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TURBINE-DRIVEN NU-8F LIAISON TEST BED

Final Report

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April 1966

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-27(T)

BEECH AIRCRAFT CORPORATION
WICHITA, KANSAS

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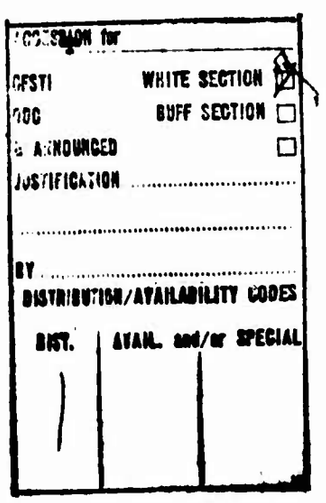
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ERRATA

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Page 4, paragraph 1 - Change sentence to read "That the NU-8, fully pressurized, be considered for the replacement and/or the aircraft to fill any future requirements for the mission support U-8D and U-8F for the following reasons:"

Page 4, paragraph 3 - Delete "in either the pressurized or the unpressurized version."

DEPARTMENT OF THE ARMY
U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA 23604

This report has been reviewed by the U. S. Army Aviation Materiel Laboratories and is considered to be technically sound. It is published for the dissemination of information and for the stimulation of discussion and consideration of turbine power for future Army aircraft.

This program was established to determine the advantages and disadvantages of turbine power in the Army's U-8F utility command aircraft. Also, through standardization of propulsion units, by using turbine power plants, a significant step may be made toward the Army's goal of single-fuel logistics.

Project 1M141809D180
Contract DA 44 177-AMC-27(T)
USAAVLABS Technical Report 66-22
April 1966

TURBINE-DRIVEN NU-8F LIAISON TEST BED

Final Report
Beech Report 20009

Prepared by

Beech Aircraft Corporation
Wichita, Kansas

for

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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ABSTRACT

The development and flight test program of the NU-8F was conducted by Beech Aircraft Corporation and United Aircraft of Canada, Ltd., for the U. S. Army Aviation Materiel Laboratories under Contract DA 44-177-AMC-27(T). A Beech Model U-8F was modified by the installation of two UACL PT6A-6 turbine engines in place of the reciprocating engines, and by replacing the stabilizer and rudder with a Beech Model A80 stabilizer and rudder. Ground and flight tests were conducted using FAA regulations as the standard of acceptability. The report contains a comparison of performance characteristics of the NU-8F and the U-8F. A discussion of each phase of the performance tests along with tables and graphs of the flight test data is presented in some detail. The NU-8F has more useful load, increased speed, and a more simple fuel system than the U-8F. It is suitable for use as a turbine-powered trainer or as a liaison and transportation aircraft.

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SUMMARY

The development and test program conducted for the turbine-engine-powered NU-8F aircraft is summarized in this report. The program was a joint effort by Beech Aircraft Corporation, the airframe manufacturer, and United Aircraft of Canada, the engine manufacturer. Product modification, installation, and testing were accomplished by Beech Aircraft Corporation.

The nucleus of the modification program consists of a Model U-8F aircraft modified by replacing the original reciprocating Lycoming 0-480-3 engines with UACL PT6A-6 turbine engines and adding a swept-back vertical stabilizer and a ventral fin below the empennage. Also, many interior and structural modifications were made. The resultant aircraft is redesignated NU-8F.

This report concerns the effect of these structural modifications specifically for the prototype NU-8F aircraft, by presenting a detailed discussion of the performance tests along with various tables and a graphical presentation of flight test data. A comparative summation of the performance characteristics of the U-8F and the NU-8F aircraft is contained in Table I.

TABLE I
COMPARATIVE PERFORMANCE
OF THE U-8F AND NU-8F AIRCRAFT

Item	U-8F Aircraft	NU-8F Aircraft
Maximum Gross Weight (lb)	7,700	9,000
Empty Weight (lb)	5,342	5,186
Fuel Weight (lb) (1)	1,380	2,340
Payload (lb) (2)	682	1,234
Power Plant Rating, Sea Level		
Take-off	340 BHP	500 SHP
Maximum Continuous	320 BHP	500 SHP
Maximum Cruise	208 BHP	471 SHP
High Speed at Sea Level (mph)	211	254 (3)
High Speed at Optimum Altitude (mph)	238	276

TABLE I (Contd.)

Item	U-8F Aircraft	NU-8F Aircraft
Maximum Cruise Speed (mph)		
At 5,000 Ft	188	259
At 10,000 Ft	196	269
At 15,000 Ft	206	269
At 20,000 Ft	217	266
At 25,000 Ft	-	258
Range at 15,000 Ft, Maximum Cruise (Statute Miles) (Mil. Spec Reserves, 682-Lb Payload)	1,028	1,140
Rate of Climb (fpm)		
Two Engines at Sea Level	1,304	2,000
Single Engine at Sea Level	184	535
Service Ceiling (ft)		
Two Engines, 100 FPM	27,100	27,400
Single Engine, 50 FPM	12,100	12,950
Stall Speed		
Gear and Flaps Up	95	105.4
Gear and Flaps Down	83	90
Take-off Distance, Flaps Up (ft)		
Ground Run	1,320	1,640
Total Over 50 Ft	2,200	1,890
Landing Distance, Flaps Down (ft)		
Maximum Landing Weight (lb)	7,350	8,550
Ground Run	1,350	1,280
Total Over 50 Ft	2,130	2,240

NOTES: (1) 6.0 pounds per gallon for gasoline.
6.5 pounds per gallon for JP-4.

(2) Payload is equal to maximum take-off gross weight less empty weight, unusable fuel and oil, full oil, full fuel, and pilot (200 pounds).

(3) Exceeds V_{M_O}.

The conclusion reflected by this report is that the NU-8F aircraft is generally superior to the U-8 aircraft throughout all performance and maintenance parameters.

CONCLUSIONS

Throughout the manufacturer's entire development and testing program on the NU-8F aircraft, product versatility, stability, and ruggedness were constantly apparent. The presence of these inherent characteristics is also promulgated by the aircraft design evaluation and reliability testing conducted and reported by the U. S. Army.

The primary function of this technical report is to describe the NU-8 aircraft and its characteristics in sufficient detail (through direct description and comparative tabulation with the U-8F aircraft) to inform, and to provide an evaluation basis for, responsible personnel. It is the conclusion of the contractor that this document fulfills these requirements.

RECOMMENDATIONS

It is recommended:

1. That the NU-8F, fully pressurized, be the attrition replacement and/or the aircraft to fill any future requirements for the mission support U-8D and U-8F for the following reasons:

   a. Immediately available "off-the-shelf".
   b. Turbine powered.
   c. Ease of maintenance.
   d. All fuels compatible.
   e. Over-the-weather type flying.
   f. Out of congested low air space.
   g. Airframe the same as U-8F.
   h. Lower maintenance cost.
   i. Lower costs per seat mile.
   j. Advanced all-weather equipment.
   k. Higher cruising speed.
   l. Higher payload.

2. That the NU-8F, unpressurized or pressurized (UP or P), be considered as multi-mission aircraft for use in the following categories:

a.	Tactical/utility aircraft.	UP
b.	High-priority cargo.	UP/P
c.	Ambulatory evacuation.	UP/P
d.	Paradrop - medics or supplies.	UP/P
e.	Mission support.	UP/P
f.	Transition training to larger turbine-powered aircraft.	UP/P
g.	High-altitude photo reconnaissance.	P
h.	Forward-area radar surveillance.	UP/P
i.	Battlefield airborne command post and/or communication relay station.	UP/P

3. That the proposed growth version of the NU-8F, the "King Air C", be considered in any future tactical-utility or mission support requirements in either the pressurized or the unpressurized version.

DESCRIPTION OF AIRCRAFT

THE AIRCRAFT

The NU-8F aircraft, shown in Figure 1, is a low-wing, all-metal utility type aircraft of versatile design featuring turbine engines and all-weather capability. Variations in interior arrangement and equipment installations permit the aircraft to be used as a command/liaison vehicle for transporting personnel and high-priority cargo, with capabilities of operating from rough or unimproved areas. Complete de-icing and anti-icing systems and instrumentation and navigation equipment installation provide maximum safety under full instrument (IFR) conditions, including icing. Wide-angle visibility is afforded the pilot, copilot, and passengers through the large safety-glass windshield and side windows.

Distinguishable features of the aircraft are the slender, streamlined engine nacelles, the square-tipped wing and tail surfaces, the swept-back vertical stabilizer, and the ventral fin installation below the aircraft's empennage section. Interior features include variable seating arrangements which may be removed, thereby providing an unrestricted loading space for cargo and equipment transportation. Cabin entrance is made through the stair-type door on the left side of the fuselage just aft of the wing trailing edge.

Advantages of turbine-powered aircraft operation, comparable to a like version of reciprocating engine aircraft, are generalized by an overall increase in aircraft performance and efficiency coupled with vibration-free operation and a remarkably low sound level.

The overall dimensions for the NU-8F aircraft are shown in Figure 2.

THE ENGINE

The NU-8F aircraft is powered by two Model PT6A-6 turbine engines manufactured by United Aircraft of Canada, Ltd. (UACL). Large, hinged cowling doors provide maximum accessibility to the engine and its controls and accessories, making routine inspections and maintenance operations easier to perform. The engine and engine cowling package are removable as a single unit from the aft engine firewall. This simplifies engine change procedures. In addition, each engine is completely interchangeable and may be installed in either the right or the left position.

