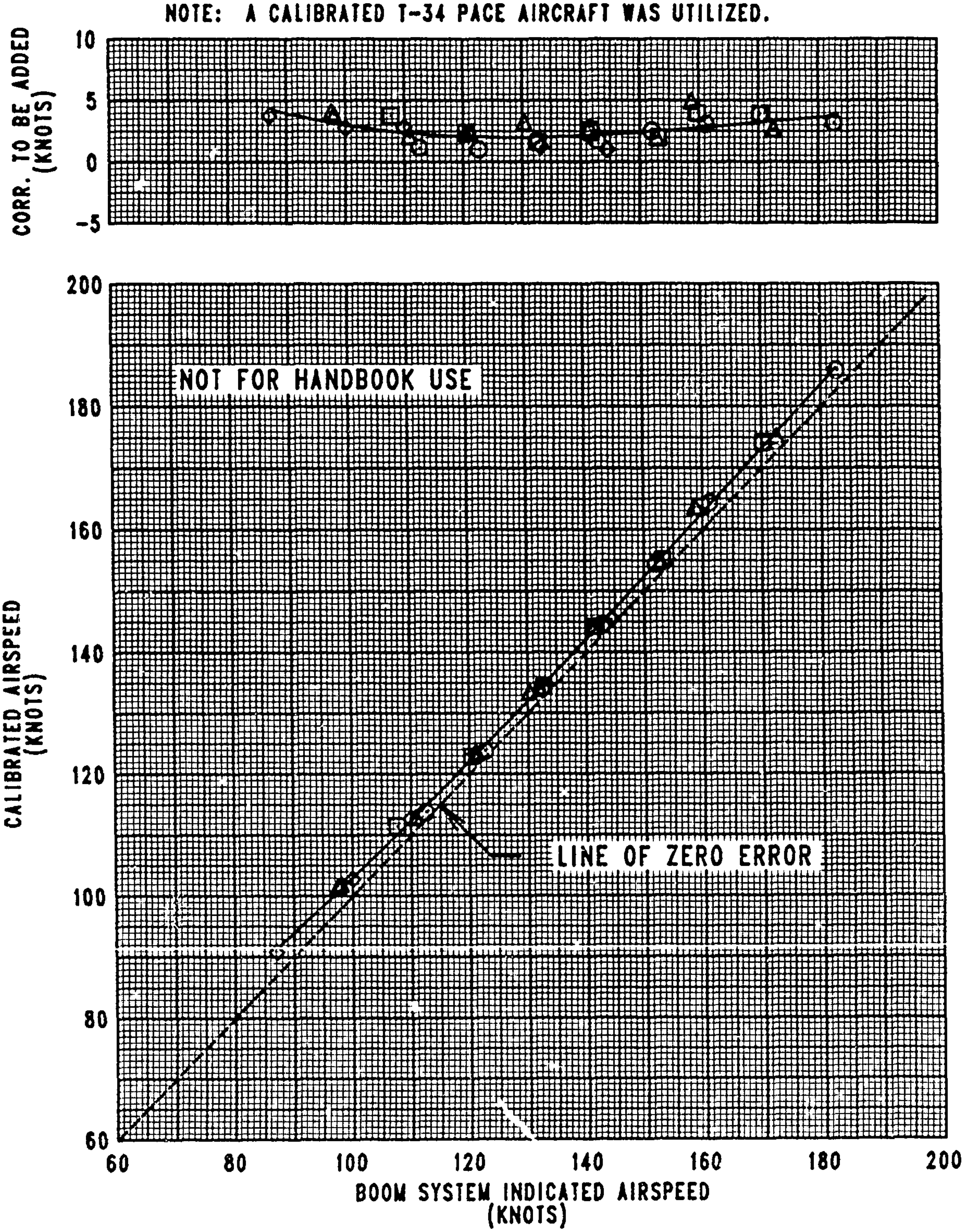
PCM Parameters (recorded by onboard tape)

Airspeed (boom)
Pressure altitude (boom)
Angle of attack
Angle of sideslip
Airspeed (ship, pilot)
Pressure altitude (ship, pilot)
Radar altimeter
Elevator position (left)
Elevator position (right)
Elevator trim tab position
Elevator force (column)
Longitudinal control position
Rudder position
Rudder trim tab position
Rudder pedal force (left)
Rudder pedal force (right)
Aileron position (left)

FIGURE C-1
BOOM SYSTEM AIRSPEED CALIBRATION
RC-12K USA S/N 85-0149

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	GEAR POSN	FLAP POSN (%)
⊙	16030	187.6(FWD)	6630	7.0	1493	UP	0
⊡	14860	187.1(FWD)	8570	3.0	1496	DOWN	0
△	15530	187.1(FWD)	8760	4.5	1498	DOWN	40
◇	15170	187.1(FWD)	8760	4.0	1495	DOWN	100

NOTE: A CALIBRATED T-34 PACE AIRCRAFT WAS UTILIZED.



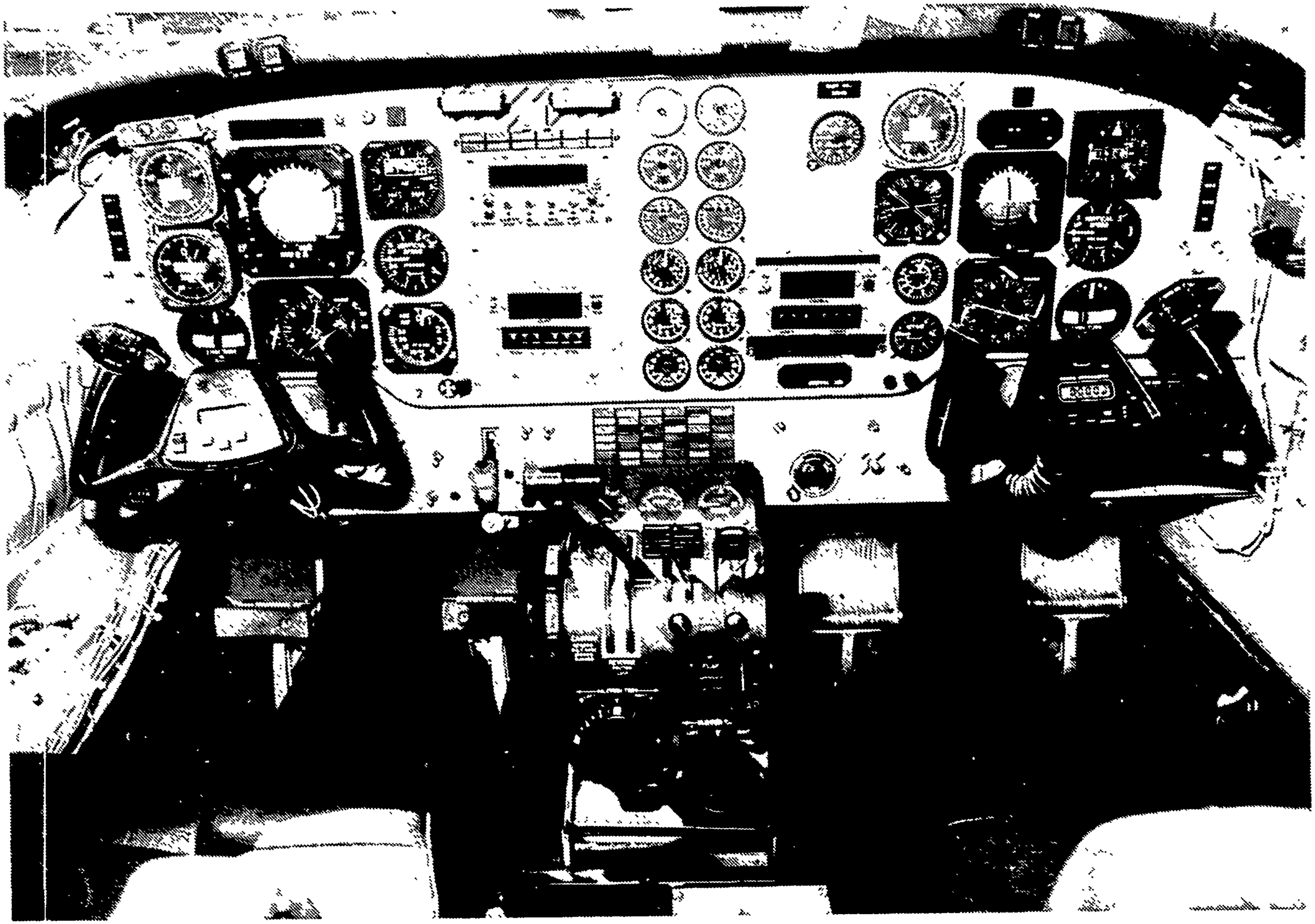


Figure C-2. Cockpit Instrumentation

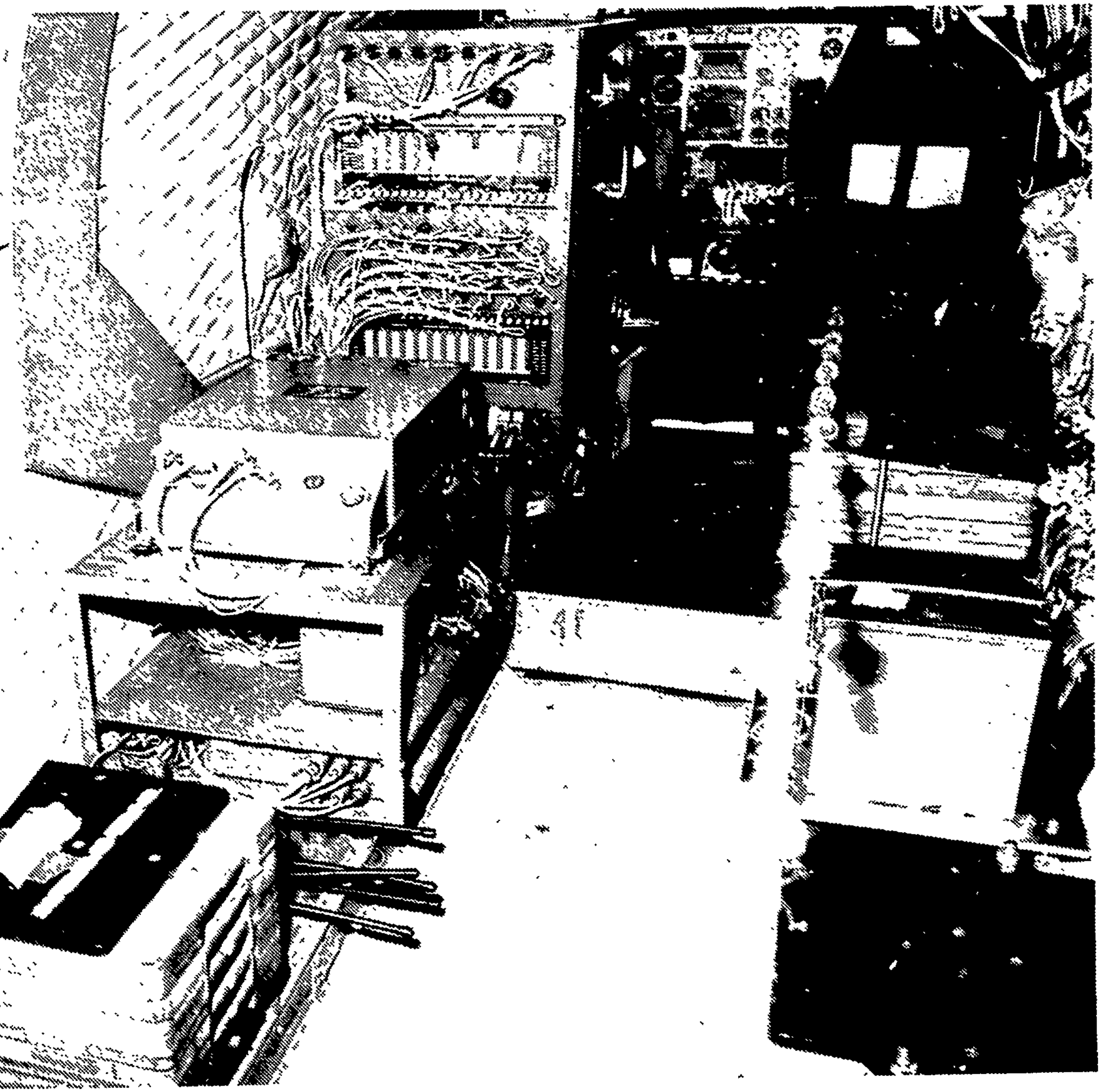


Figure C-3. Aircraft Instrumentation and Ballast

Aileron position (right)
Aileron trim tab position
Aileron force (wheel)
Lateral control position
CG normal acceleration
Pitch attitude
Roll attitude
Pitch rate
Roll rate
Yaw rate
Engine torque (left)
Engine torque (right)
Gas generator speed (left)
Gas generator speed (right)
Propeller speed (left)
Propeller speed (right)
Outside air temperature
Fuel temperature (left)
Fuel temperature (right)
Fuel used (left)
Fuel used (right)
Fuel flow (left)
Fuel flow (right)
Measured gas temperature (left)
Measured gas temperature (right)
Run-stop locator
R-CAL
Event
Run number
Stall warning
Squat switch

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. This appendix contains some of the data reduction techniques and analysis methods used to evaluate the RC-12K aircraft. Topics discussed include glide, level flight performance, field performance, and aircraft weight and balance.

GLIDE

2. The propeller stopped glide method was used to define the drag of the RC-12K aircraft in the cruise configuration. The method involved obtaining flight data while the aircraft was stabilized in a constant airspeed descent with both engines shutdown and propellers feathered and stopped. Parameters measured include airspeed, pressure altitude, outside air temperature, gross weight, and elapsed time. The airspeed range from 110 to 170 knots indicated airspeed with the propeller stopped was investigated for a target pressure altitude (H_P) band of 15,500 to 13,500 feet. The technique used to develop the baseline drag equation is shown below.

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

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

$$D \; x \; V_T = T \; x \; V_T + W \; x \; V_T \; x \; \sin\gamma \tag{3}$$

$$-V_T \; x \; \sin\gamma = \frac{dh}{dt} = \frac{T \; x \; V_T - D \; x \; V_T}{W} \tag{4}$$

Where:

L = Lift force (lb)
W = Aircraft gross weight (lb)
$\gamma =$ Descent angle (deg) $= \sin^{-1} \dfrac{\frac{dh}{dt}}{V_T}$
T = Net thrust (lb) = zero with propeller stopped
D = Drag force (lb)
$V_T =$ Aircraft true aispeed on flight path (ft/sec) $= a\left\{5\left[\left(\frac{q_c}{P_a}+1\right)^{2/7}-1\right]\right\}^{0.5}$

a = Ambient speed of sound (ft/sec) = $a_o \; x \; \theta^{0.5}$
$a_o = 1116.45$ (ft/sec)
$\theta =$ Temperature ratio $= \dfrac{OAT + 459.67}{T_o\left\{1 + Tk\left[\left(\frac{q_c}{P_a}+1\right)^{2/7}-1\right]\right\}}$

OAT = Outside ambient temperature (deg C)

T_o = 518.67 deg R
TK = Temperature probe recovery factor

q_c = Dynamic pressure (lb/sq ft) = $P_o\left\{\left[0.2\left(\frac{1.68781\ xV_c}{a_0}\right)^2 + 1\right]^{3.5} - 1\right\}$

1.68781 = Conversion factor (ft/sec-kt)
P_o = 2116.22 (lb/sq ft)
V_c = Calibrated airspeed (kt) = $V_{ic} + V_{pc}$

V_{ic} = Instrument corrected airspeed (kt)

V_{pc} = Static source position error obtained using boom airspeed calibration presented in figure C-1 (ft)

P_a = Ambient air pressure (lb/sq ft) =

$$P_o(1 - 6.875586E-06\ x\ Hp_{ic})^{5.255876} - DPPB$$

Hp_{ic} = Instrument corrected pressure altitude (ft)

DPPB = $q_c - q_{cic}$

q_{cic} = Instrument corrected dynamic pressure (lb/sq ft) =

$$P_o\left\{\left[0.2\left(\frac{1.68781\ x\ V_{ic}}{a_o}\right)^2 + 1\right]^{3.5} - 1\right\}$$

$\frac{dh}{dt}$ = Tapeline rate of descent (ft/sec) = $\frac{dH_p}{dt} x \frac{Ta_t}{Ta_s}$

$\frac{dHp}{dt}$ = Measured slope of pressure altitude versus time (ft/sec)

Ta_t = Test day ambient temperature (deg k)

Ta_s = Standard day ambient temperature (deg k)

Considering the drag and lift force equations and applying poweroff glide conditions, the following nondimensional relationships can be developed:

$$C_D = \frac{D}{q\ x\ s} \tag{5}$$

$$C_D = \frac{W\ x\ \sin\gamma}{q\ x\ s} \tag{6}$$

$$C_L = \frac{L}{q \ x \ s} \tag{7}$$

$$C_L = \frac{W \ x \ \cos\gamma}{q \ x \ s} \tag{8}$$

ϱ = Air density (slug/cu ft)

The base-line drag equation (CDBL) was then developed by plotting C_D versus $C_L{}^2$ and fitting a first order equation to the test points resulting in the following equation:

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

Where:

CDBL = Base-line drag coefficient

C_{D_o} = Minimum parasite drag coefficient = intercept of C_D axis

$\frac{\Delta C_D}{\Delta C_L{}^2}$ = Curve slope from plot of C_D versus C_L for glide test

LEVEL FLIGHT PERFORMANCE

3. In dual engine powered flight, the total drag of the aircraft is defined as:

$$CDTC' = C_D + \frac{\Delta C_D}{\Delta C_L^2} \ x \ C_L^2 + B \ x \ TC' + C \tag{10}$$

Where:

CDTC' = Total drag coeffcient of the aircraft in powered flight
B, C = Coefficient constants

TC' = Coefficient of thrust = $\frac{T}{q \ s}$

T = Thrust (lb) = $\frac{550 \ x \ THP}{V_T}$

THP = Thrust horsepower (hp) = $\eta_p \ x \ shp + \frac{F_n \ x \ V_T}{550}$

SHP = Shaft horsepower (shp) = $\frac{QE \ x \ Np \ x \ 2 \ x \ \pi}{33,000}$

η_p = Propeller efficiency
Fn = Exhaust thrust (lb)

QE = Engine torque (ft-lb)
Np = Propeller speed (rpm)

The coefficient constants (B, C) were found by first subtracting equation 9 from 10 and defining the differences as the increase drag due to thrust effect.

$$\Delta CD_{TC'\,-\,BL} = CDTC' - CDBL \tag{11}$$

CDBL was calculated from the poweroff glide drag plar for each powered flight test point and CDTC' was calculated from the powered flight thrust horsepower. A plot of $\Delta CD_{TC'\,-\,BL}$ versus TC' was made and a first order equation was fitted through the test points as shown in figure D-1. The value for B and C is the slope and intercept of the faired line, respectively.