The PT6A-6 engine, illustrated in Figure 3, is a free-turbine engine. The engine utilizes two independent turbines: the compressor turbine which



Figure 1. Flight View of NU-8F Aircraft.

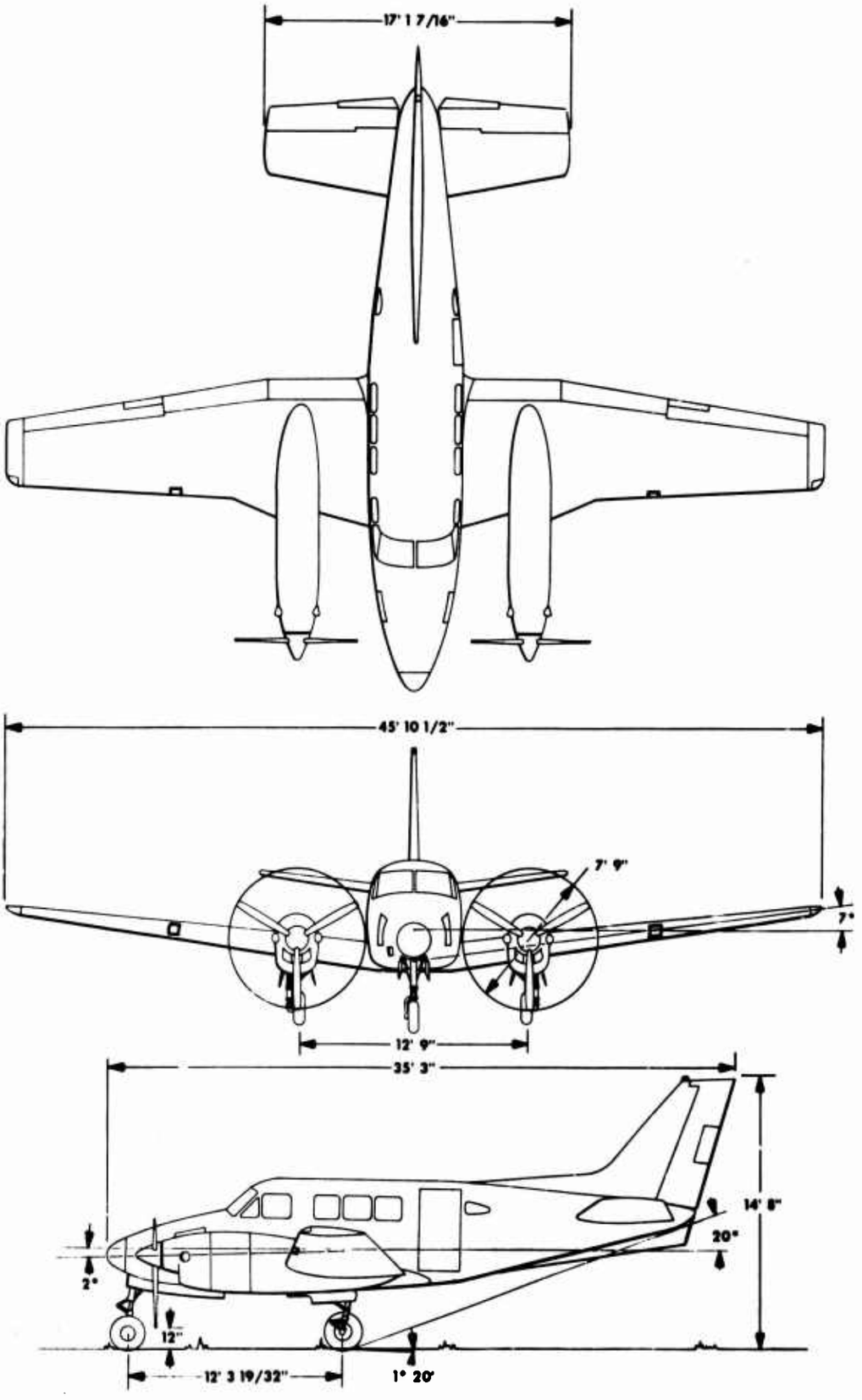
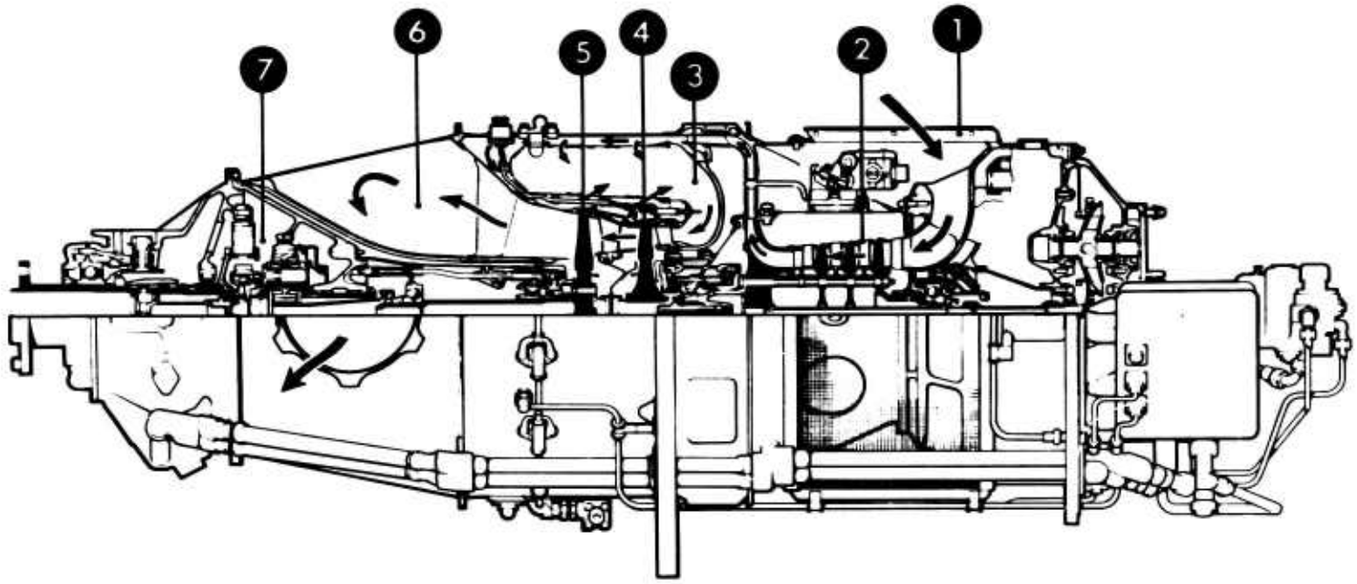


Figure 2. NU-8F Three-View.



Air Induction
1. Engine Inlet
2. Compressor
3. Combustion Chamber
4. Compressor Turbine
5. Power Turbine
6. Exhaust
7. Reduction Gear

Figure 3. PT6A-6 Turbine Engine.

drives the compressor assembly, and the power turbine which drives the propeller shaft through a reduction gear train. The compressor assembly consists of three axial stages and one centrifugal stage, assembled as an integral unit. It provides a compression ratio of 6.3:1. The single-stage power turbine is connected to the reduction gearbox at the front of the engine. The flanged propeller shaft is connected to the output gear of the reduction gearbox. The two-stage planetary gear train in the reduction gearbox provides a reduction ratio of 15:1. Thus, when the power turbine is operating at 33,000 rpm, the propeller shaft will be operating at 2200 rpm.

The PT6A-6 turbine engine utilizes an annular, reverse-flow combustion chamber. The combustion chamber encircles both the compressor and the power turbines. The flow of compressed air must reverse direction before entering the combustion chamber. The combustion chamber air inlet consists of a circular shroud and perforations of various sizes in the chamber wall. The reverse air flow pattern and the circular air inlet shrouding the compressor section combine to eliminate the high-pitched "whistle" normally associated with turbine engines.

The accessory gear case, located at the rear of the engine, contains the necessary accessory drives and mounting pads. Power is supplied to the accessory gear case by a stub shaft connected to the compressor. A torquemeter instrumentation system is included as an integral part of the reduction gearbox to provide an accurate measurement of the engine power output at all times. The integral engine oil tank is located between the compressor air inlet and the accessory gear case.

The PT6A-6 turbine engine develops 500 shaft horsepower at take-off power (maximum power), and produces 70 pounds of jet thrust at a propeller shaft speed of 2200 rpm. Normal rated power (maximum continuous power) is also 500 shaft horsepower at a propeller shaft speed of 2000 rpm.

MODIFICATION PROGRAM

On 1 November 1962, a new production Beech Model U-8F aircraft, serial number LD-75, was assigned for modification to the turbine-engined configuration.

The mcdification was performed in the Beech Experimental Department.

The turbine engines were supplied by United Aircraft of Canada, Ltd., and electronic equipment was supplied by the U. S. Army.

The aircraft was completed and ready for final inspection and ground test programs on 4 May 1963.

Conformity and safety inspections, a flutter and vibration survey, and the miscellaneous ground and taxi tests were completed by 14 May 1963.

The initial test flight occurred on 15 May 1963.