4. Thrust horsepower required for level flight was calculated for average test day conditions using the following equation:

$$THP = \frac{V_{tcor}\ (C_{D_O}\ x\ q_{avg}\ x\ S + \frac{\Delta C_D}{\Delta C_L{}^2}\ x\ \frac{W_{avg}}{q_{avg}\ x\ S} + C\ x\ q_{avg}\ x\ S)}{550(1-B)} \tag{12}$$

Where:

$C_{D_O}, \frac{\Delta C_D}{\Delta C_L{}^2}$, C, B = Drag polar coefficients contained in table 5 in the Results and Discussion section of this report.

Each level flight test point was corrected to average test day conditions using the following equations:

$$V_{T_{cor}} = \left(\frac{2\ x\ W_{avg}}{\varrho_{avg}\ x\ C_L\ x\ S}\right)^{0.5} \tag{13}$$

$$THP_{cor} = \frac{550\ x\ C_D\ x\ \varrho_{avg}\ x\ V_{T_{cor}}{}^3\ x\ S}{2} \tag{14}$$

Where:

$V_{T_{cor}}$ = True airspeed corrected for average test day conditions (ft/sec)
W_{avg} = Average test day gross weight (lb)
q_{avg} = Average test day dynamic pressure (lb/sq ft) = $1/2\ x\ \varrho_o\ x\ V_{T_{cor}}{}^2$
ϱ_{avg} = Average test day ambient air density (slugs/cu/ft) = $\varrho_o(1-6.8755856E-06\ x\ Hd_{avg})^{4.255876}$
ϱ_o = 0.002376892 (slugs/cu ft)
Hd_{avg} = Average test day density altitude (ft)

Specific range (SR) data were derived from the test level flight power required and fuel flow rate. Level flight performance shaft horsepower and fuel flow rate data for each engine were referred as follows:

$$SHP_{REF} = \frac{SHP_t}{\delta \ x \ \theta^{0.5}} \tag{15}$$

$$W_{F_{REF}} = \frac{W_{F_t}}{\delta \ x \ \theta^{0.5}} \tag{16}$$

SHP_{REF} = Referred shaft horsepower (shp)

SHP_t = Measured shaft horsepower (shp)

$W_{F_{REF}}$ = Referred fuel flow (lb/hr)

W_{F_t} = Measured fuel flow (lb/hr)

δ = Pressure ratio

A fairing was applied to the referred data of each engine as shown in figure D-2. W_{F_t} for each test point was corrected to average test day conditions using the following equation:

$$W_{F_{cor}} = W_{F_t} + \Delta W_F \tag{17}$$

Where:

$W_{F_{cor}}$ = Fuel flow corrected to average test day conditions (lb/hr)

ΔW_F = Change in fuel flow between SHP_t and SHP_{cor} (lb/hr)

SR was then calcuated by:

$$SR = \frac{V_{T_{cor}}}{W_{F_{cor}}} \tag{18}$$

In addition to the referred parameters of SHP and WF, the gas generator speed (N1) and turbine gas temperature (MGT) were referred as follows:

$$N1_{REF} = \frac{N1}{\theta^{0.5}} \tag{19}$$

$$MGT_{REF} = \frac{MGT + 273.15}{\theta} - 273.15 \tag{20}$$

Where:

$N1_{REF}$ = Referred gas generator speed (% rpm)

MGT_{REF} = Referred turbine gas temperature (deg C)

TAKEOFF, LANDING AND ACCELERATE-STOP

5. Prior to starting takeoff, landing and accelerate-stop performance, the runway was measured and marked off in 100 foot increments from the 1000 foot touchdown markers.

6. Takeoff roll distance was obtained by starting at a known point and noting and measuring the liftoff points with ground observers. Tower reported wind speed and direction, and a hand-held anemometer were used to enter the performance chart to determine predicted ground roll.

7. Landing performance was evaluated similar to takeoff performance. The copilot called "mark" at the 50 foot point on the radar altimeter and ground observer marked a spot on the ground to be used to determine the horizontal distance back of the touchdown point. The touchdown and stop point were also noted and measured. These distances were compared with the operator's manual predicted distances.

8. Accelerate-stop (rejected takeoff) distances were measured by starting at a known point with ground observers noting the stop point and measuring the total distance. The test result distances were compared with operator's manual predicted distances.

9. Accelerate-go (single-engine continued takeoffs) were measured to 50 feet above ground level (AGL) by starting at a known point and by having a ground observer note the runway point where the copilot called "mark" at 50 feet on the radar altimeter.

AIRCRAFT WEIGHT AND BALANCE

10. Prior to the start of the preliminary airworthiness evaluation program, a weight and balance determination was conducted on the aircraft using calibrated scales. The aircraft was weighed in the following configurations:

a. Full oil, trapped fuel, no crew, and instrumentation.

b. Full oil, full fuel, no crew, and instrumentation.

c. Full oil, full fuel, no crew, instrumentation, and ballast loading for a mission takeoff gross weight/forward center of gravity (cg) condition.

d. Full oil, full fuel, no crew, instrumentation, and ballast loading for a mission takeoff gross weight/aft cg condition.

The aircraft basic weight and cg with full oil, trapped fuel, no crew, and instrumentation installed was 10,358 lb at fuselage station 187.69. The standard cockpit fuel quantity gage was calibrated by inputting known quantity of fuel from a calibrated fuel truck. The following results were attained: left main, 1274 lb (193.0 gals at 6.6 lb/gal); right main 1261 lb (191.0 gals at 6.6 lb/gal); left aux, 514 lb (77.9 gals at 6.6 lb/gal; right aux, 503 lb (76.2 gals at 6.6 lb/gal). The fuel weight for each test flight was determined prior to engine start and after engine shutdown by using the calibrated standard cockpit fuel quantity gage. The installed test instrumentation fuel totalizers and fuel temperature sensors were used to determine fuel used during a test point.

RIGGING CHECK

11. Mechanical rigging of engine and flight controls was checked for compliance with applicable Beech Aircraft Corporation documents. Control surface travels are presented in table D-1.

DEFINITIONS

12. Results were categorized as deficiencies or shortcomings in accordance with the following definitions.

Deficiency

13. A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued, or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

Shortcoming

14. An imperfection or malfunction discovered during the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the usability of the material or end product.

Table D-1. Control Surface Travels

MODEL RC-12K A200CT | SHIP SERIAL No. FE3 S/N: 85-0149 | AT 77 to 85°F Converted to 59°F

	Ships Control Travels	Print Control Travel and Tolerance	Ships Cable Tension	Print Control Travel and Tolerance
Wing Flap L/H	Up 1/2° App 13° DN 36°	UP 0° to 1° App 14° ±1° DN 35° $\pm^{1^\circ}_{2^\circ}$		
Wing Flap R/H	Up 1/2° App 13° DN 36°	UP 0° to 1° App 14° ±1° DN 35° $\pm^{1^\circ}_{2^\circ}$		
Aileron L/H	Up 23° down 17°	Up 24° $\pm^{2^\circ}_{1^\circ}$ down 16° $\pm^{2^\circ}_{1^\circ}$	80 lb	75 ± 5 lb
Aileron R/H	Up 25 1/2° down 16°	Up 24° $\pm^{2^\circ}_{1^\circ}$ down 16° $\pm^{2^\circ}_{1^\circ}$	78 lb	75 ± 5 lb
Aileron Trim Tab	Up 15° down 15°	Up 15° ± 1 1/2° down 15° ± 1 1/2°	10 lb	9 + 3, -2 lb
Aileron Servo Tab L/H	Up 1/2° down 1/2°	Up 1/2° max down 1/2° max		
Aileron Servo Tab R/H	Up N/A down N/A	Up N/A Down N/A		
Elevator L/H	Up 20 1/2° down 15°	Up 20° ± down 14° $^{+10^\circ}_{-0}$ 0	35 lb	36 ± 8 lb
Elevator R/H	Up 20 1/2° down 15°	Up 20° ± down 14° ± $^{0\,+\,10^\circ}$	35 lb	36 ± 8 lb
Elevator Trim Tab L/H	Up 3 3/4° down 16°	Up 3 1/2° $^{0\,+\,1\,1/2^\circ}_{-0}$ down 15° $^{0\,+\,10^\circ}_{-0^\circ}$	15 1/2 lb	14 + 3, -2 lb
Elevator Trim Tab R/H	Up 3 3/4° down 16°	Up 3 1/2° $^{0\,+\,1\,1/2^\circ}_{-0}$ down 15° $^{0\,+\,10^\circ}_{-0^\circ}$	15 1/2 lb	14 + 3, -2 lb
Elevator Servo Tab L/H	Up 1/2° down 1/2°	Up 1/2° max down 1/2° max		
Elevator Servo Tab R/H	Up 1/2° down 1/2°	Up 1/2° max down 1/2° max		
Rudder	Left 25 1/4° Right 25 1/2°	25 $^{0\,+\,1^\circ}_{-0^\circ}$ 25 $^{0\,+\,1^\circ}_{-0^\circ}$	78 lb	70 + 15, -0 lb
Rudder Trim Tab	Left 15 1/2° Right 15°	Left 0 + 1 1/2° Right 0 + 1 1/2° Right 15 -0° Left 15 -0°	10 1/2 lb	9 + 3, -0 lb
Rudder Servo Tab	Left 7 1/4° Right 7 1/4°	up 7 1.2 ± 1/2° down 7 1/2 ± 1/2°		

FIGURE D-1
DUAL ENGINE LEVEL FLIGHT
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL

NOTE: THE ORDINATE INTERCEPT REPRESENTS THE CHANGE IN DRAG COEFFICIENT BETWEEN POWERED AND UNPOWERED FLIGHT AT ZERO LIFT, ZERO THRUST CONDITIONS.

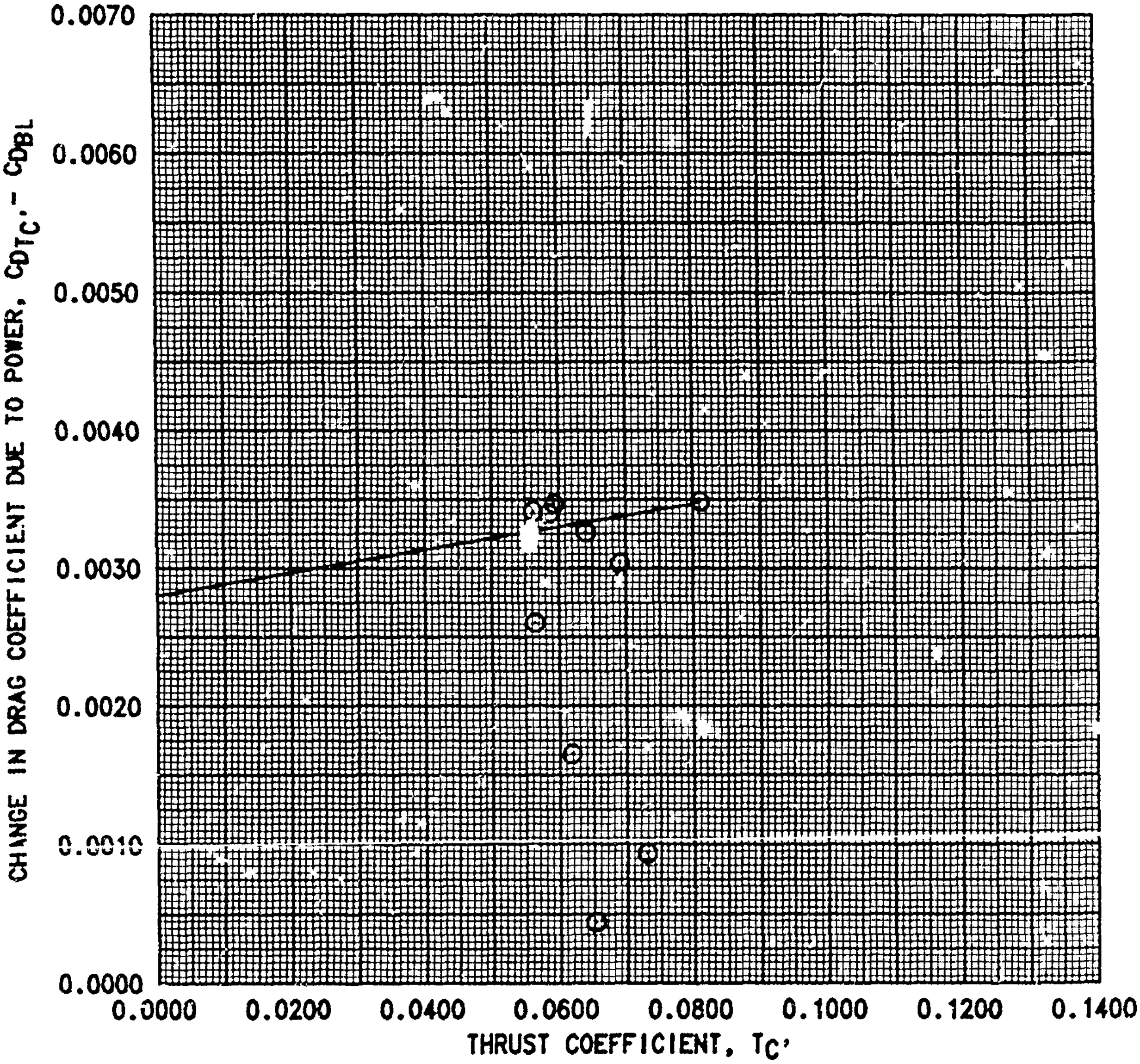
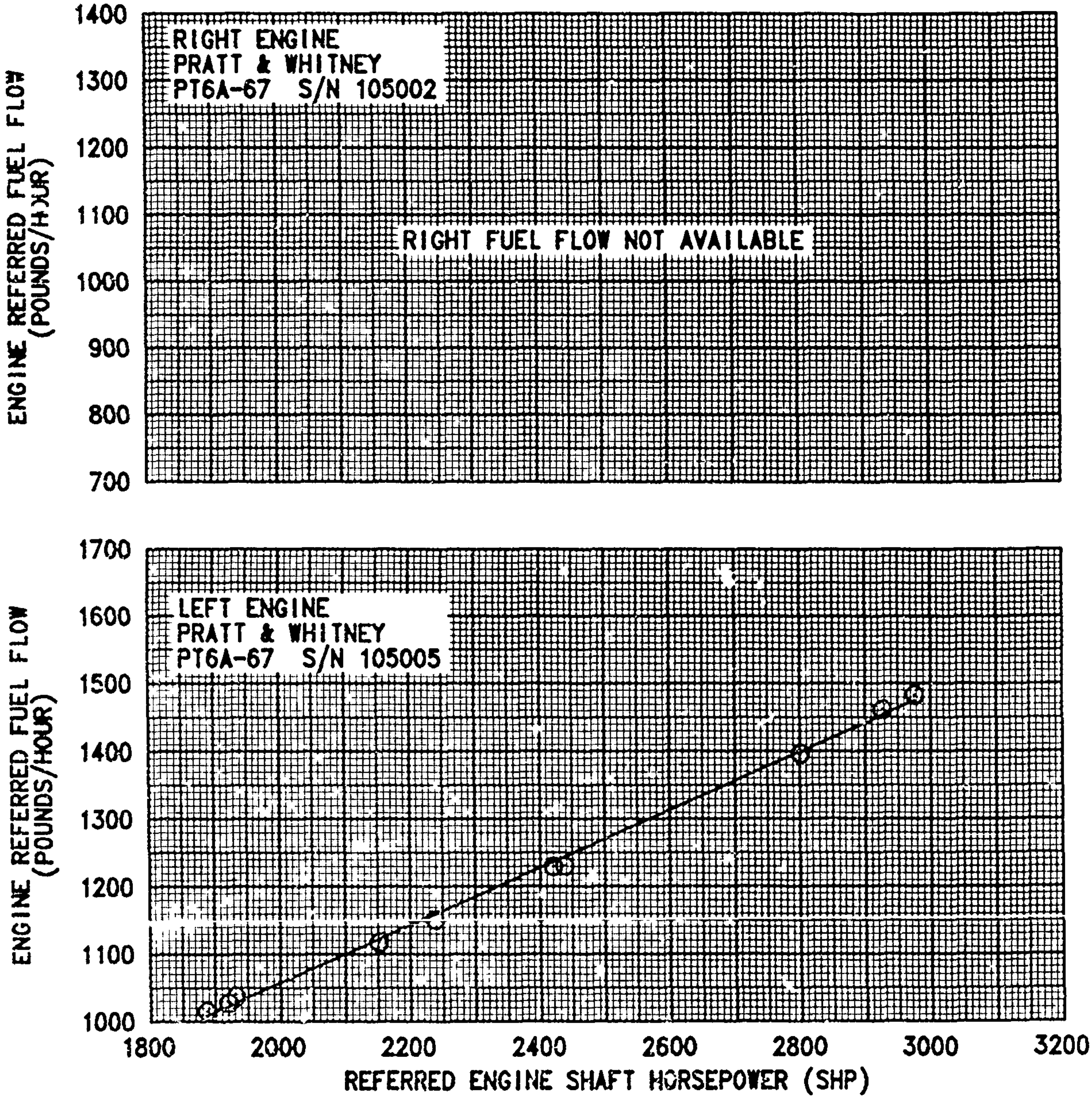


FIGURE D-2
ENGINE CHARACTERISTICS
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL



APPENDIX E. TEST DATA

Table E-1. Takeoff, Landing and Accelerate-Stop Performance[1]

Maneuver	Liftoff (ft)	(Operator's Manual)	50 foot	(Operator's Manual)	Stop	(Operator's Manual)
Takeoff Flaps Up	2970	(3000)	4270	(4300)	N/A	
Takeoff, 40% Flaps	2400	(2700)	3320	(3600)	N/A	
Accelerate-Go Flaps Up	3155	(N/A)	5580	(6400)	N/A	
Accelerate-Go 40% Flaps	2561	(N/A)	4128	(4750)		
Accelerate-Stop Flaps-Up	N/A		N/A		4293	(4800)
Accelerate-Stop 40% Flaps	N/A		N/A		4078	(4500)
Land 100% Flaps	N/A		N/A		2241 (Over 50' obstacle)	(2600)
Land Flaps Up	N/A		N/A		2351 (Over 50' obstacle)	(3600)
Land Single-Engine Flaps 100%	N/A		N/A		2488 (Over 50' obstacle)	(2800)
Land Flaps 100% Maximum Reverse	NA		N/A		1852	(N/A)

NOTE:

[1]Landing and stopping maneuvers were performed using power lever maximum ground fine position except where noted.

Table E-2. Stall Performance

Airplane Configuration	Engine Torque Left/Right (%)	Average Gross Weight (lb)	Artificial Stall Warning (KIAS)	Aerodynamic Buffet (KIAS)	Stall (KIAS)	Remarks
TO, 40% Flaps	10/10	15,580	96.0	90.0	90.0	
TO, 40% Flaps	97/97	15,554	85.0	83.0	83.0	
TO, 0% Flaps	8/8	15,482	106.0	99.0	97.0	
TO, 0% Flaps	98/98	15,513	97.0	95.0	92.0	Unacceleratead (1g) stalls
CR, 0% Flaps	46/46	15,380	102.0	102.0	101.0	
PA, 100% Flaps	6/8	15,148	79.0	74.0	70.0	
L, 100% Flaps	98/96	15,183	69.0	68.0	68.0	
L, 0% Flaps	5/4	15,245	100.0	99.0	97.5	
TO, 40% Flaps	97/95	14,917	129.0	133.0	111.0	1.95g
TO, 0% Flaps	96/96	14,884	137.5	138.0	134.0	1.9g
CR, 0% Flaps	45/45	14,808	155.0	150.0	136.0	2.0g
CR, 0% Flaps	5/6	14,846	166.0	160.0	147.0	1.9g
L, 100% Flaps	4/6	14,774	145.0	140.0	106.0	2.0g
PA, 100% Flaps	35/35	14,593	115.0	115.0	110.0	1.95
TO, 40% Flaps	0/95	14,553	87.0	80.0	77.0	
TO, 0% Flaps	0./95	14,544	97.5	97.0	90.0	Left engine shutdown and prop feathered unaccelerated (1g) stalls
PA, 100% Flaps	0/80	14,535	72.0	71.0	65.0	

NOTE:

[1]All tests performed at FS 187.1 to 187.7 longitudinal cg and an average density altitude of 14,796 ft.

Table E-3. Control System Characteristics

Control	Freeplay (in.)	Breakout Plus Friction	Autopilot/ Electric Trim Rate
Aileron	3/8	3–4 lb left and right	Full right to full left: 5 sec Full left to full right: 5 1/2 sec
Elevator	1/8	N/A	52 sec stop to stop
Rudder	1/8	10 lb left and right	N/A

FIGURE E-1
PROPELLER FEATHERED/STOPPED GLIDE DRAG POLAR
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15269	187.1 (FWD)	15370	-7.5	CRUISE	GLIDE

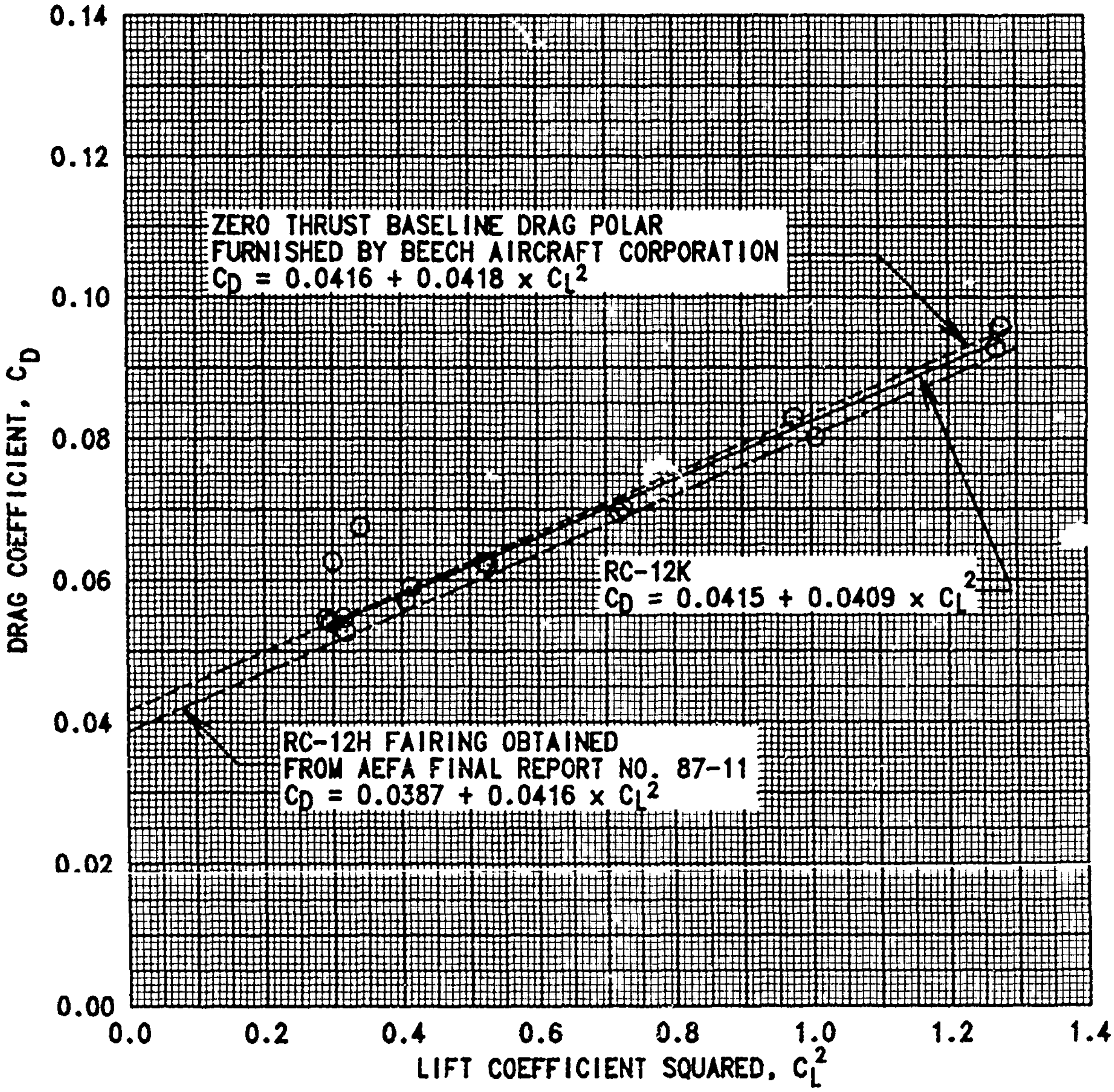


FIGURE E-2
DUAL ENGINE LEVEL FLIGHT DRAG POLAR
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	[illegible] DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL

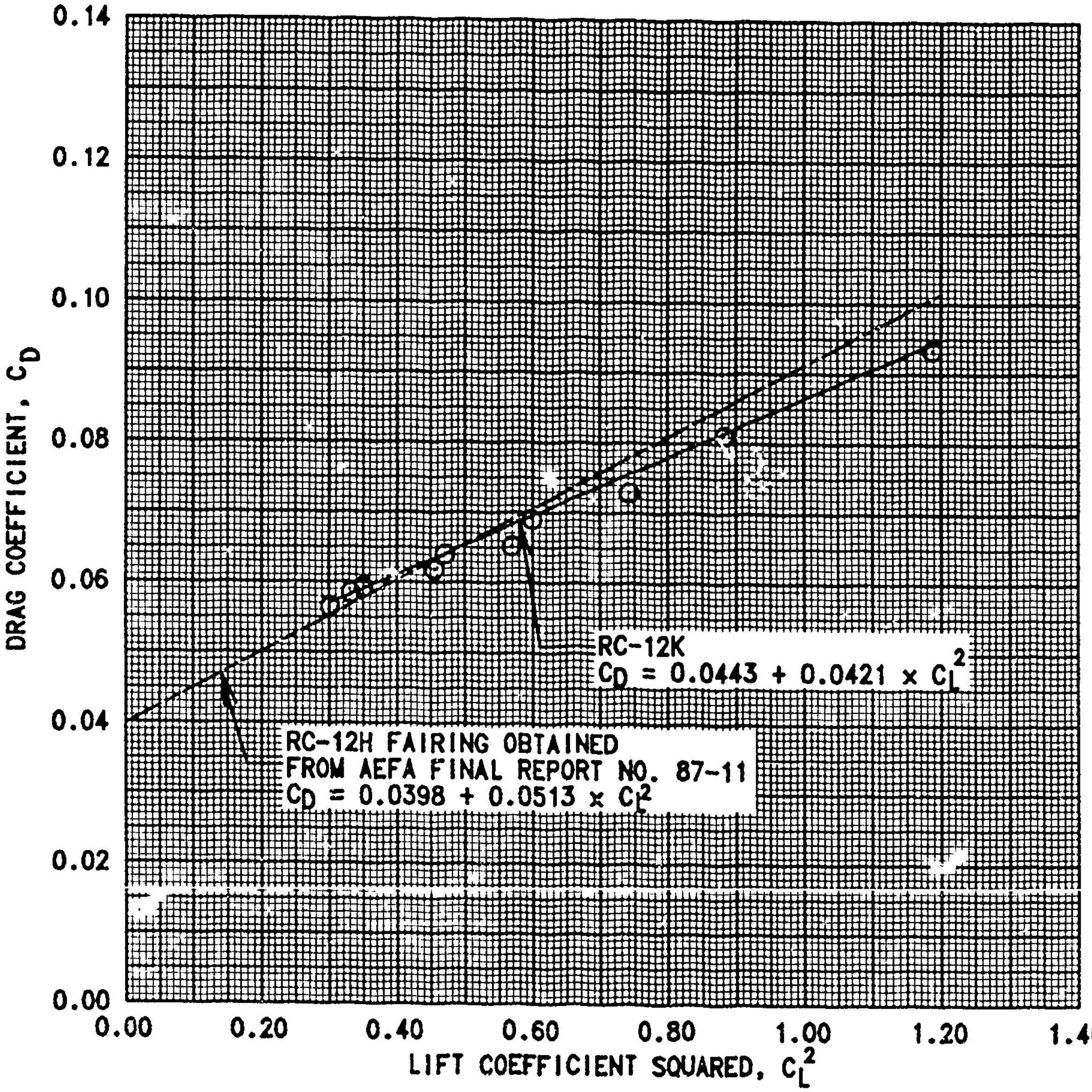


FIGURE E-3
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL

NOTE: SPECIFIC RANGE CALCULATED USING LEFT ENGINE FUEL FLOW ONLY.