STRUCTURAL AND EQUIPMENT CHANGES

The following structural and equipment changes were required to provide for the turbine installation:

1. The Lycoming 0-480-3 reciprocating engines were replaced with UACL PT6A-6 turbine engines.
2. Hartzell HC-93Z20-2C1/10151B-8 propellers were replaced with HC-B3TN-2/T10173B8 propellers.
3. The engine controls, engine mounts, cowlings, and nacelles were redesigned to suit the turbine engine installation.
4. The engine instruments installation was redesigned to suit the turbine engine installation.
5. The fuel system was redesigned with the addition of a 60-gallon tank in each nacelle (see Figure 4).
6. As the oil tanks are integral with the turbine engines, the existing oil tanks were removed and the oil system was redesigned.
7. Goodyear single-disc brakes and wheels were replaced with Goodyear double-disc brakes and wheels because of the increased gross weight.
8. The orifice systems on all landing gear shock struts were changed because of the increased gross weight.
9. The wing structure was "beefed-up" because of the increased gross weight. The lower main spar wing bolts were replaced with bolts of a higher heat-treat material, and preload indicating washers were added.
10. The flap travel was increased from 30° down to 45° down.
11. The battery installation was moved from the left-hand nacelle to the right-hand wing, just inboard of the nacelle. The external power source connection was moved from the left-hand nacelle to the underside of the right-hand wing, outboard of the nacelle.

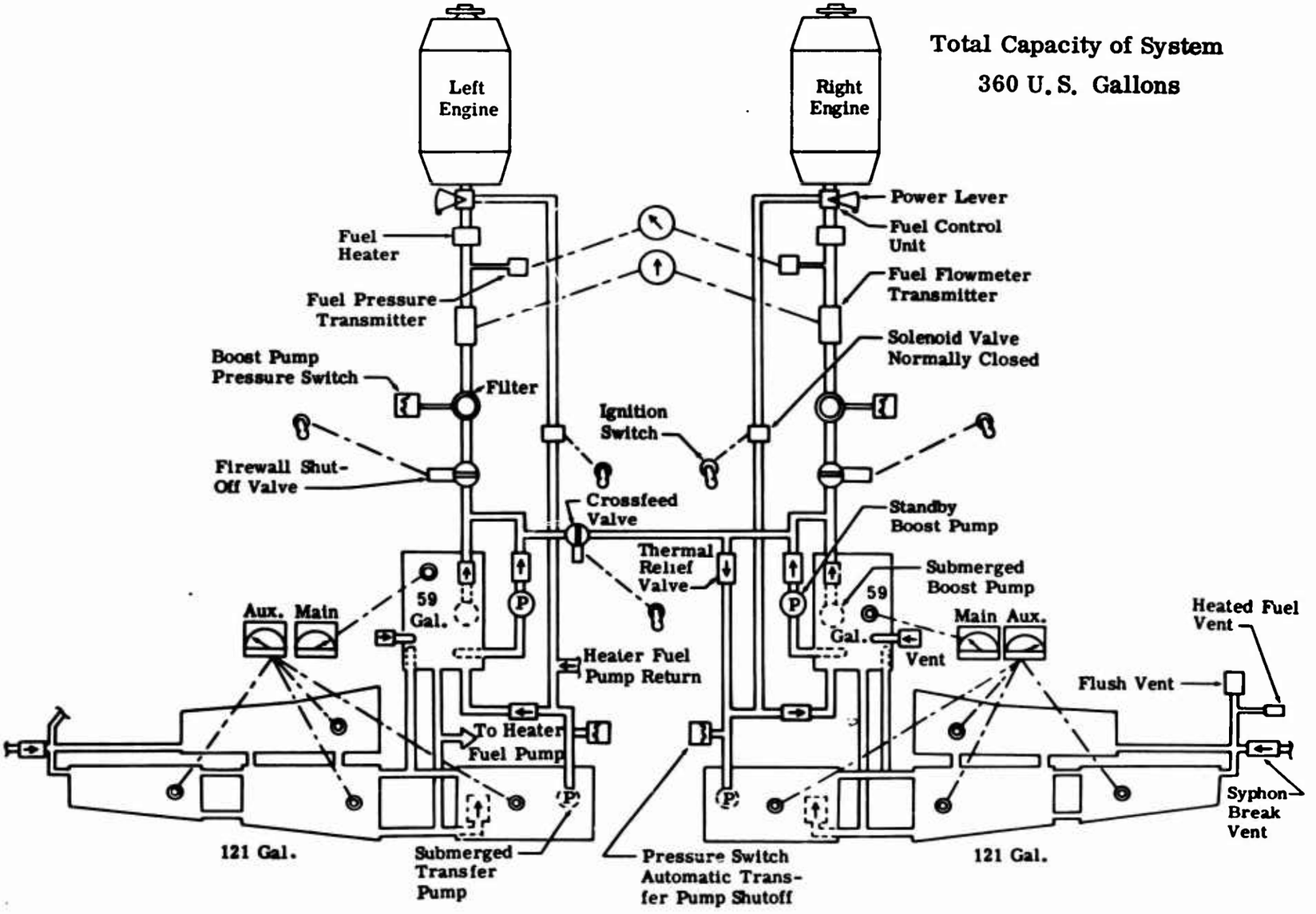


Figure 4. Schematic Diagram - Fuel System.

DISCUSSION

GROUND TEST PROGRAM

Ground Resonance Survey

A ground resonance survey was performed to determine the resonance characteristics of the NU-8F turboprop. Resonance tests were performed on three loadings, which consisted of a heavy aircraft, a light aircraft, and a light-wing configuration.

This survey in conjunction with flutter analysis work on the Model U-8F and propeller whirl mode analysis shows the aircraft to be free from flutter and potential propeller whirl modes up to and considerably above the design dive speed.

Main and Nose Gear Drop Tests

The landing gear orifice system was redesigned to keep load factors within requirements at higher gross weights. The landing gear was subjected to 230 limit drops for 5,500-, 7,700-, 8,500-, 9,500-, and 11,000-pound gross weight airplanes. Two drops were at reduced temperatures. Five reserve energy drop tests were run for an 8,500- and a 9,500-pound gross weight aircraft without permanent set or failure. Both main and nose gears as tested will accept limit landing loads and descent velocities without exceeding the design limit load factors.

Undrainable Fuel and Fuel System Functional Tests

The purpose of these tests was to determine the undrainable fuel in the entire fuel system and to obtain preliminary data on the functioning of the fuel system. The following were tested:

1. Fuel quantity gage calibration (right-hand nacelle tank only).
2. Volume of fuel transferred to the nacelle tank.
3. Flow rate of transfer pump.
4. Cycle rate of transfer pump at various engine fuel consumption rates.
5. Operation of transfer pump circuit when fuel is depleted in the wing fuel cells.

6. Fuel flow versus pressure at the inlet to the engine fuel pump (normal boost pump only).

7. Operation of boost pump circuit in the event of a normal boost pump failure.

8. Operation of the crossfeed system.

9. Level of fuel in the nacelle tanks when gravity feed from the wing tanks starts.

The system functioned satisfactorily except that there was an excessive amount of undrainable fuel in the nacelle tanks and the pressure drop in the system was excessive. Design changes were made to correct these deficiencies.

Auxiliary Tank Expansion Space

The expansion space was measured to be slightly less than 2 percent of the tank capacity..

Additional Ground Testing on the Fuel System

The configuration test was the same as the fuel system functional tests except for modifications resulting from those tests, which included:

1. Plumbing line sizes were changed from 1/2-inch to 5/8-inch tubing.

2. The primary and standby boost pumps were changed from two line-mounted boost pumps operating in parallel to one submerged tank-mounted boost pump and one line-mounted boost pump operating in parallel. Each pump had an independent fuel inlet from the nacelle tank.

3. Fuel strainer size was increased.

4. A standpipe was added in the filler neck of the nacelle tank.

The purpose of this test was to check the operation of the boost pump system, to measure the pressure drop in the fuel system, and to measure the nacelle tank capacity, expansion airspace, sump capacity, and undrainable fuel. All items tested satisfactorily except the nacelle tank expansion space. This has since been remedied.

Pressure Drop in Fuel Strainer and Fuel Flow Transmitter

The purpose of this test was to check the pressure drop in the fuel strainer and flowmeter in normal and blocked conditions. The results showed compliance with FAA regulations and the recommendations of the power plant manufacturer pertaining to fuel flow and pressure with either boost pump in operation.

FLIGHT TEST PROGRAM

Commencing with the initial flight on 15 May 1963, an experimental flight test program was initiated to evaluate the performance and characteristics of the turbine-powered NU-8F aircraft.

During this program, 132 flight hours were logged, comprising 119 individual flights on 3 sets of engines, as follows:

1. The initial test flights were conducted using PT6A-4 engines, which were the only ones available at that time. Between 15 May 1963 and 3 July 1963, 39 hours were flown with PT6A-4 engines installed to obtain preliminary data.

2. When the experimental temporarily-rated PT6A-6X engines were available, they were substituted for the PT6A-4 engines. Between 20 August 1963 and 18 November 1963, 73 hours were flown with PT6A-6X engines installed.

3. When the PT6A-6 certified engines became available, they were substituted for the temporarily-rated PT6A-6X engines. Between 24 December 1963 and 8 February 1964, 20 hours were flown with the PT6A-6 engines installed.