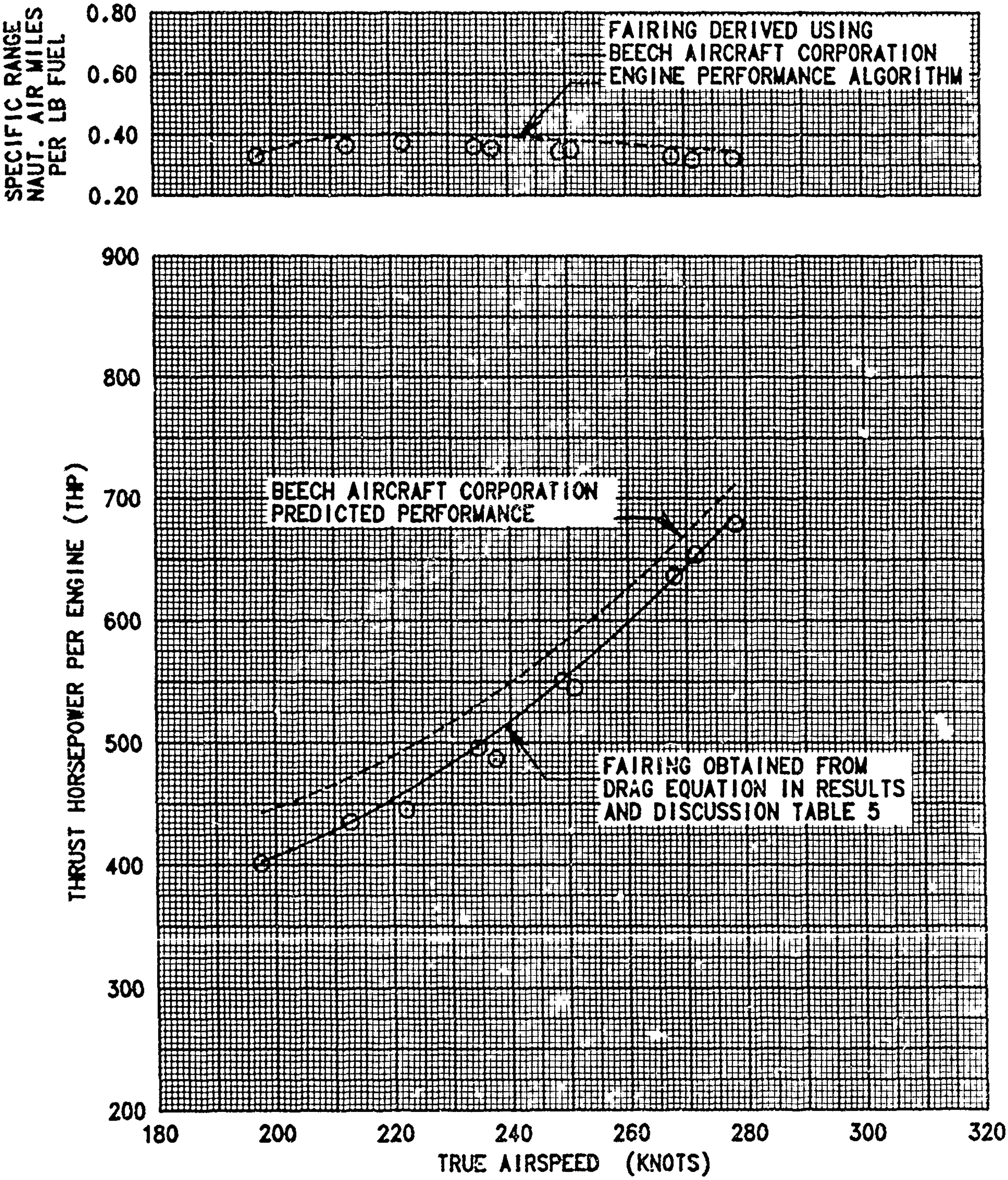


FIGURE E-4
ENGINE CHARACTERISTICS
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL

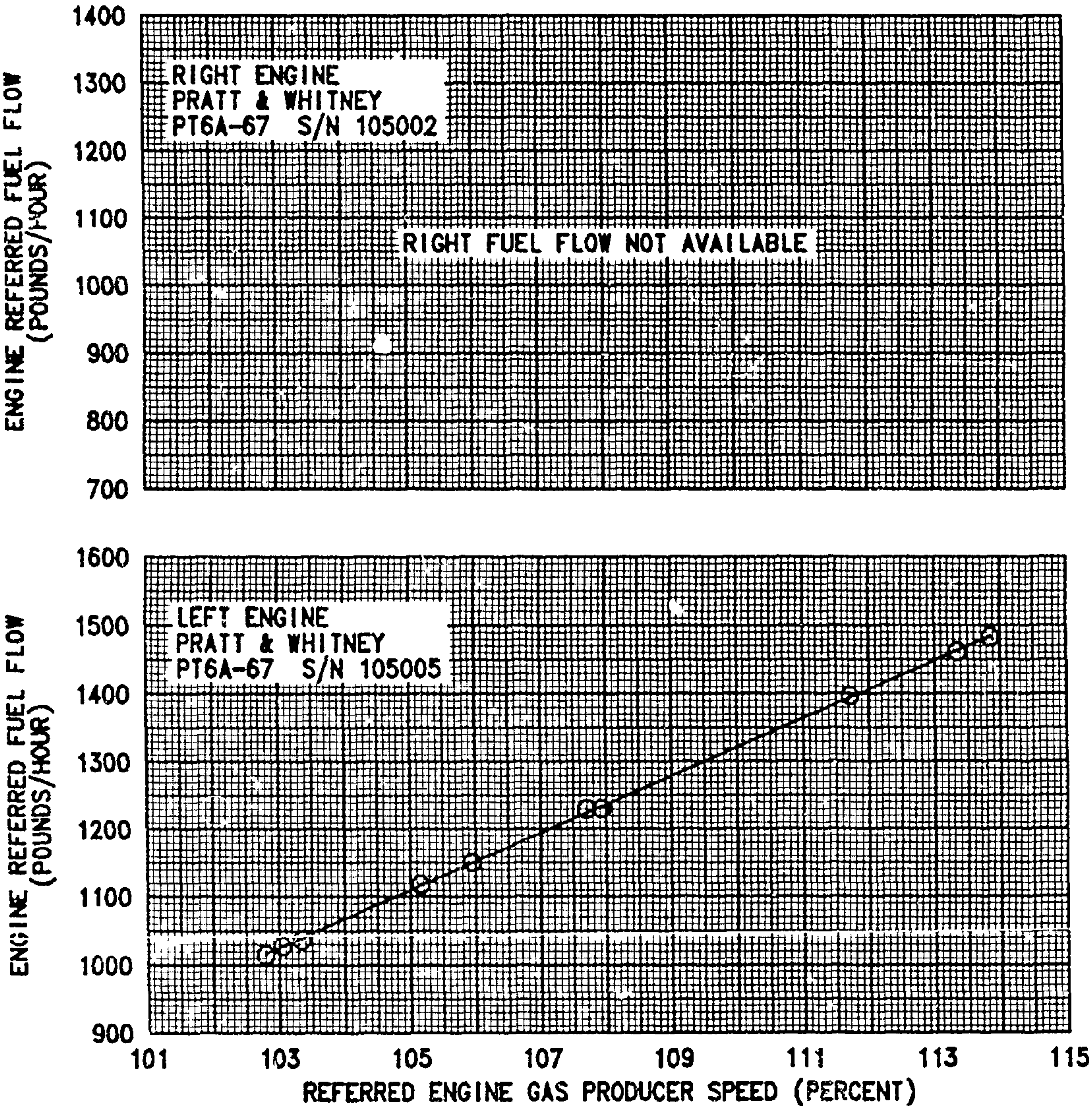
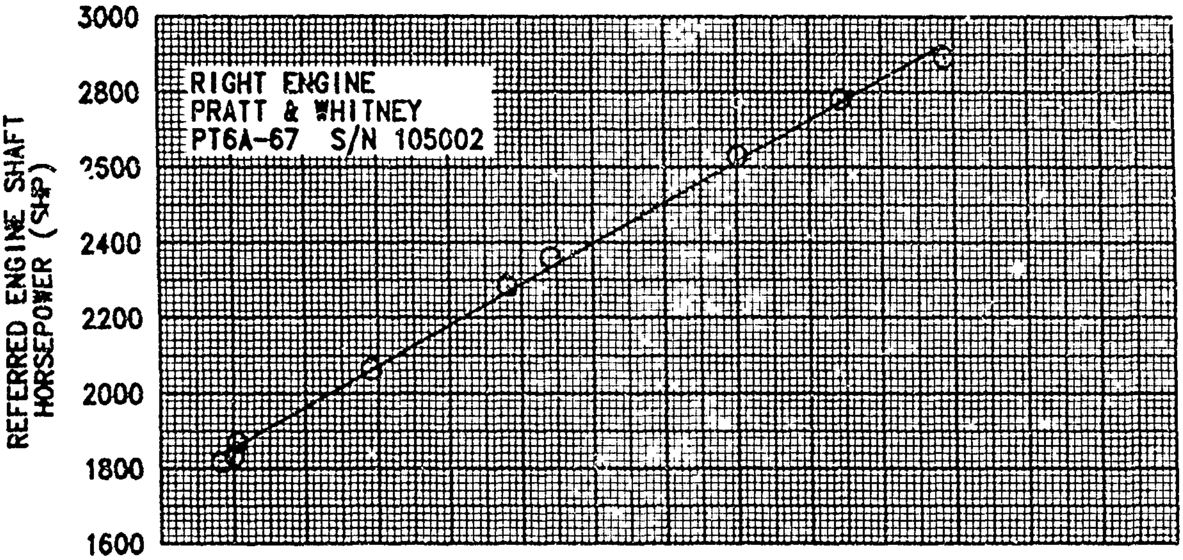


FIGURE E-5
ENGINE CHARACTERISTICS
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	31510	-44.5	1499	CRUISE	LEVEL



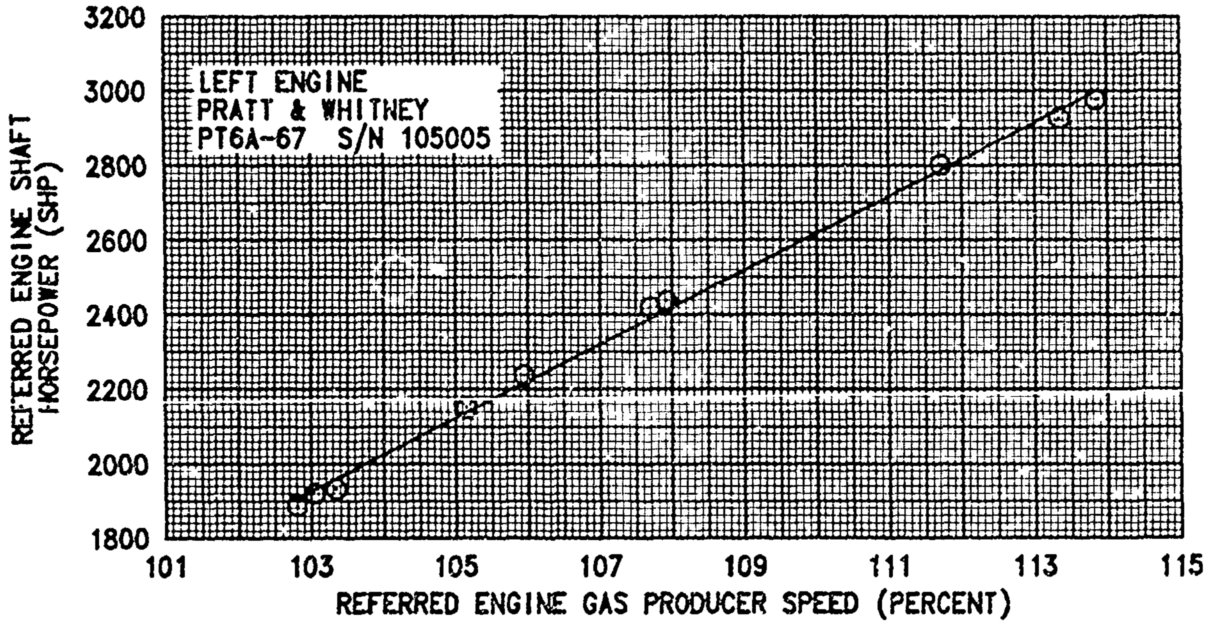
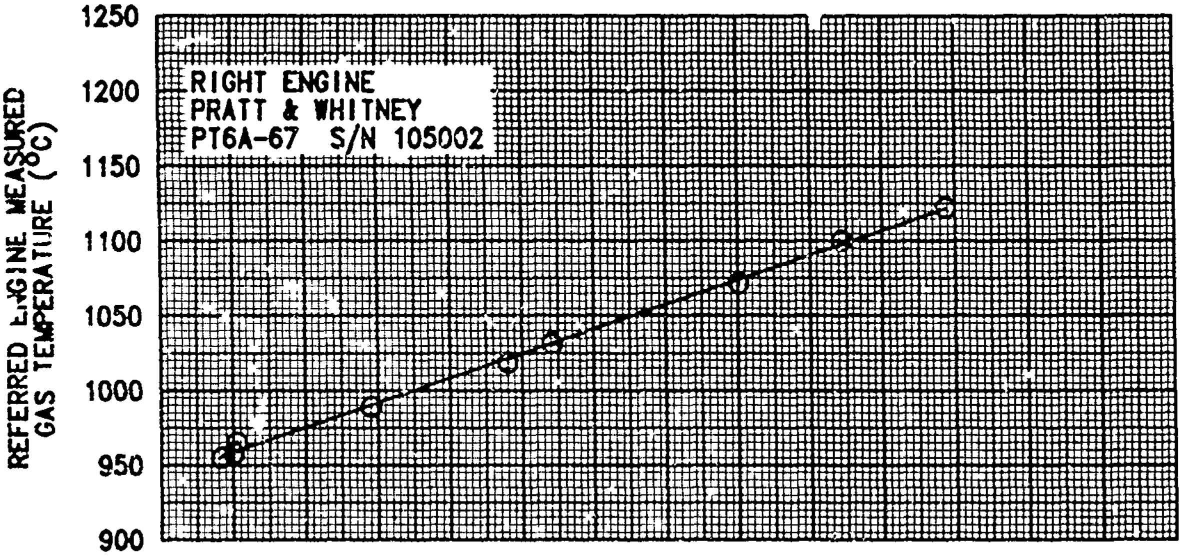


FIGURE E-6
ENGINE CHARACTERISTICS
RC-12K USA S/N 85-50149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION	TRIM FLIGHT CONDITION
15450	187.2 (FWD)	[illegible]	-44.5	1499	CRUISE	LEVEL



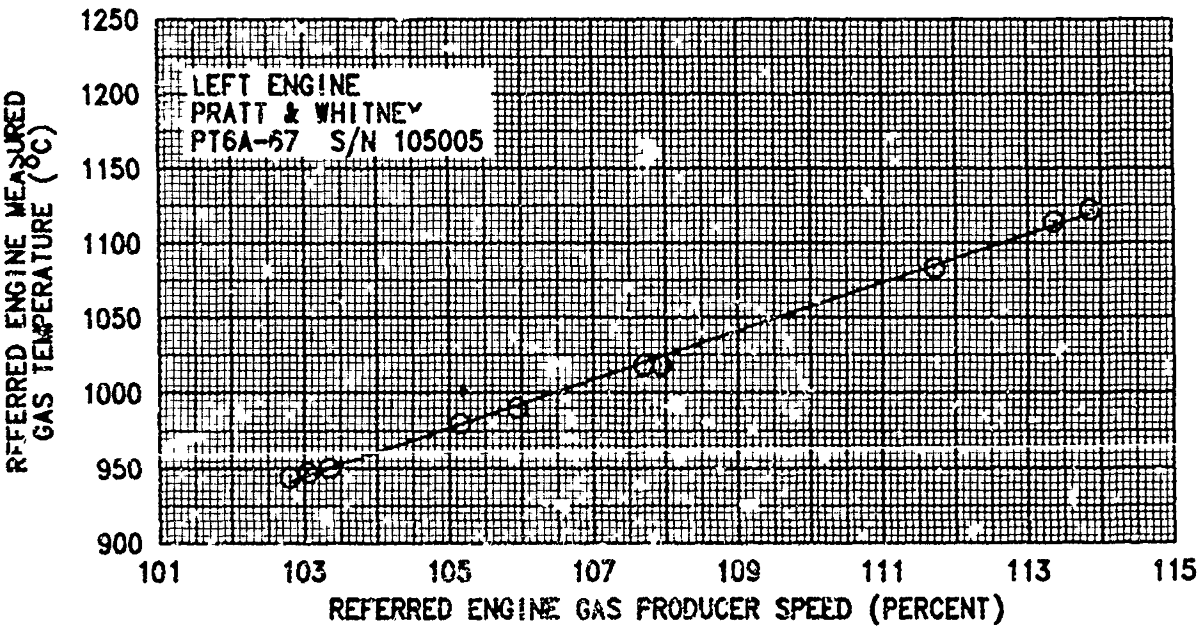


FIGURE E-7
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION
15450	187.2 (FWD)	31500	-44.[illegible]	1500	CRUISE

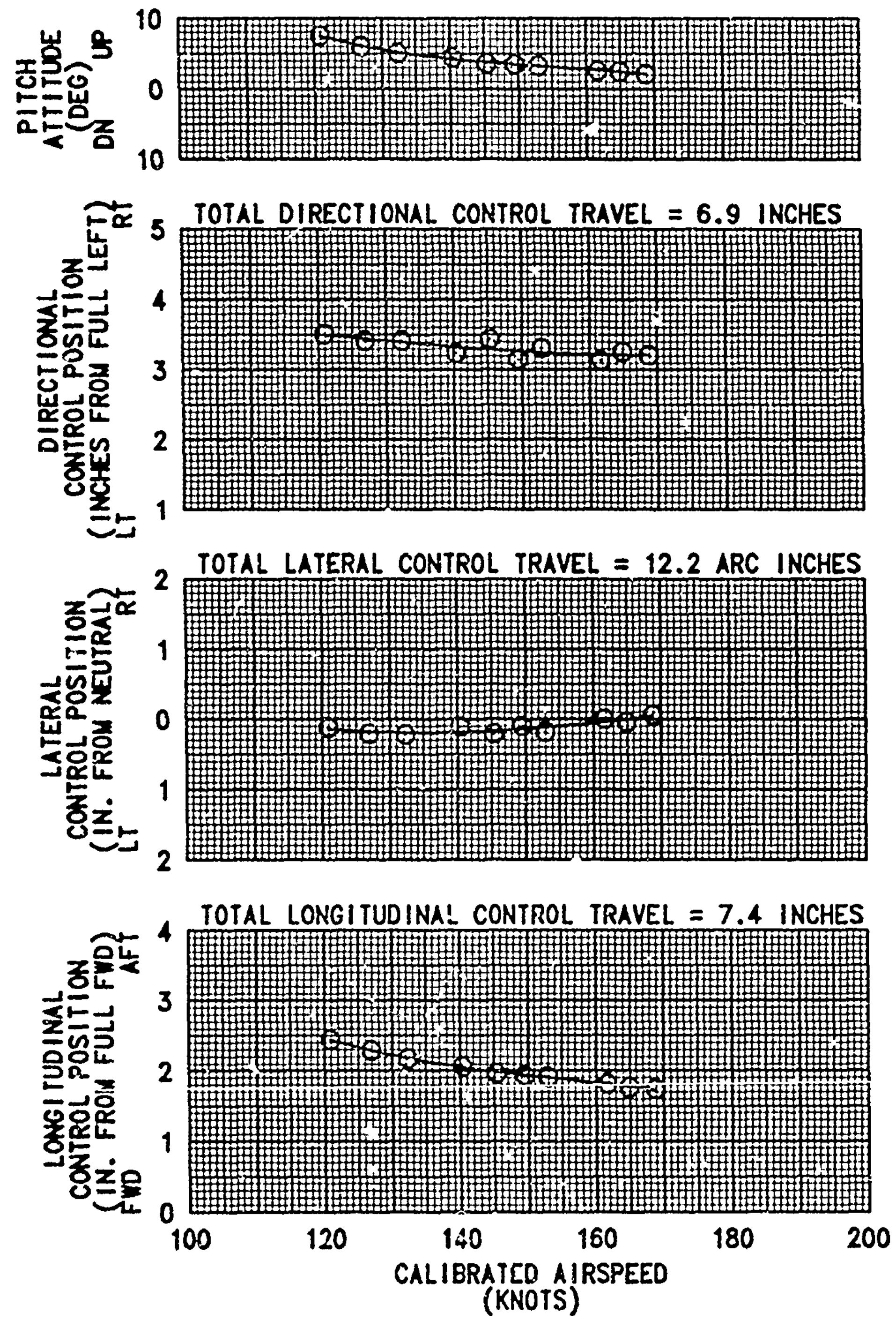


FIGURE E-8
STATIC LONGITUDINAL STABILITY
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIGURATION
16160	194.9 (AFT)	6260	22.5	1683	POWER APPROACH

NOTES: 1. SHADED SYMBOLS DENOTE TRIMMED IN LEVEL FLIGHT.
2. ⊡ DENOTE CONTROL FORCES.