   The PT6A-6 engines were derated from 550 shp to 500 shp for the purpose of obtaining a minimum control speed, V_{mc}, which was desired for this aircraft.

   These engines were subsequently delivered with the aircraft.

The test data were recorded on a photo-panel, a Brown recorder, and an oscillograph.

Concurrently with these flight tests, an additional 40 hours were flown as follows:

1. Propeller synchronization - Governor modifications accomplished at the manufacturer's facilities have produced a governor configuration which is satisfactory. Approximately 11 flights were flown.

2. Icing tests - Aircraft operational capability during icing conditions (both artificially induced and natural) was tested by the U. S. Army during the period from 20 March 1964 to 23 March 1964. All deicing systems functioned satisfactorily except the engine alcohol anti-ice system, which has since undergone extensive modification and is currently the same type used in the commercial counterpart of the NU-8F aircraft.

3. Demonstration - Approximately 15 flights were made for the purpose of demonstrating the airplane to Beech and Military personnel.

4. Photographic flights - 2 flights were flown.

5. Sound level testing - 4 flights were flown.

6. Shakedown and familiarization - 5 flights were flown.

Problems encountered during the flight test program were (a) directional trim, (b) rudder pedal forces with one engine out, and (c) stall characteristics.

1. Directional trim - The directional trim inadequacy was solved by the installation of a new rudder trim tab which was longer in span, shorter in chord, and larger in area.

2. Rudder pedal forces - The rudder pedal forces were reduced by lowering the tension on the rudder return springs. This was made possible by the addition of a ventral fin which decreased the angle of yaw and thus reduced rudder lock tendencies.

   Other configurations which were tested, and which proved to be ineffective in reducing rudder pedal forces, were (a) vortex generators on fin and rudder to change the center of pressure, (b) an increased area dorsal fin to reduce the angle of yaw, and (c) a modified rudder leading edge.

3. Stall characteristics - The stall characteristics with power on or off, rear cg, and gear and flaps down are marginal by FAA standards.

Aerodynamically, the aircraft meets all the requirements applicable to the NU-8F except the above-mentioned stalls. The stall characteristics are docile to the extent that this noncompliance is considered to be safe and satisfactory without any precautionary qualification being made in the aircraft flight manual.

In addition to the flight tests to measure performance, considerable testing was performed on aerodynamic cleanliness. The aircraft was flown with flush exhaust stacks, flush heater exhaust stacks, flush landing light lenses, flush screws in critical areas, and metal wing tips. No performance gain was attributed to these items. The aircraft was then flown with antennas removed to simulate flush antennas; this produced a 2-mph increase in air speed.

PERFORMANCE*

Take-Off and Initial Climb

Take-off distance over a 50-foot obstacle was established using take-off power (500 shp), maximum aircraft gross weight (9,000 pounds), and wing flaps retracted (up). Normal take-off technique is to accelerate to 115 mph CAS (both engines developing take-off power), to lift off and immediately retract the landing gear, and then to climb out at 115 mph CAS. Tests were also conducted for a single-engine take-off (with rotation and assumed engine failure at 115 mph) and normal take-off with a rotation at 90 mph. These tests and the resultant take-off distances are all based on sea-level standard, no-wind conditions, and are shown in Table II, Figure 6, and Figure 7.

Ground rolls obtained by rotating at 90 mph CAS were less than the normal take-off configuration; however, total distances to 50 feet were about the same. Therefore, rotating before normal take-off speed is not necessary.

*Gross take-off weights are shown in the following performance data at the maximum aircraft gross take-off weight of 9,000 pounds, and at various reduced weights. These weight variations are shown to illustrate comparative performance characteristics and the various weight valuse used during aircraft flight testing. Center-of-gravity limitations for maximum and reduced weight conditions are shown in Figure 5, page 27.

TABLE II
TAKE-OFF PERFORMANCE

Take-Off	Gross Weight (lb)	Take-Off CAS (mph)	Take-Off Ground Dist (ft)	CAS at 50 Ft (mph)	Total Dist to 50 Ft (ft)
Normal	8700	115	1565	115	1800
Normal	9000	115	1640	115	1890
Single-Engine(1)	9000	115	1640(2)	115	2500(3)
Rotation at 90 mph	9000	90	920	115	1900

(1) Single-engine operation simulated at 115 mph (safe single-engine speed).

(2) Acceleration distance assumed the same as normal take-off.

(3) See Figure 8. Flight test CAS at 50 feet was approximately 140 mph CAS, giving a total distance at 50 feet of 3675 feet.

Accelerate-Stop

Testing was accomplished to determine the critical engine failure speed for an aborted take-off. Initial ground run procedure is the same as for take-off procedure: accelerate with both engines developing take-off power to accelerate-and-stop decision speed of 115 mph CAS. If engine failure occurs at or before "safe single engine speed" is attained, 115 mph CAS, abort the take-off by closing power levers, feather both propellers, and immediately apply maximum braking. The accelerate-stop distance corrected to standard-day (ISA), sea-level, no-wind conditions is shown in Table III and Figure 9.

TABLE III
ACCELERATE-STOP DISTANCE

Gross Weight (lb)	Ground Roll (ft)	Inertia-Decel Dist (ft)	Total Dist (ft)
9000	1640*	1625	3265
8700	1585	1620	3205

*Acceleration distance assumed the same as for normal take-off.

Climb

The airspeed climb schedule for two-engine and single-engine climbs is shown in Table IV, Figure 10, and Figure 11. These speeds were determined from sawtooth climbs at maximum climb power with gear and flaps up.

TABLE IV
BEST CLIMB SPEED

Altitude (ft)	Two-Engine CAS (mph)	Single-Engine CAS (mph)
SL	130.0	122.5
5000	128.5	120.5
10000	127.0	118.5
15000	125.5	-
20000	124.0	-

The best climb speed obtained during flight testing established a rather steep angle of ascent relative to passenger comfort. For normal operation, a higher speed would be used (as selected by the pilot) with a corresponding compensation being made for the reduction in rate of climb and time to climb to altitude. Rate of climb can be determined for a higher climb speed from Figure 12. Two-engine and single-engine climb data are shown in Tables V and VI and in Figures 12 through 15.

The climb performance meets certification requirements. Climb gradients for both two-engine and single-engine operations at 5000 feet, ISA + 40°F, were above certification requirements.

TABLE V
TWO-ENGINE CLIMB PERFORMANCE

Altitude (ft)	Rate of Climb (ft/min)	SHP/Eng	Fuel Flow/Eng (lb/hr)	CAS (mph)
SL	2000	500 (limit)	315	130.0
5000	1800	468	284	128.5
10000	1420	413	246	127.0
15000	1040	363	212	125.5
20000	660	315	180	124.0
25000	280	270	152	122.4
27400 (S. C.)	100	245	140	121.5

TABLE VI
SINGLE-ENGINE CLIMB PERFORMANCE

Altitude (ft)	Rate of Climb (ft/min)	SHP/Eng	Fuel Flow/Eng (lb/hr)	CAS (mph)
SL	535	500 (limit)	315	122.5
5000	408	470	284	120.5
10000	183	415	246	118.5
12950 (S.C.)	50	382	225	117.5

Level Flight

Level flight performance data were obtained at 5000, 10,000, 20,000 and 25,000 feet. Data were corrected to standard sea level condition at a mid-range weight of 7900 pounds and as shown in Figure 16.

Additional level flight performance tests were conducted to evaluate an exhaust stack configuration for the Pratt and Whitney PT6A-6 production engines. Results of this testing are presented in Figure 17. Length of exhaust stack had no significant effect on level flight performance. The longer exhaust stacks were selected as the standard exhaust configuration. The tests show the airplane to be 8 mph faster with the PT6A-6 engines than with the PT6A-6X engines.

The maximum speed of the airplane with PT6A-6 engines at maximum climb power is 276 mph TAS at 15,000 feet. The high speed condition at maximum cruise power (NRP) is 269 mph TAS at 15,000 feet.

Specific range data are based on speed-power data obtained with the PT6A-6X engines. A summary of level flight performance is given in Table VII.

Airplane drag polars for climb, cruise, and landing are shown in Figures 18 through 21.

Engine Calibration

Engine calibration checks were conducted for the Pratt and Whitney PT6A-6X and PT6A-6 engines. These checks were made concurrently with climb and speed-power tests. Plots of nondimensionalized exhaust gas temperature (EGT/θ a), fuel flow ($W_f/\delta_2\sqrt{\theta_2}$), and gas generator speed ($N_1\sqrt{\theta_2}$)/DPS versus shaft horsepower ($SHP_{Opt}/\delta_2\sqrt{\theta_2}$) are presented on Figures 22 and 23.

TABLE VII
LEVEL FLIGHT PERFORMANCE

Altitude (ft)	Max TAS (MCP) (1) (mph)	Max Cruise TAS (NRP) (mph) (1)	Recommended Cruise - TAS (mph) (2)	Air Miles / Lb of Fuel Cruise (2)	Cruise SHP/Eng (2)
5000	259 (CAS limit)	259 (CAS limit)	182	.5430	200
10000	276	269	188	.6200	200
15000	276	269	193	.6875	200
20000	273	266	196	.7590	200
25000	264	258	197	.8100	200

(1) Based on PT6A-6 engine data.
(2) Based on PT6A-6X engine data.

Descent

Rate-of-descent tests were conducted to determine an optimum descent configuration. Descents were made in the clean configuration at idle power at V_{mo} (240 mph CAS) and at a cruise speed of about 200 mph CAS. These were accomplished at a constant 1000 and 500 fpm rate of descent. The descents were initiated from pressure altitudes of 22,500 feet and 17,000 feet.

Stall Speeds

Power-off stall speeds are shown in Table VIII.

TABLE VIII
POWER-OFF STALL SPEEDS

Gross Weight (lb)	CG Position (percent)	Power (Prop Windmilling)	Flap Setting	Gear Position	V_{S_1} CAS (mph)
9000	20.8	Off	100%	Down	90.0
8700	20.8	Off	100%	Down	88.5
9000	20.8	Off	Up	Up	105.4
8700	20.8	Off	Up	Up	104.0
9000	29.9	Off	100%	Down	87.0
9000	29.9	Off	Up	Up	103.5
7750	16.0	Off	100%	Down	82.0
7750	16.0	Off	Up	Up	97.0

Stall speeds obtained at the forward center-of-gravity position (9000 pounds, 20.8 percent) are listed in Table IX.

TABLE IX
FORWARD CG STALL SPEEDS

Configuration				V_{S_1} (Pilot)	V_{S_1} (Boom)
Power	Propeller	Flaps	Gear	CAS (mph)	CAS (mph)
On	Oper	Up	Up	89.9	86.9
Off	WM	Up	Up	105.8	105.4
On	Oper	50%	Down	73.6(IAS)	74.5
Off	WM	50%	Down	91.2(IAS)	92.1
On	Oper	100%	Down	73.4	78.3
Off	WM	100%	Down	86.5	90.0
On	Lt - WM Rt - Oper	Up	Down	110.5	105.2
On	Lt - Fea Rt - Oper	Up	Down	105.2	109.1
Off	Fea	100%	Down	93.8	88.6

Zero thrust stall speeds are as shown in Table X.

TABLE X
ZERO THRUST STALL SPEEDS

Configuration		V_{S_1} (Boom)	Extended Leading Edge V_{S_1} (Boom)
Flaps	Gear	CAS (mph)	CAS (mph)
100%	Down	86.5	79.9
Up	Up	105.0	97.4

Landing

Landing technique over a 50-foot obstacle is to reduce power to maintain an 800-fpm rate of descent. Wing flap setting is 100 percent (43-1/2 degrees) and approach speed is 115 mph CAS. All landing test data are based on sea-level standard no-wing conditions at the design maximum gross landing weight of 8550 pounds. Refer to Figure 24 and Table XI for landing performance results.

TABLE XI
LANDING DISTANCES

Gross Weight (lb)	CAS at Touchdown (mph)	Ground Roll (ft)	CAS at 50 Feet (mph)	Total Distance From 50 Feet (ft)
8550	100	1280	115	2240
8265	100	1240	115	2170

Aircraft landing characteristics are normal, with satisfactory landing distance and ground roll. However, these particular aspects of a normal landing are largely dependent upon pilot technique, approach speed, wing flap usage, landing surface condition, and wheel brake usage.

Airspeed Calibration

Airspeed calibration for test purposes was determined using pace aircraft and the speed course method. Except as noted, all airspeeds given in this report are calibrated airspeed (CAS). Indicated airspeeds (IAS) used for data reduction were airspeed test boom readings. Airspeed position error correction for the standard system is considered satisfactory throughout the entire aircraft speed range.

Airspeed calibration data are presented in Figures 25 and 26.

Temperature Probe Calibration

Temperature probe calibrations to determine engine inlet adiabatic heat rise are shown in Figures 27 and 28.

STABILITY AND CONTROL

Flight Control System

Control friction tests were conducted on the elevator, aileron, and rudder. Additional friction tests were made on the rudder control with the nose wheel steering disconnected, with and without the rudder return spring connected. The breakout forces on all controls are high as compared to the maximum allowable limits in Specification MIL-F-8785 (ASG), as shown in Table XII.

TABLE XII
FLIGHT CONTROL FORCES

Control	Breakout Force (lb)	MIL-F-8785 Limit (lb)
Elevator	7-1/2	7
Aileron	5	6
Rudder (Standard)	35	14

Elevator Control

Take-offs and landings were made at the most forward center of gravity (cg) to determine elevator effectiveness. A cg of 15.9 percent mean aerodynamic chord (MAC) was used for take-off and a cg of 15.4 percent MAC for landing. Time histories of these tests can be found on Figures 29, 30, and 31. Elevator control during these tests was considered satisfactory.

Static Longitudinal Stability

Tests were accomplished at both forward (16.0 percent MAC at 7750 pounds) and aft (29.9 percent MAC at 9000 pounds) center of gravity positions. Static longitudinal stability was positive for all configurations tested (climb, cruise, glide, and landing) and is in accordance with the FAA requirements. Static longitudinal data are presented in Figures 32 through 35.

The configurations used throughout this report are as follows:

Climb	Flaps up, gear up, power on
Cruise	Flaps up, gear up, power on
Glide	Flaps up, gear up, power off
Landing	Flaps down, gear down, power off
Go-Around	Flaps down, gear down, power on

Dynamic Longitudinal Stability

Dynamic oscillations were made with a phugoid in climb, cruise, and landing configurations. Short-period oscillations were accomplished for stick-fixed and stick-free conditions in cruise, glide, and landing configurations.

The climb phugoid oscillation was initiated by reducing the airspeed approximately 20 mph. The damping of the oscillation in the climb and

cruise configuration was light but stable. The damping of the oscillation in the landing configuration was heavier. In the landing configuration, it took about three cycles (80 seconds) to reduce the oscillation 50 percent.

Dynamic oscillations induced by sudden movement of the control wheel for stick-fixed and stick-free configurations were heavily damped for both positive and negative changes in normal acceleration. For all configurations, oscillations were damped within one cycle (approximately one second). Time histories of the data are presented on Figures 36 through 44.

Static Directional Stability

The static directional stability, as determined from sideslip test, was considered satisfactory in all configurations tested. Sideslip data are presented in Figures 45 through 52.

During initial testing, rudder lock was experienced in the go-around configuration. Tests were conducted during the final phase to reduce rudder forces which led to the installation of a ventral fin. With the ventral fin installed, no rudder lock was experienced. Results of rudder lock tests are presented on Figures 52 through 56. The maximum rudder pedal force recorded was 120 pounds during the landing configuration, which is below the 150-pound limit specified by the FAA.

Dynamic Lateral-Directional Stability

Stick-free dynamic lateral-directional damping meets the FAA requirements. Time histories of rudder deflection, both right and left, for cruise, glide, go-around, and landing configurations are presented in Figures 58 through 66. Oscillations induced by sudden application of rudder control are damped in approximately four cycles. Oscillations induced by bumping the rudder pedal were dampened within two cycles.

Time histories of aileron deflections, both right and left, for cruise, go-around, glide, and landing configurations are shown in Figures 67 through 70.

Lateral Control

Aileron rolls were conducted to determine aileron control effectiveness. Both left and right deflections in cruise, glide, and landing configurations were tested. No dangerous or adverse characteristics were encountered. Lateral control for rolling performance is presented in Figures 71 through 76.

Single-Engine Characteristics

Directional control with asymmetric power was determined at a center of gravity of 30.3 percent MAC. Effects of power on rudder power and minimum control speed (V_{mc}) are summarized in Table XIII and are presented on Figures 77, 78, and 79.

TABLE XIII
SINGLE-ENGINE CHARACTERISTICS

Right Engine	Left Engine	Power (shp)	CAS (mph)	Rudder Position (deg)	Rudder Force (lb)	Bank (deg)
Oper	Windmill	500	106.0	-	-	5
Oper	Windmill	500	115.4	-	-	0
Oper	Feather	500	106.6	24	118.0	0
Oper	Feather	450	103.4	24	113.0	0
Oper	Feather	400	101.4	23	106.5	0

From these results, V_{mc} (106 mph CAS) meets the FAA requirements.

A time history of a simulated left-engine failure is presented in Figure 78.

The effects of rudder trim on minimum directional trim and effects of rudder tab on V_{mc} are presented in Figures 80 and 81.

Stall Characteristics

Original prototype NU-8F aircraft configuration stall characteristics were undesirable within the scope of FAA Certification requirements in that excessive left roll occurred (FAA maximum allowable is 15 degrees) during landing configuration stalls with either power on or off. Power-on stalls in general exhibited the same tendency but to a lesser degree. This condition, however, has since been corrected to meet FAA requirements specifically in the subject aircraft and in the commercial counterpart; currently, there are no unfavorable stall characteristics.

Stall time histories are presented in Figures 82 through 85 and summarized in Table XIV.