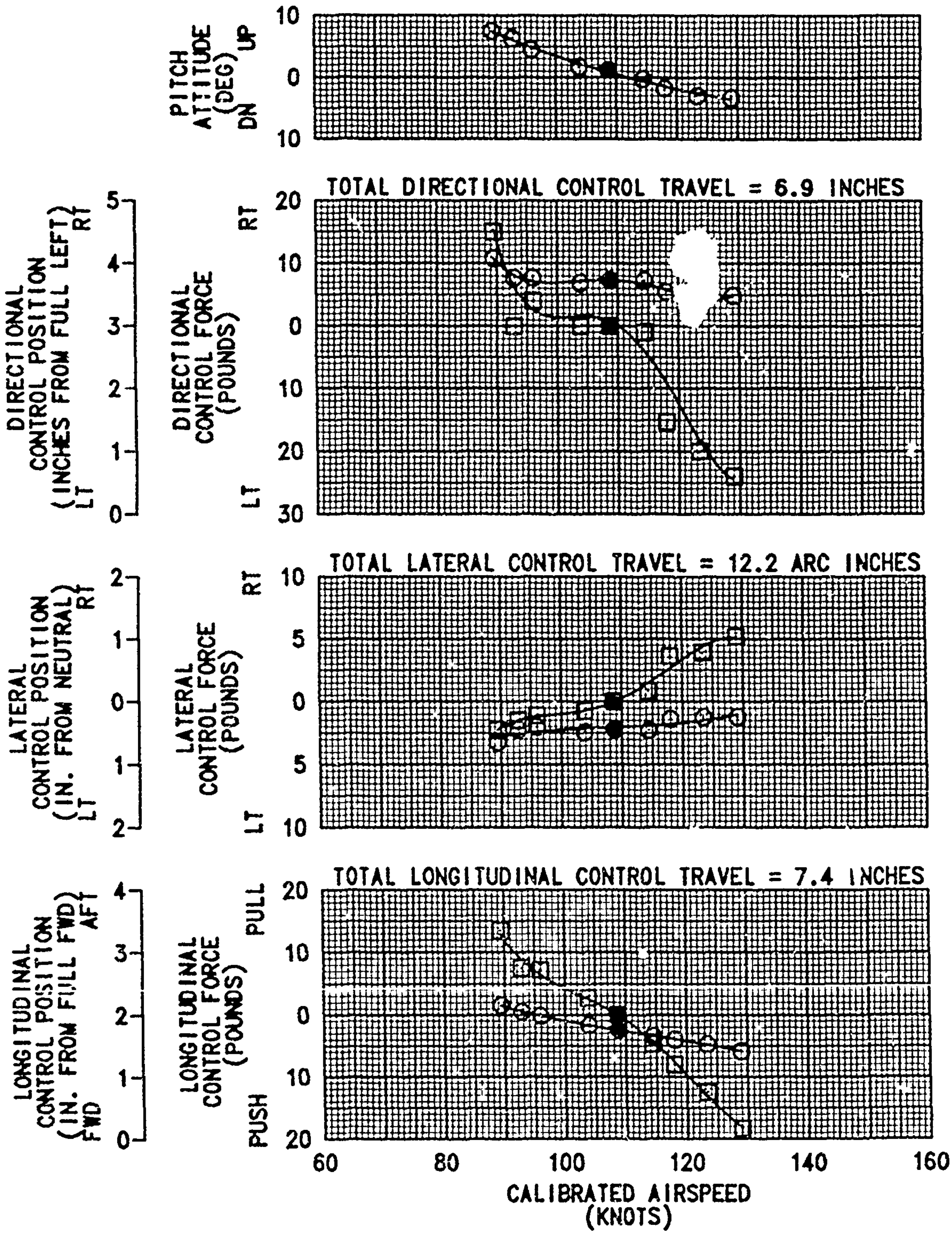


FIGURE E-9
STATIC LATERAL-DIRECTIONAL STABILITY
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15980	194.7 (AFT)	32060	-38.0	1491	131	CRUISE	OFF	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIMMED IN LEVEL FLIGHT.
2. ⊡ DENOTE CONTROL FORCES.

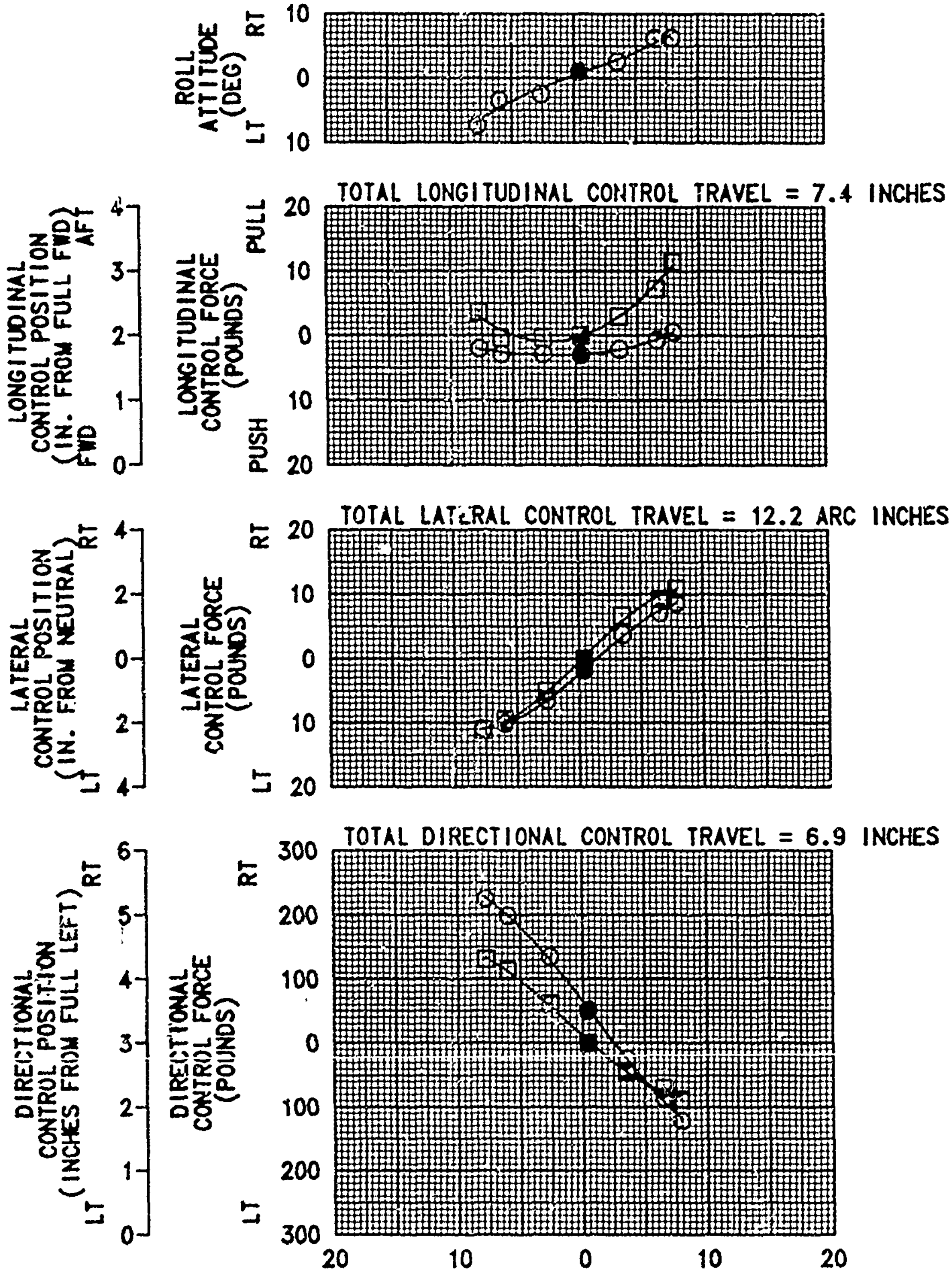


FIGURE E-10
STATIC LATERAL-DIRECTIONAL STABILITY
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15420	194.6 (AFT)	32090	-38.5	1497	161	CRUISE	OFF	OFF

NOTES: 1. SHADED SYMBOLS DENOTE TRIMMED IN LEVEL FLIGHT.
2. ⊡ DENOTE CONTROL FORCES.

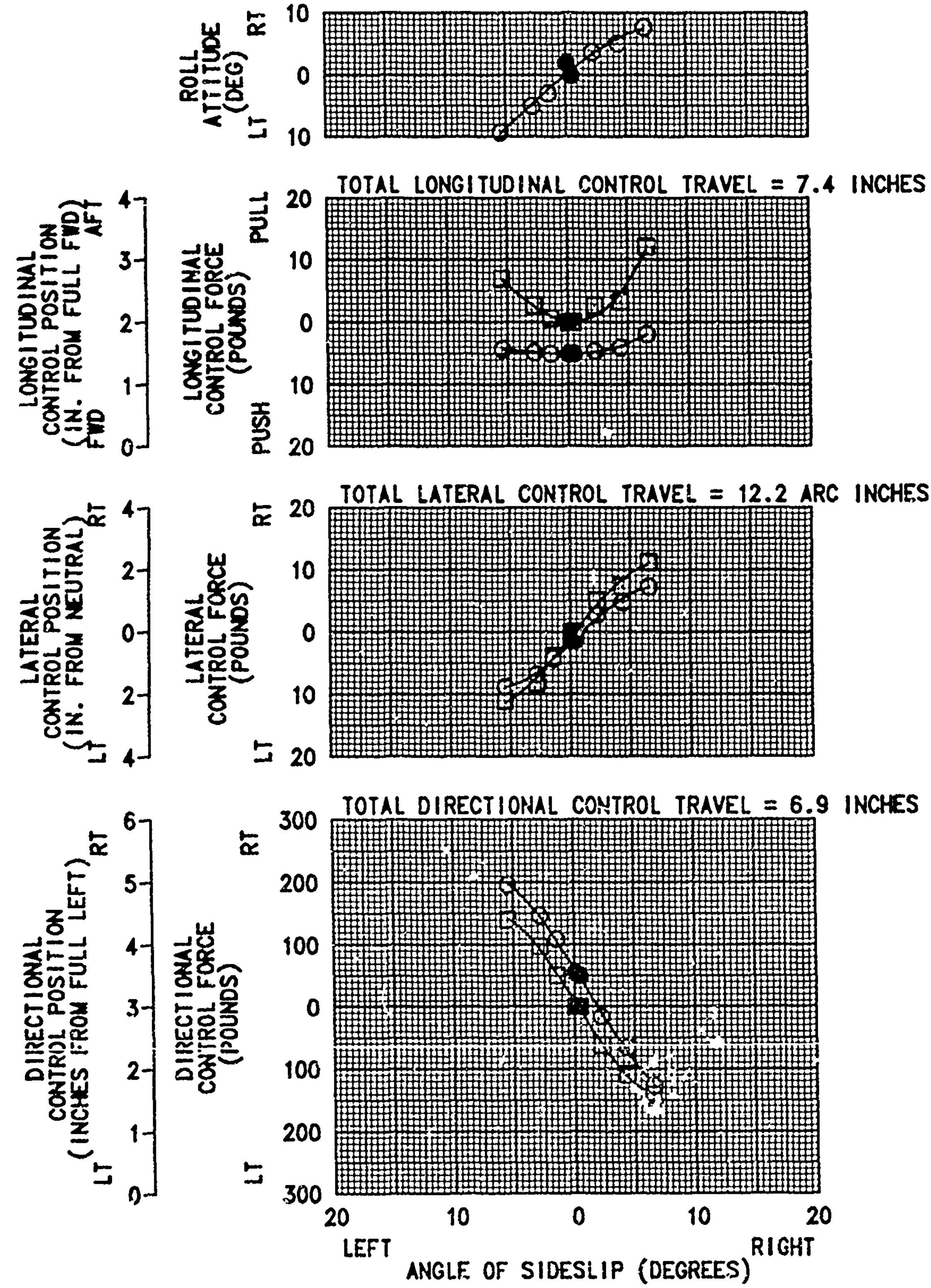


FIGURE E-11
DYNAMIC LONGITUDINAL STABILITY (PHUGOID)
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (^{o}C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
16020	194.8 (AFT)	6970	22.5	1679	109	PWR APP	OFF	ON

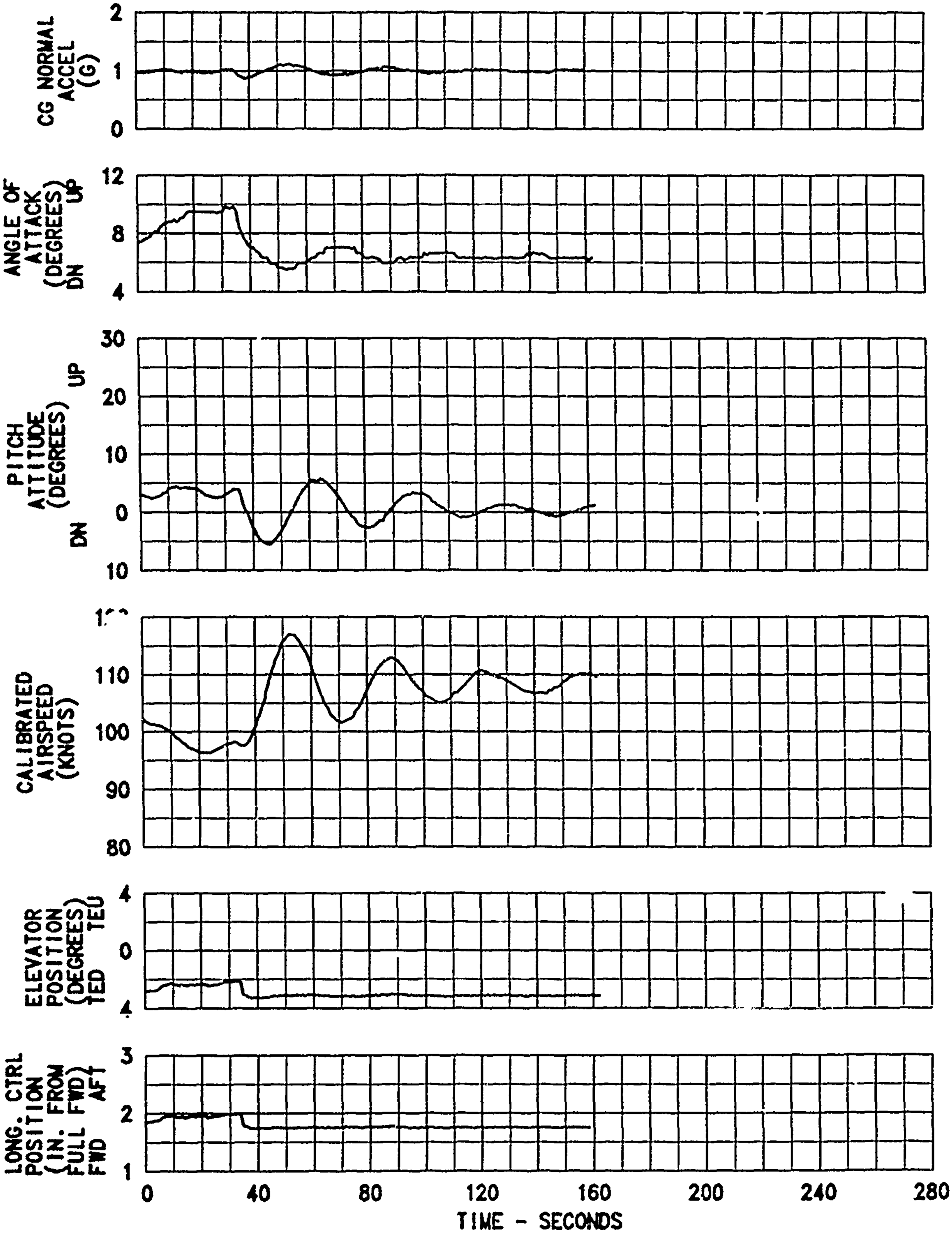


FIGURE E-12
DYNAMIC LONGITUDINAL STABILITY (SHORT PERIOD)
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15130	194.7 (AFT)	31920	-38.5	1493	131	CRUISE	OFF	ON