TABLE XIV
STALL CHARACTERISTICS

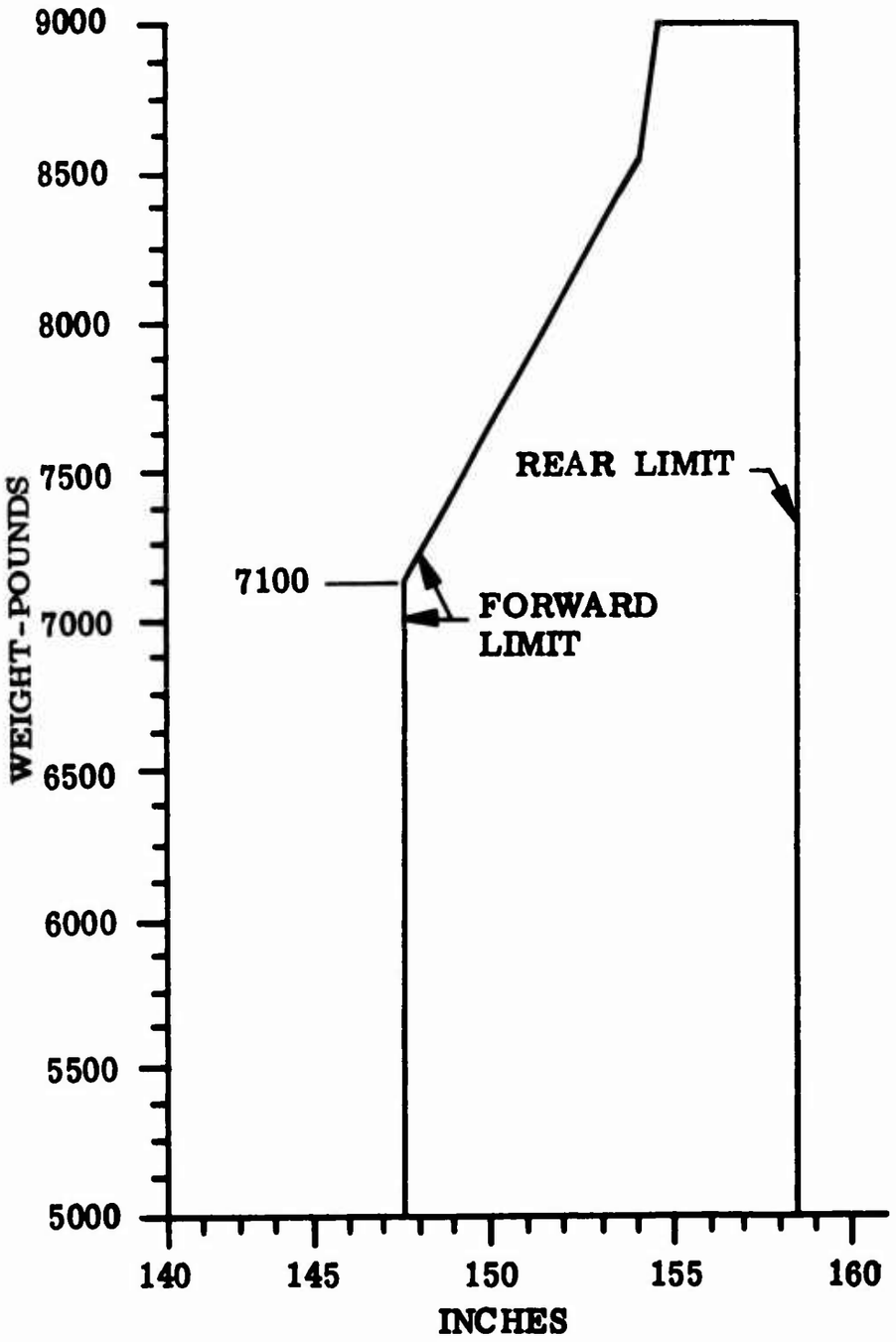
Configuration			Calibrated Airspeed (mph)	Gross Weight (lb)	CG (percent MAC)
Power (shp)	Gear	Flaps (percent)			
Idle	Up	0	115.0	8909	30.3
369	Up	0	73.0	8894	30.3
Idle	Down	100	106.2	8873	30.2
384	Down	100	60.0	8860	30.2

High-Speed Dive

A time history of a high-speed dive is shown in Figure 86. This test was conducted to check flight characteristics at high speeds. The dive was initiated at an altitude of 12,500 feet. A maximum indicated airspeed of 310 mph was obtained at a normal acceleration of 1.25g.

Elevator Tab

The effect of the elevator tab deflection on minimum longitudinal speed is shown on Figure 87.



The NU-8F forward CG limit is 16.0 percent MAC at 7100 pounds or less, progressing uniformly from 16.0 to 24.4 percent MAC from 7100 to 8550 pounds. The forward CG limit is 24.4 percent MAC at 8550 pounds and progresses uniformly to 25.0 percent MAC at 9000 pounds. The aft CG limit is 29.9 percent MAC at all weights.

Figure 5. Center-of-Gravity Range Envelope.

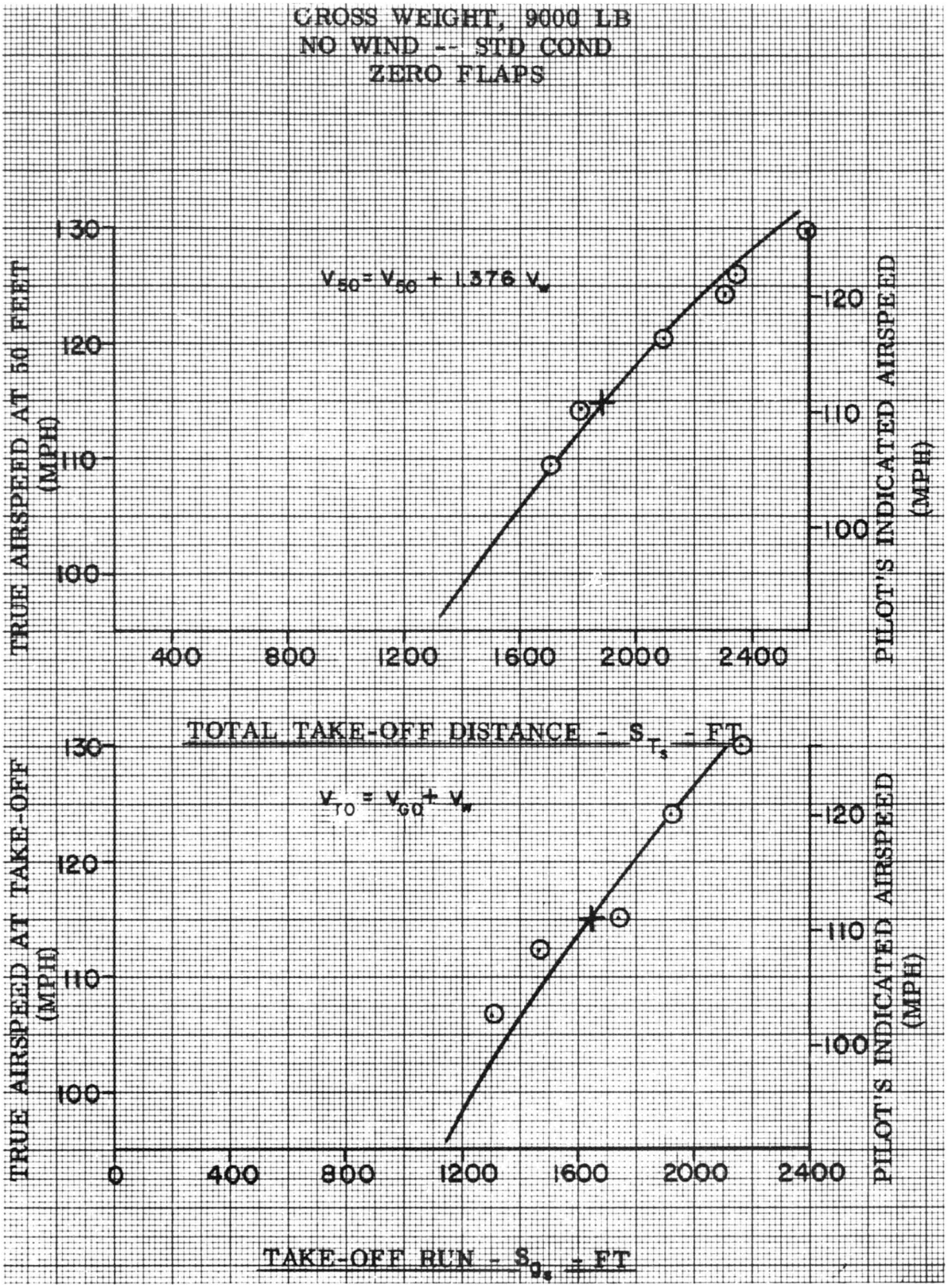


Figure 6. FAA Take-Off Distance.

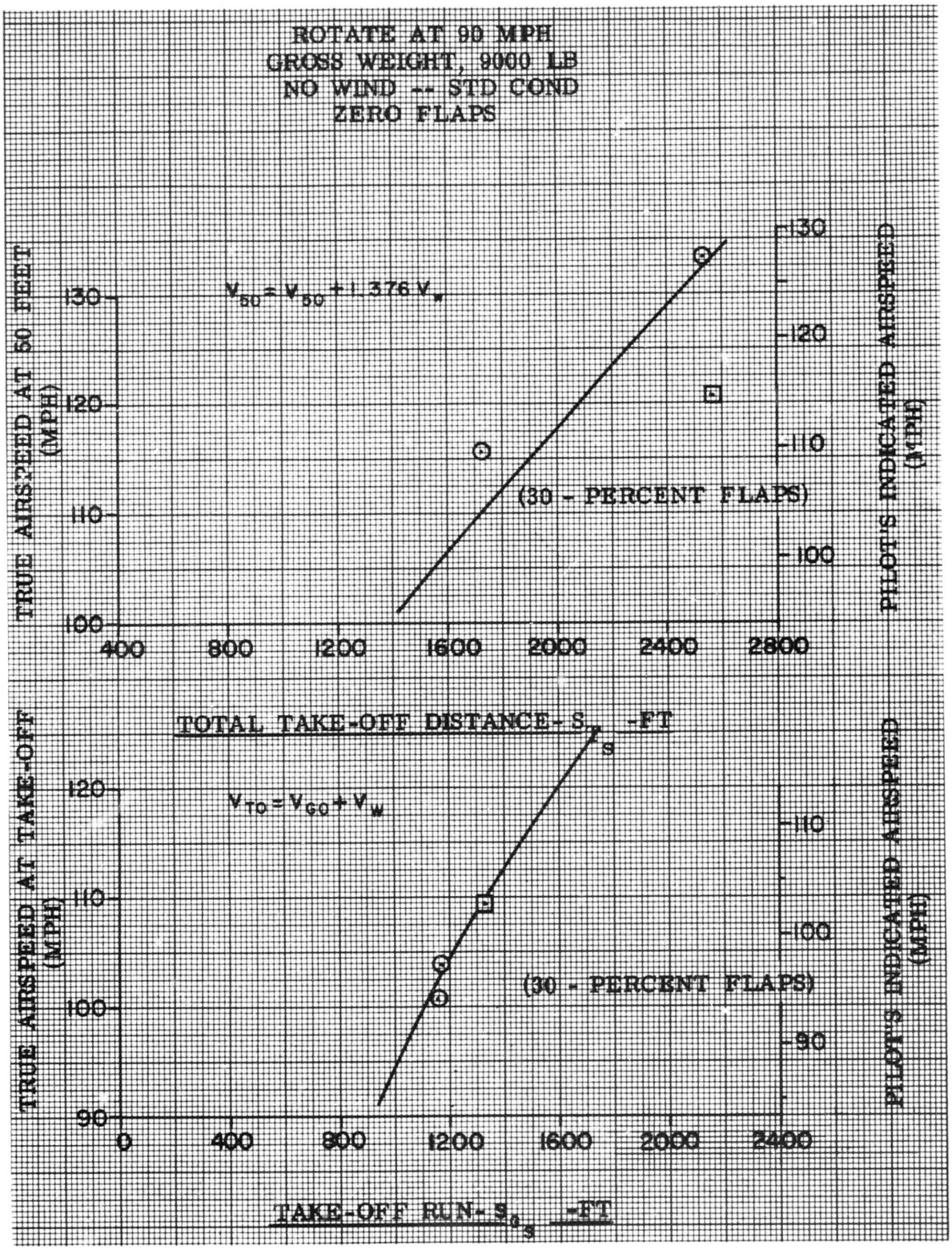


Figure 7. Take-Off Distance.

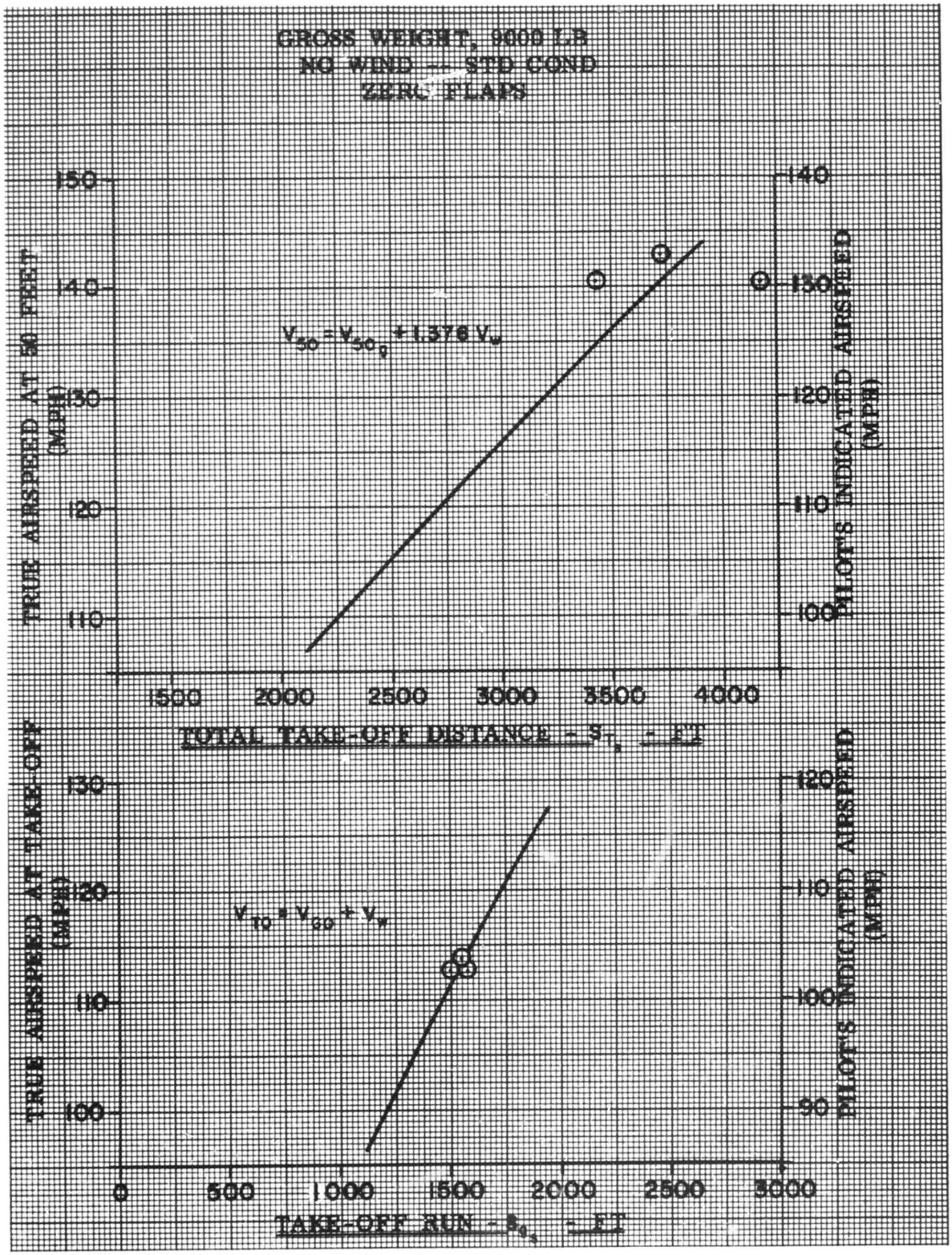


Figure 8. Single-Engine Take-Off.