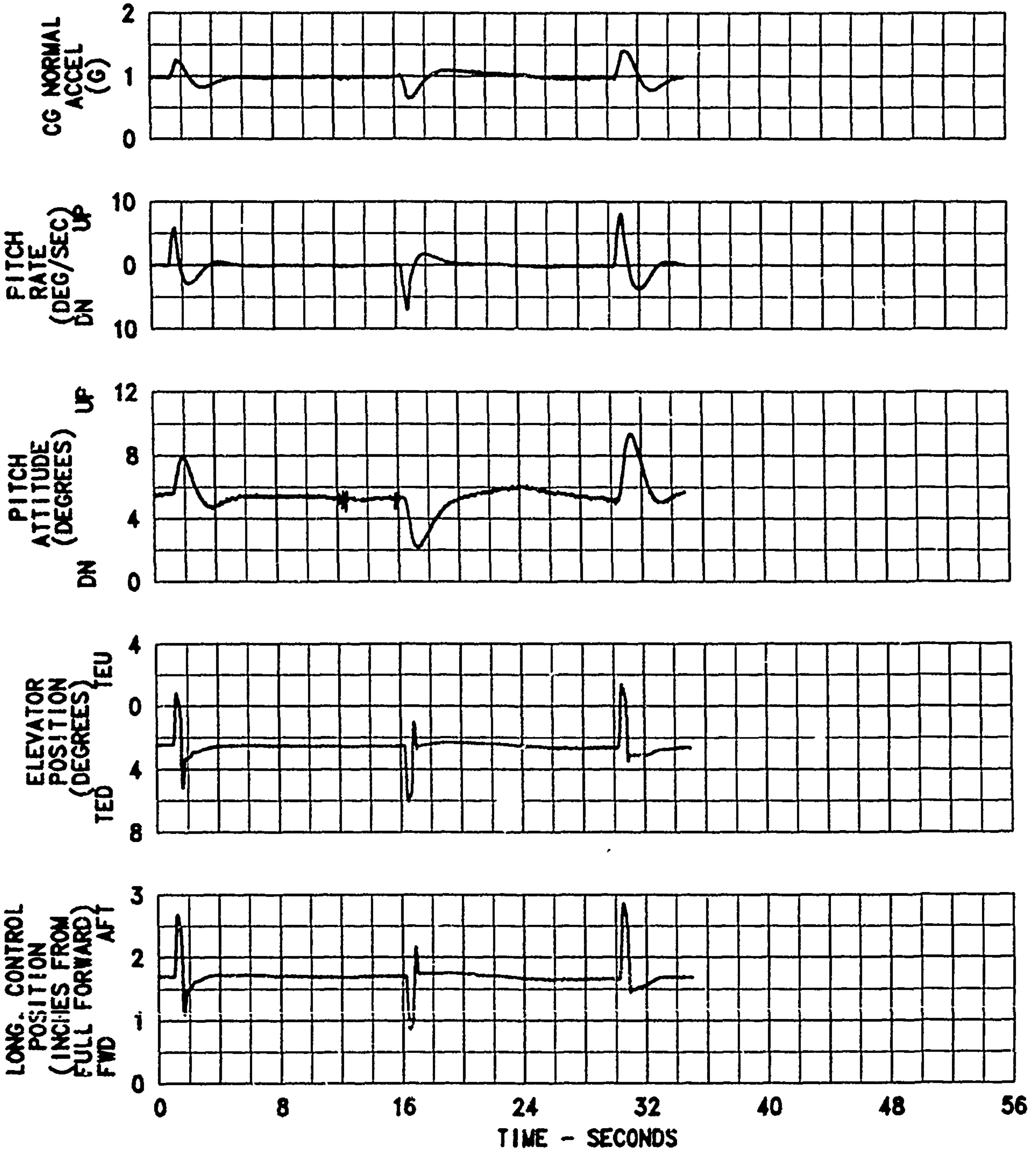


FIGURE E-13
DUTCH ROLL RESPONSE
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15660	194.5 (AFT)	32110	-38.5	1490	131	CRUISE	OFF	OFF

SOLID LINE SHORT DASH

NOTE: DIRECTIONAL CONTROLS FIXED.

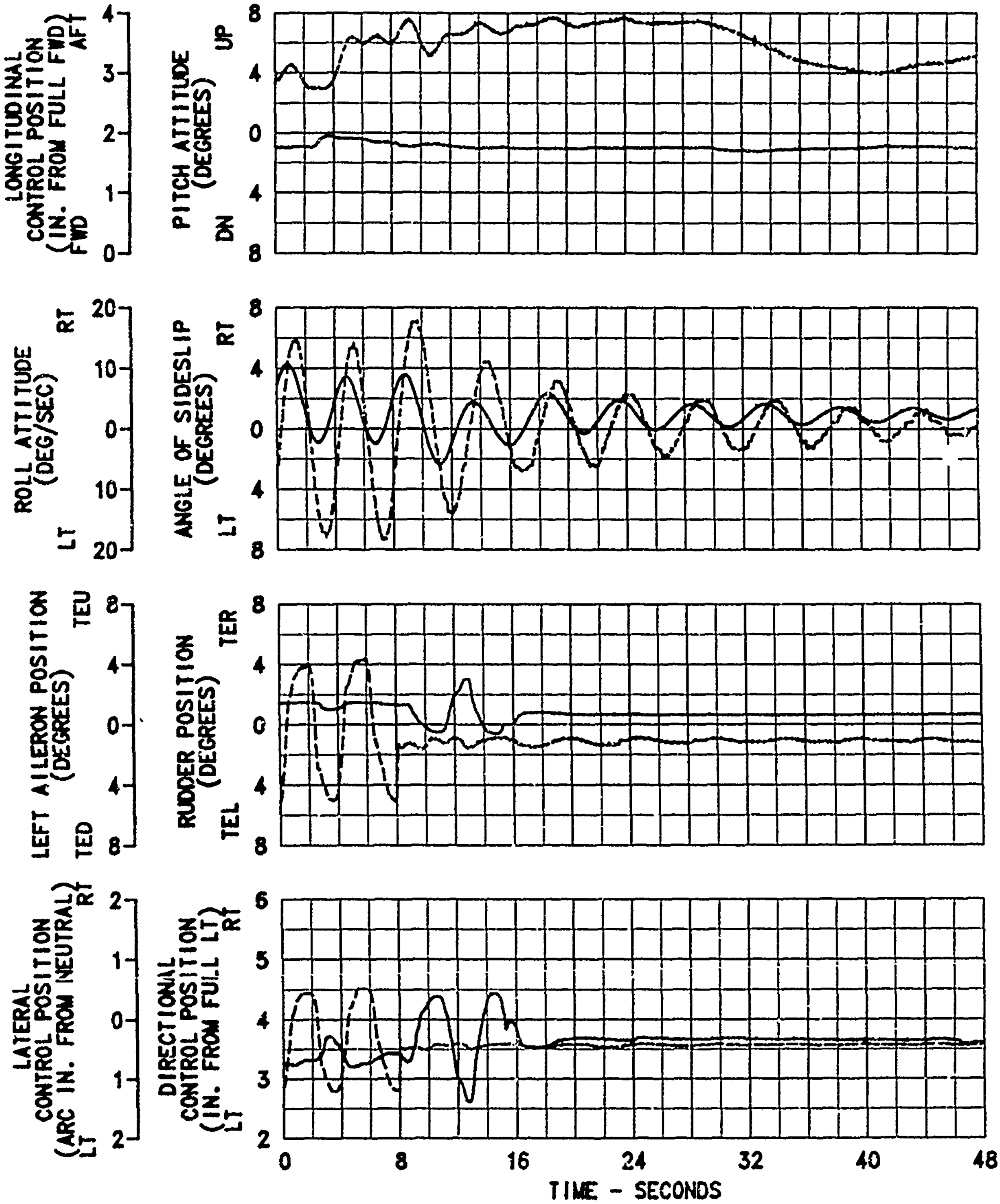


FIGURE E-14
DUTCH ROLL RESPONSE
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15670	194.6 (AFT)	32050	-38.0	1494	132	CRUISE	OFF	ON

SOLID LINE SHORT DASH

NOTE: DIRECTIONAL CONTROLS FREE.

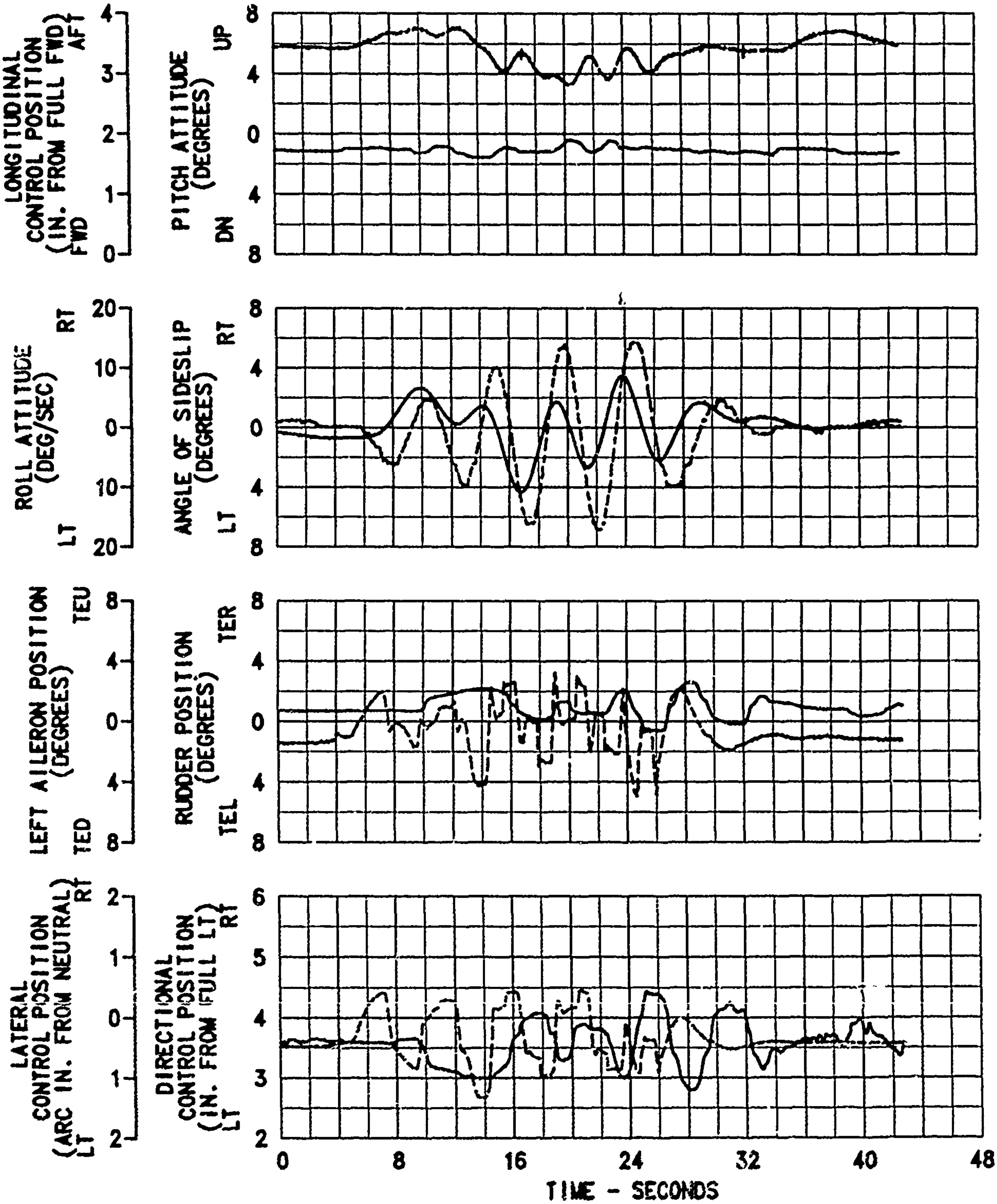


FIGURE E-15
STATIC SINGLE ENGINE MINIMUM CONTROL AIRSPEED (Vmc)
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIG	AFCS	YAW
14300	195.2 (AFT)	10390	19.3	1660	GO AROUND	OFF	OFF

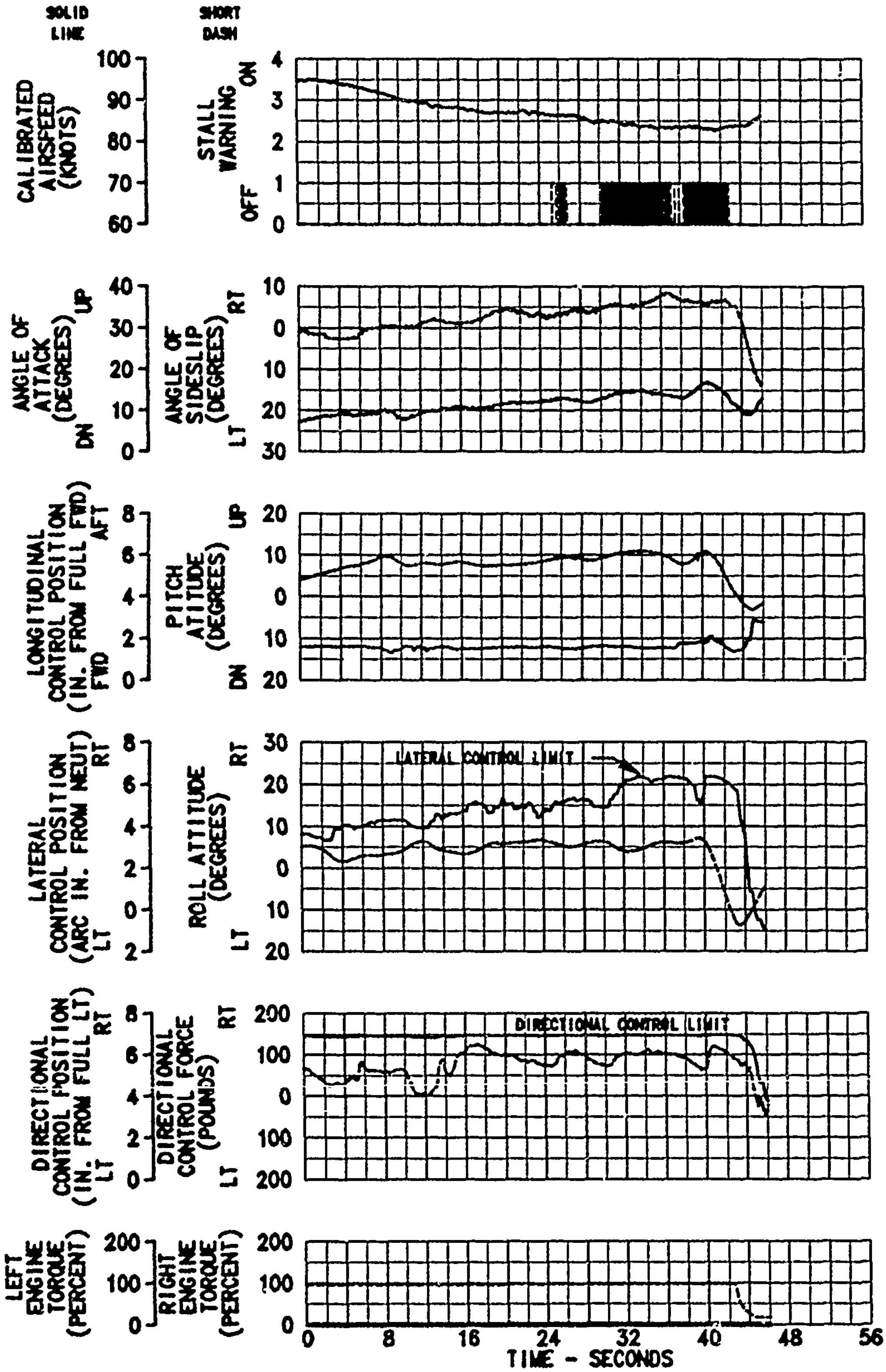


FIGURE E-16

DYNAMIC SINGLE ENGINE MINIMUM CONTROL AIRSPEED (Vmc)

RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
14740	194.9 (AFT)	14580	6.5	1677	TAKEOFF	OFF	OFF

SOLID LINE SHORT DASH

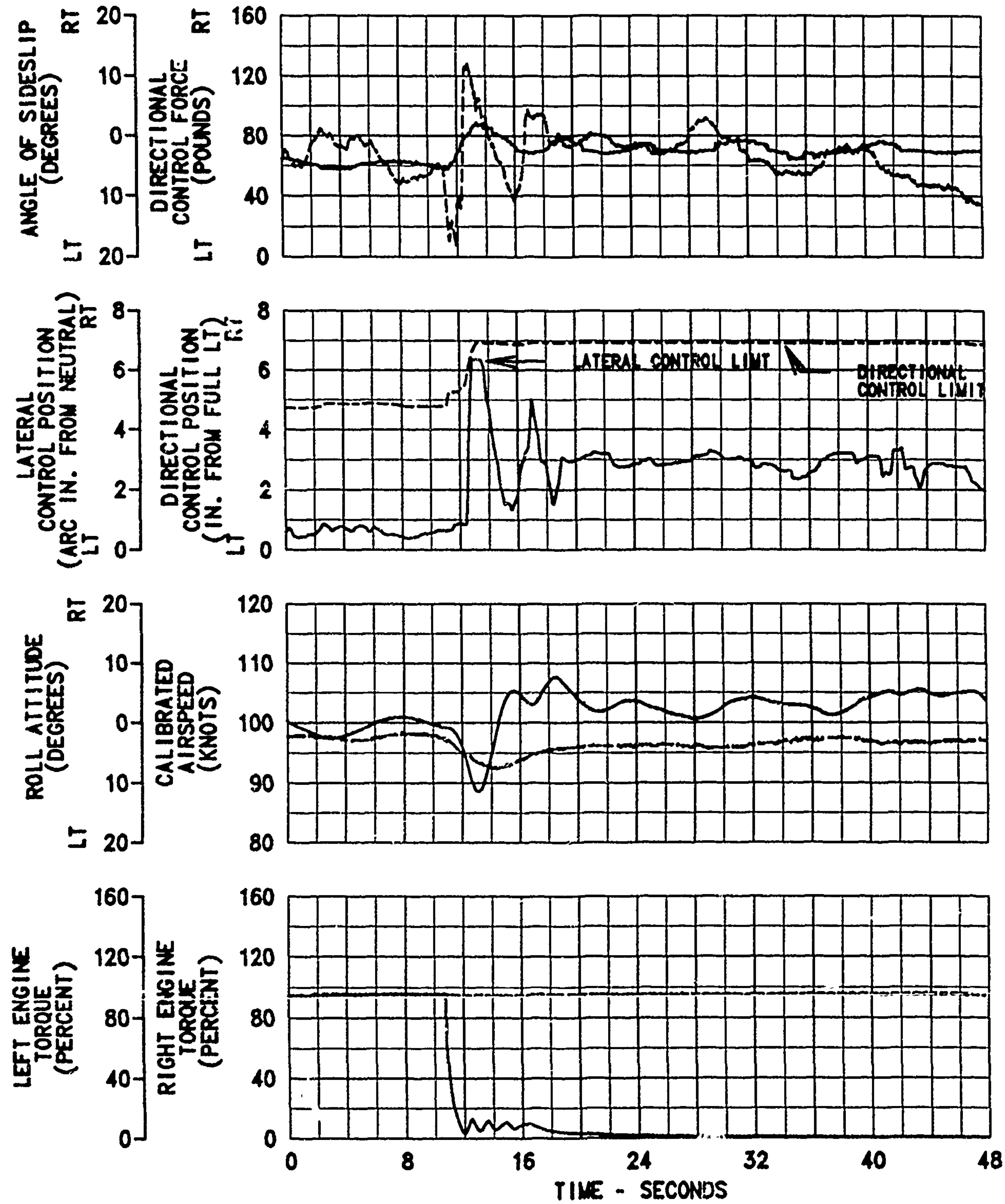


FIGURE E-17
ROLL PERFORMANCE
RC-12K USA S/N 85-0149

SYM	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG
⊙	15110	194.8 (AFT)	31390	-38.5	1494	132	CRUISE
⊡	15020	194.8 (AFT)	30970	-36.0	1502	164	CRUISE

NOTE: TIME FOR 60° BANK ANGLE CHANGE MEASURED FROM 30° BANK ANGLE IN ONE DIRECTION TO 30° BANK ANGLE IN THE OPPOSITE DIRECTION.

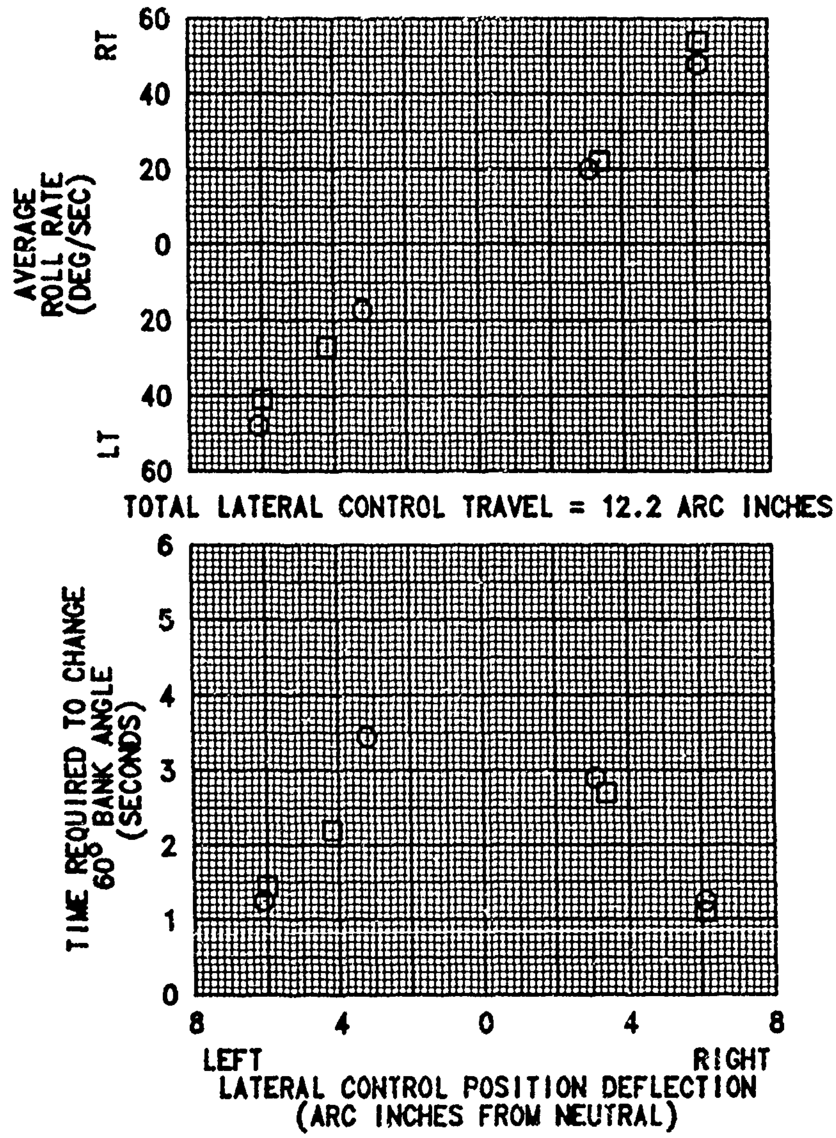


FIGURE E-18
ROLL PERFORMANCE
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15020	194.8 (AFT)	31070	-36.0	1501	162	CRUISE	OFF	ON

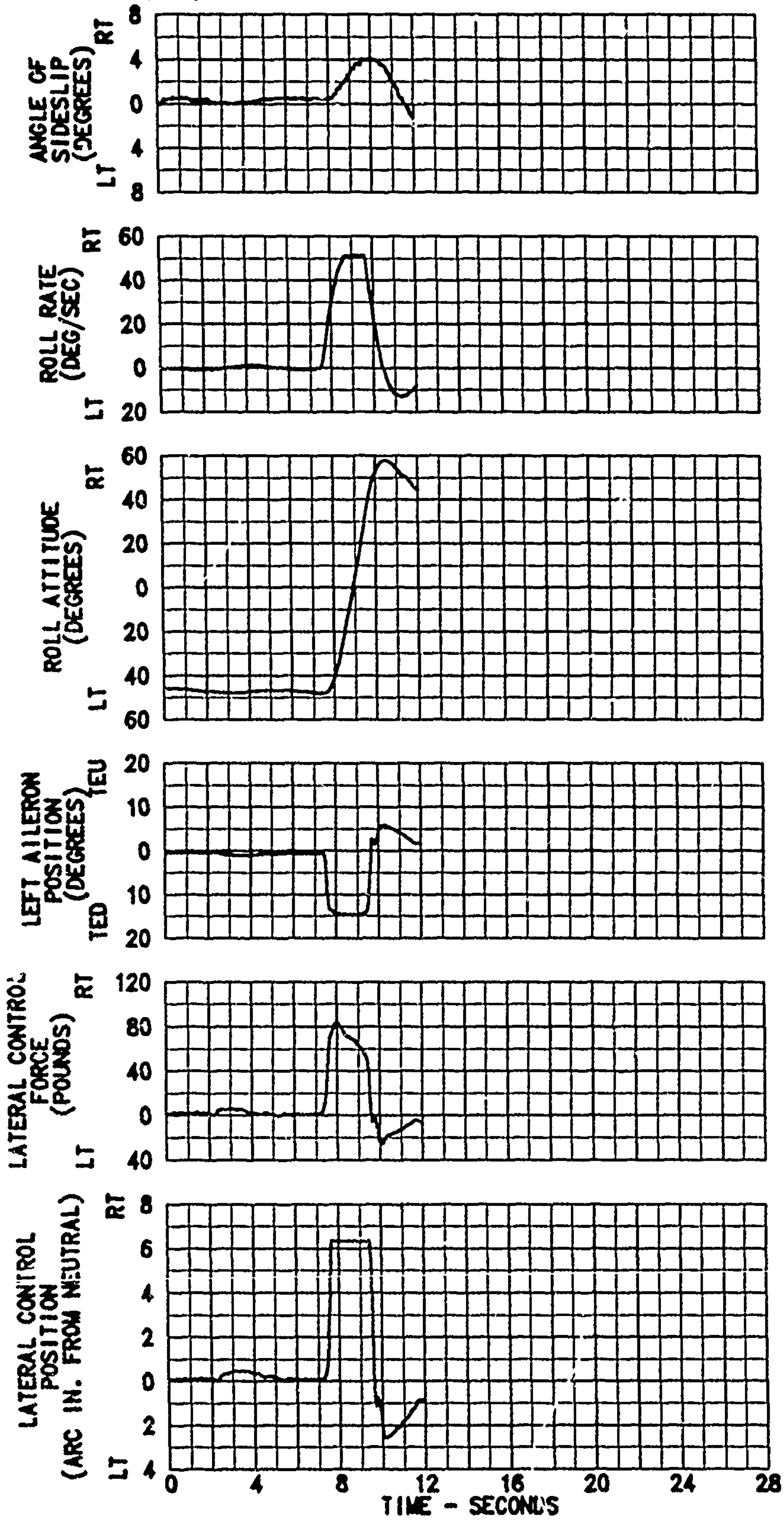


FIGURE E-19
ROLL PERFORMANCE
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15020	194.8 (AFT)	31140	-36.5	1504	164	CRUISE	OFF	ON

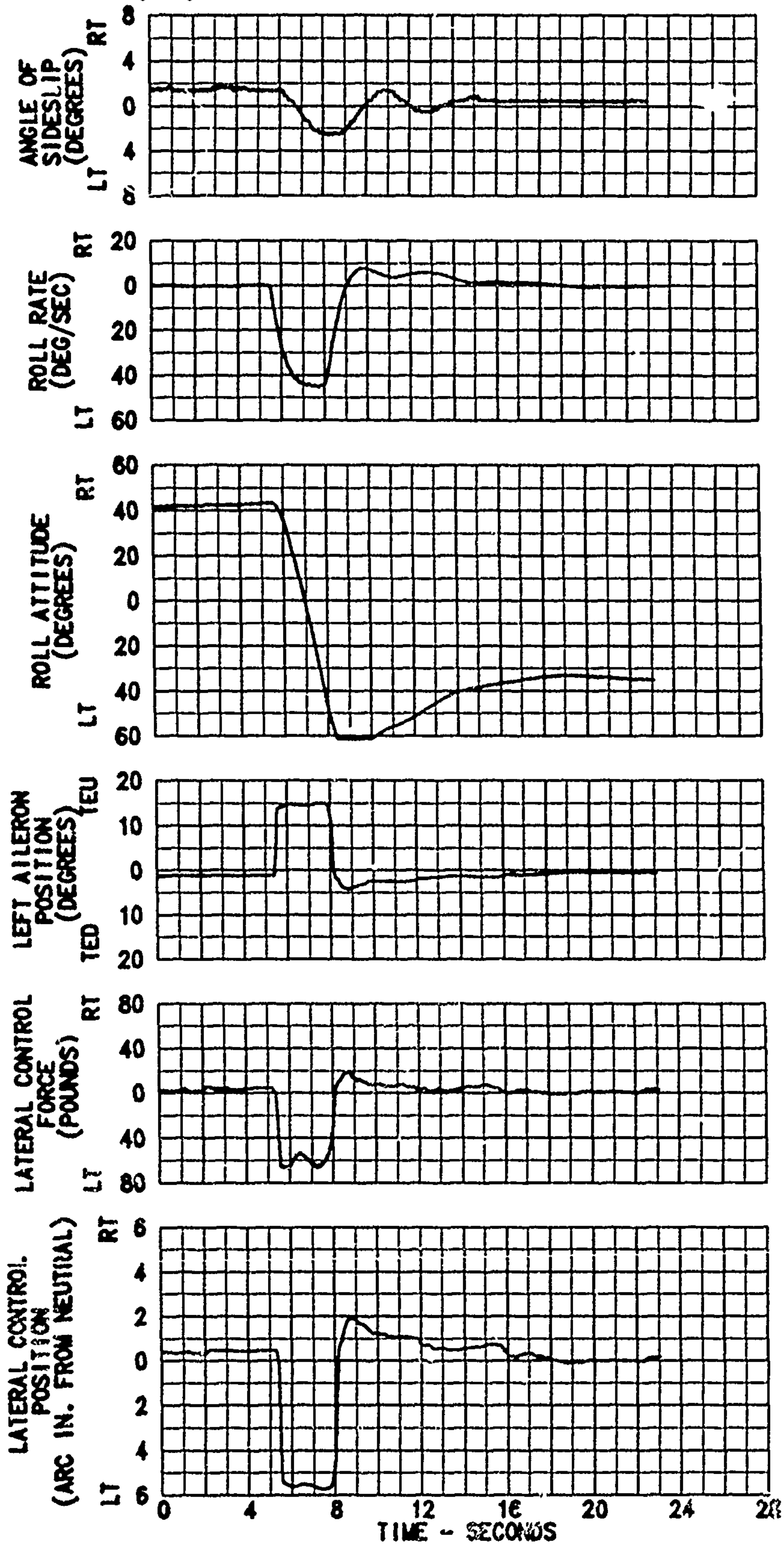


FIGURE E-20
AILERON OSCILLATION
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15220	187.0 (FWD)	29300	-35.0	1706	127	PWR APP	ON	ON

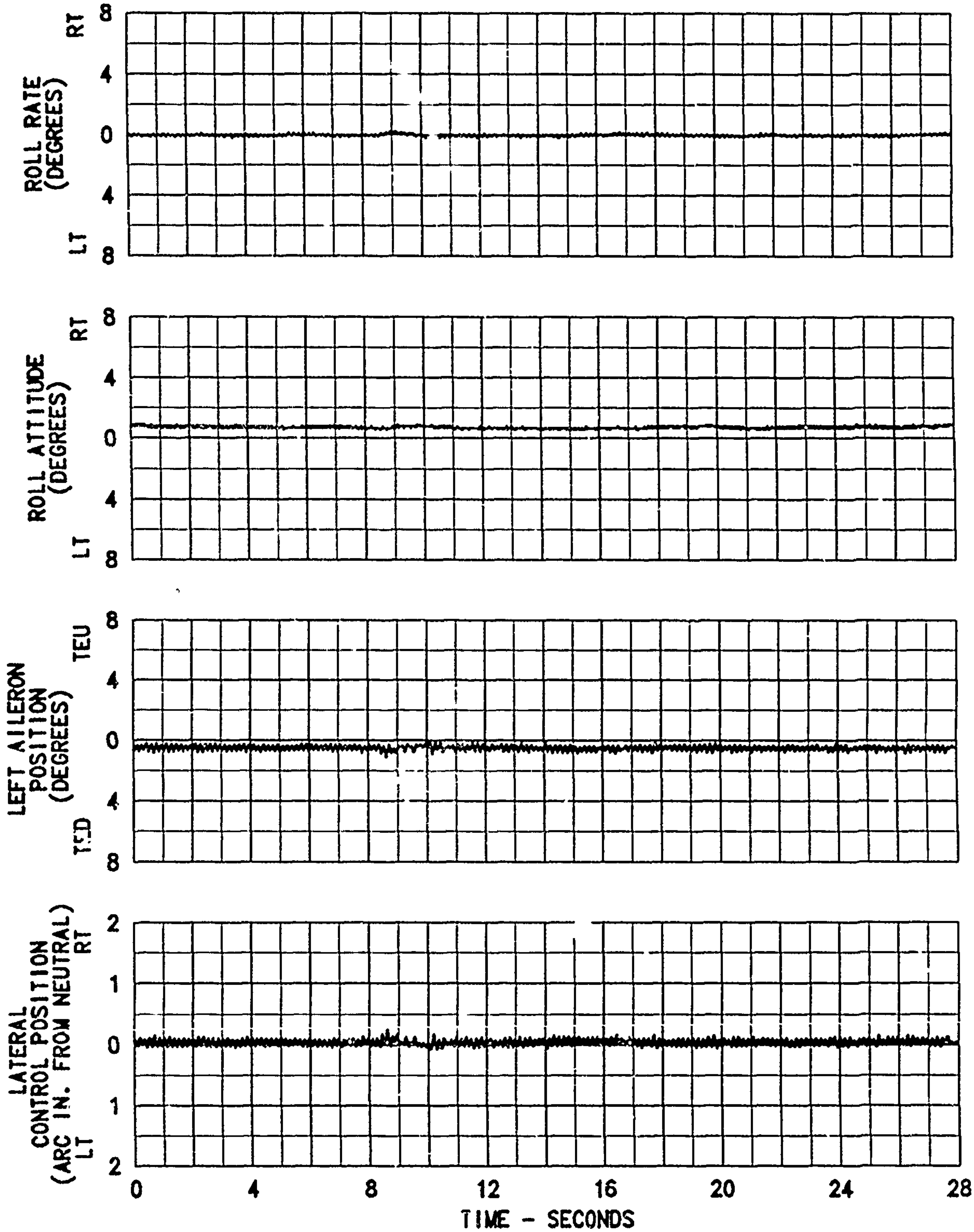


FIGURE F-21
AILERON OSCILLATION
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15080	187.0 (FWD)	25260	-25.0	1470	196	CRUISE	ON	ON

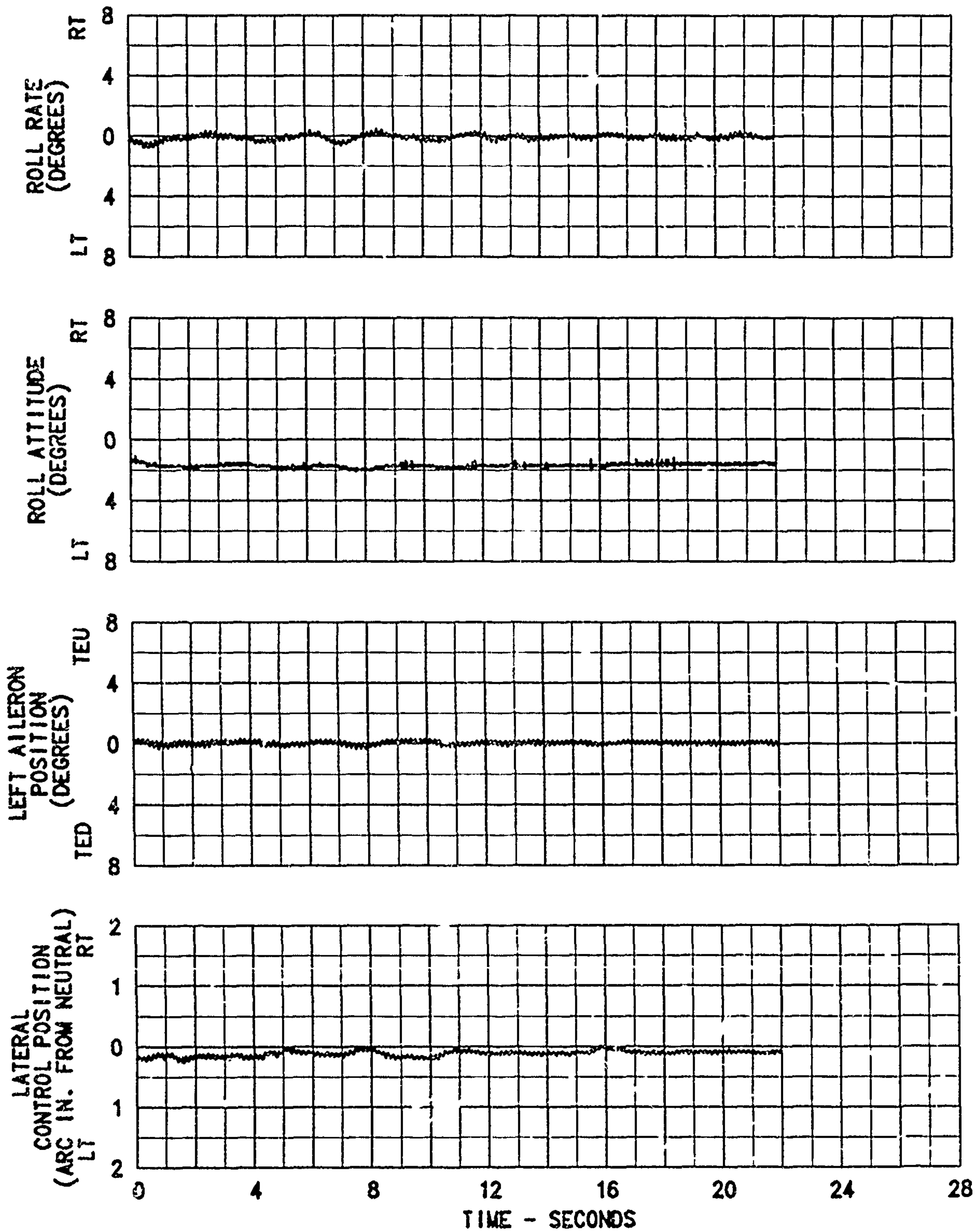


FIGURE E-22
PEDAL OSCILLATION
RC-12K USA S/N 85-0149

AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	AIRCRAFT CONFIG	AFCS	YAW DAMPER
15910	194.8 (AFT)	4800	24.0	1682	107	PWR APP	ON	ON

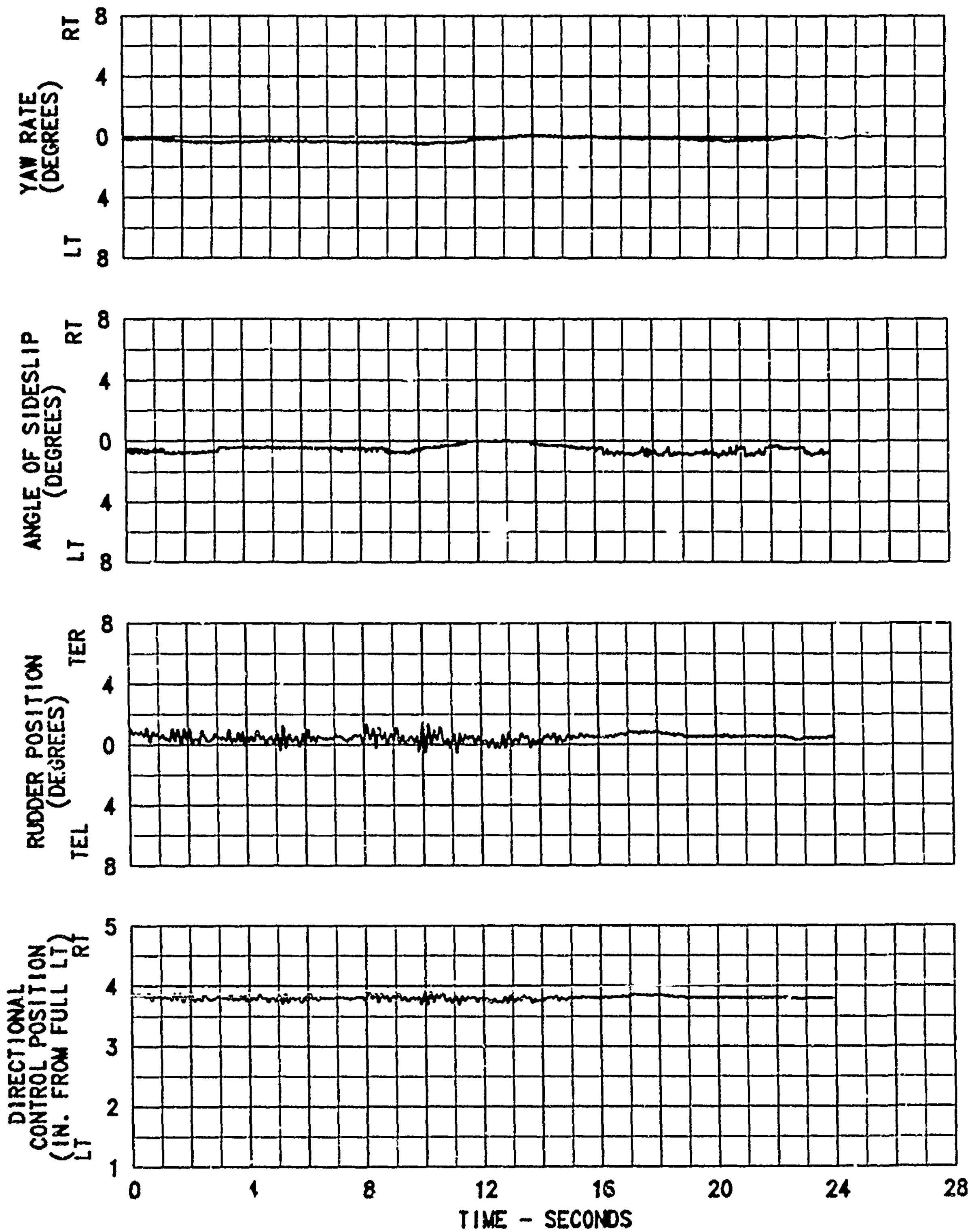
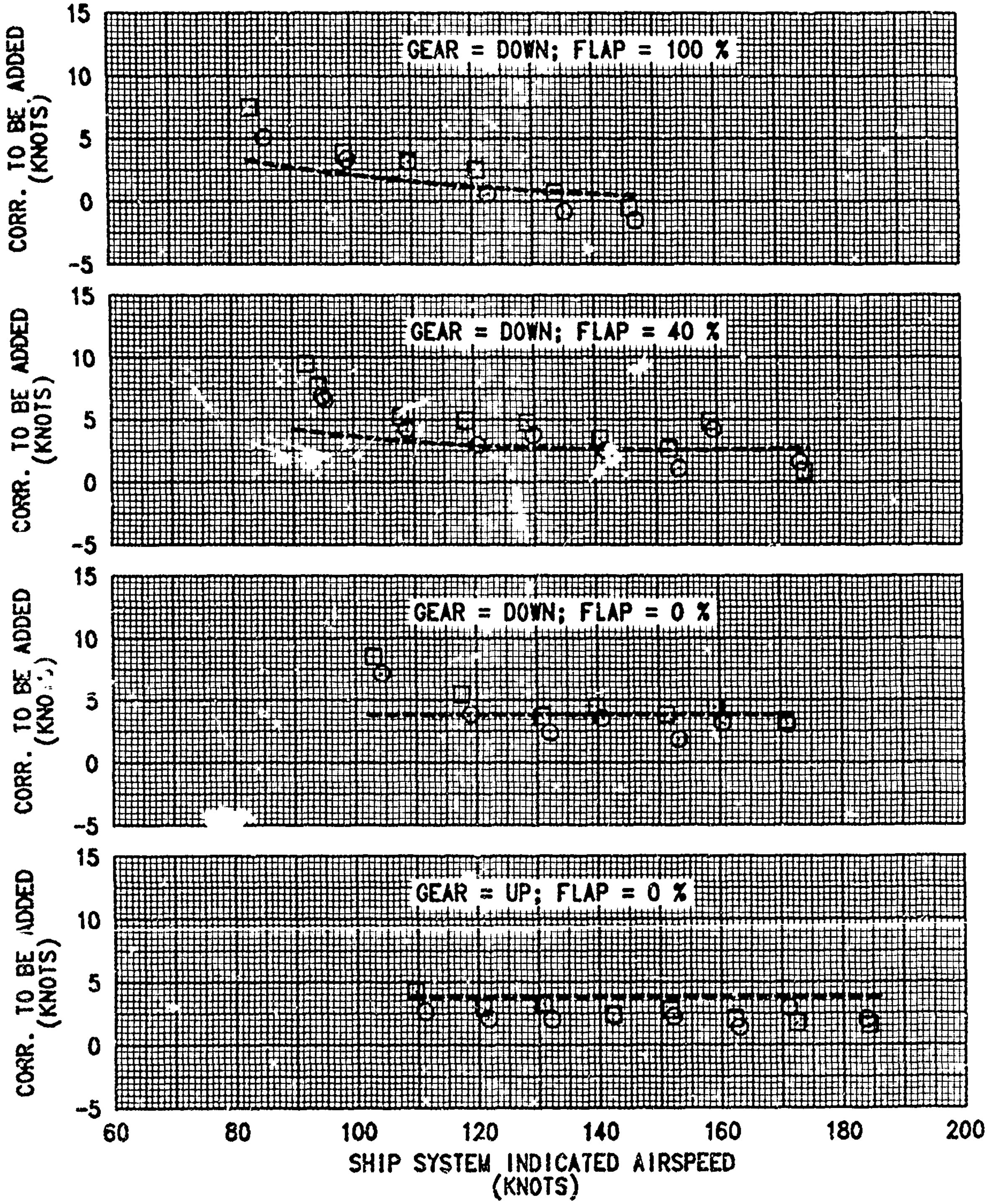


FIGURE E-23
SHIP SYSTEM AIRSPEED CALIBRATION
RC-12K USA S/N 85-0149

GEAR POSN	FLAP POSN (%)	AVG GROSS WEIGHT (LB)	AVG LONGITUDINAL CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG PROP SPEED (RPM)
UP	0	16030	187.6(FWD)	6630	7.0	1493
DOWN	0	14860	187.1(FWD)	8570	3.0	1496
DOWN	40	15530	187.1(FWD)	8760	4.5	1498
DOWN	100	15170	187.1(FWD)	8760	4.0	1495

NOTES:
1. A CALIBRATED T-34 PACE AIRCRAFT WAS UTILIZED.
2. ---- DENOTE BAC CALIBRATION CURVES
3. ⊙ DENOTE PILOT AIRSEED SYSTEM.
4. ⊡ DENOTE COPILOT AIRSPEED SYSTEM.



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ASD/AFXT, ASD/ENF	2
US Army Aviation Development Test Activity (STEBG-CT)	2
Assistant Technical Director for Projects, Code: CT-24 (Mr. Joseph Dunn)	2
6520 Test Group (ENML)	1
Commander, Naval Air Systems Command (AIR 5115B, AIR 5301)	3
Defense Intelligence Agency (DIA-DT-2D)	1
School of Aerospace Engineering (Dr. Daniel P. Schrage)	1
Headquarters United States Army Aviation Center and Fort Rucker (ATZQ-ESO-L)	1
US Army Aviation Systems Command (AMCPM-AE)	2