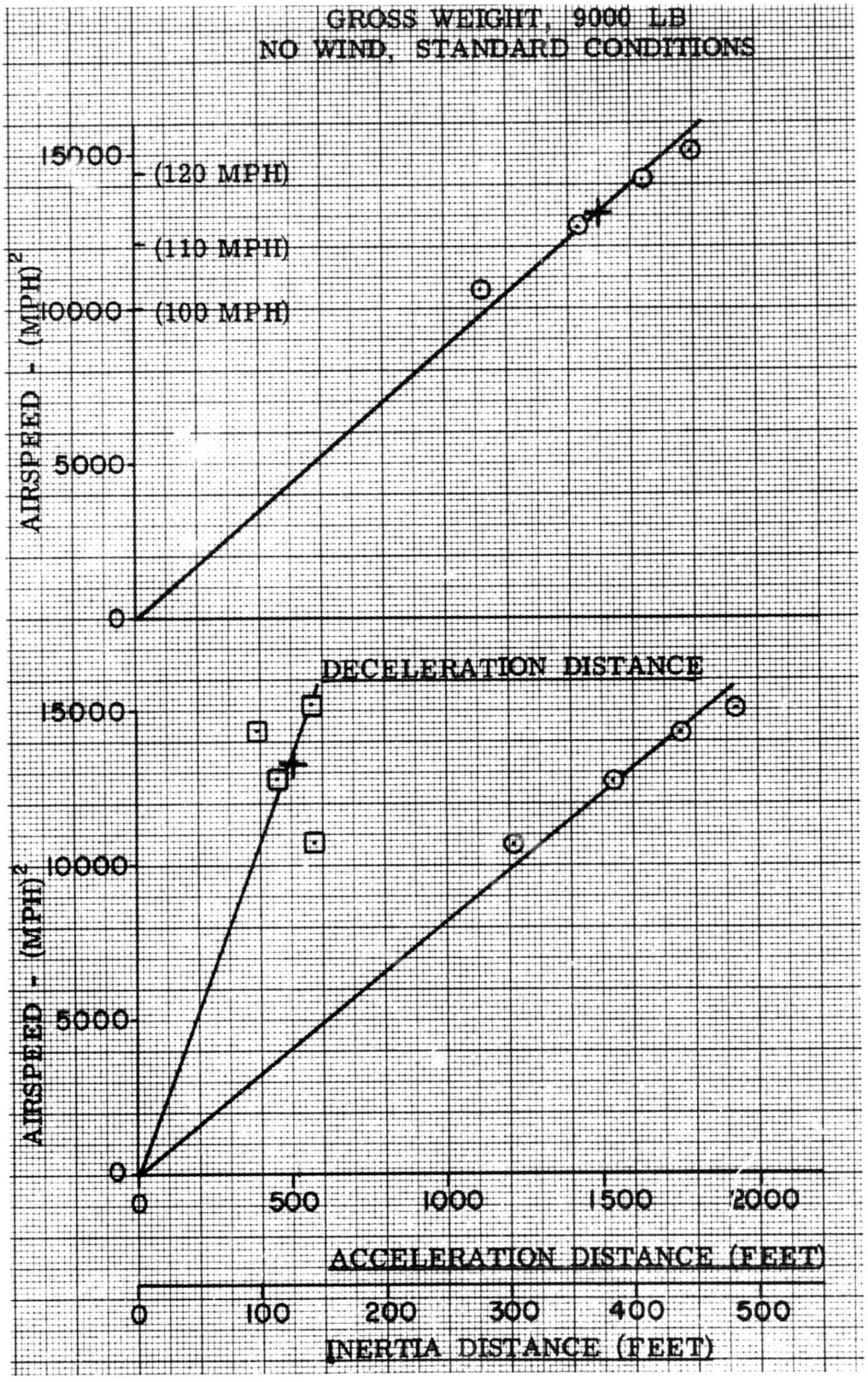


Figure 9. Accelerate and Stop Distance.

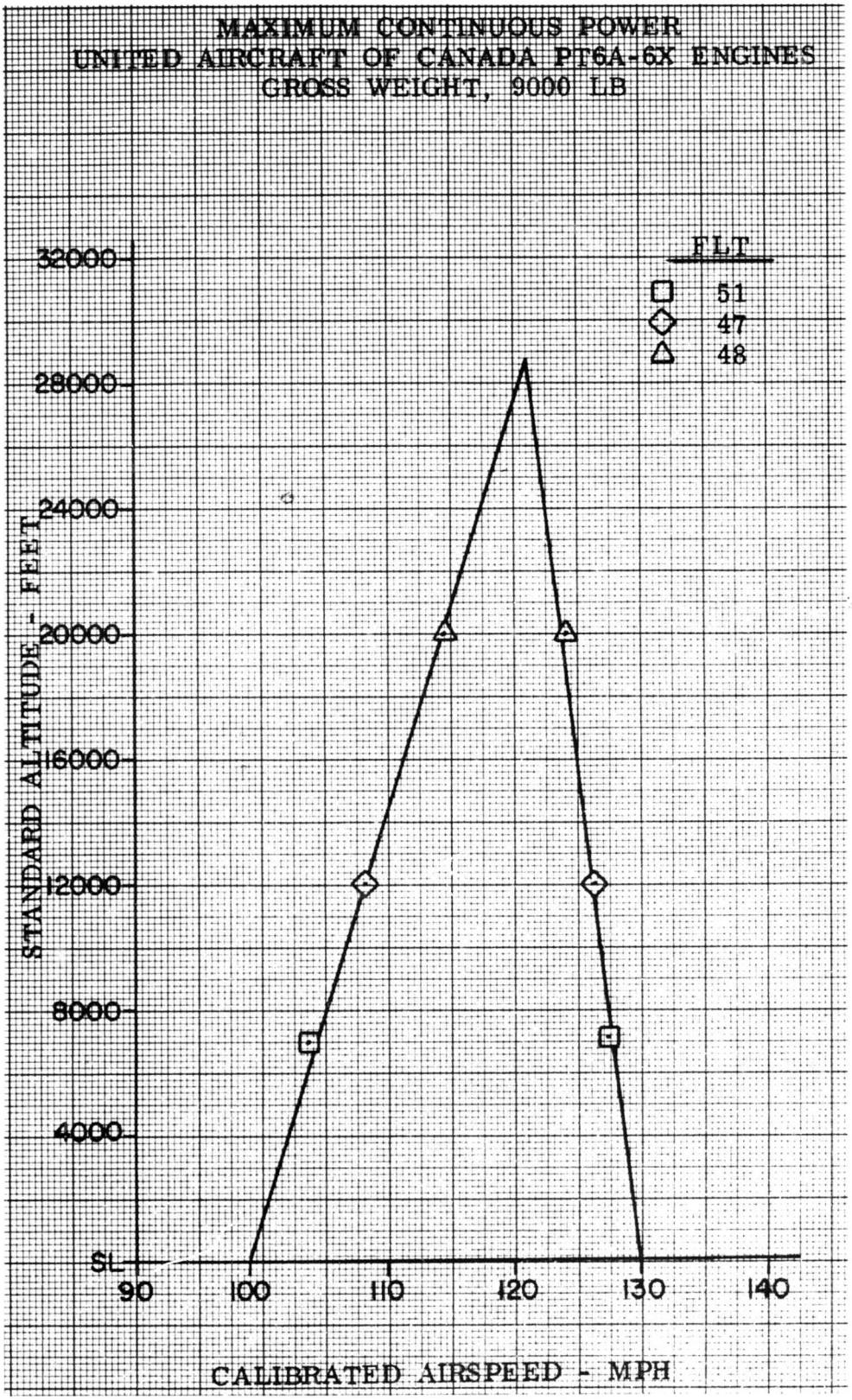


Figure 10. Normal Climb Speed.

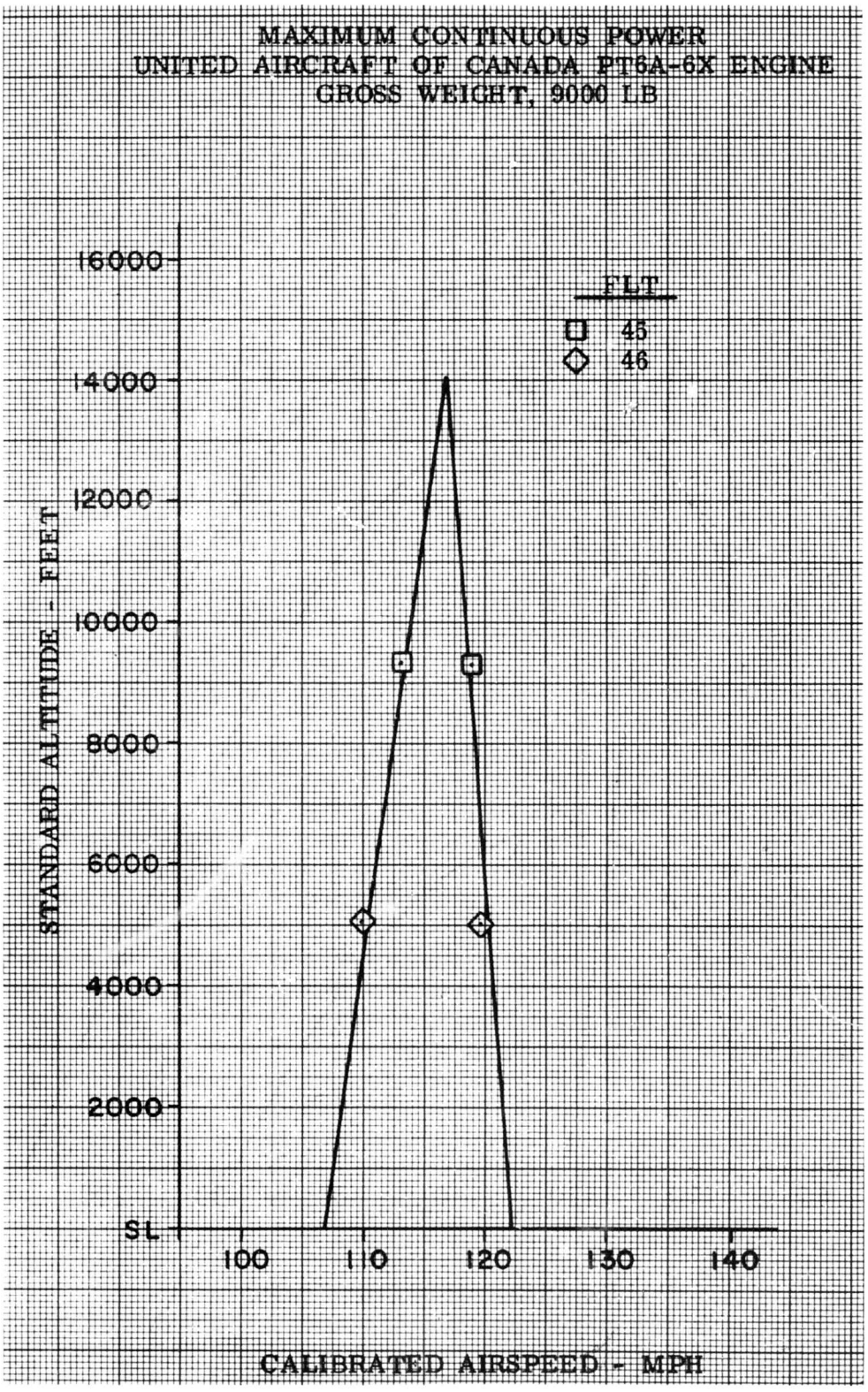


Figure 11. Single-Engine Climb Speed.

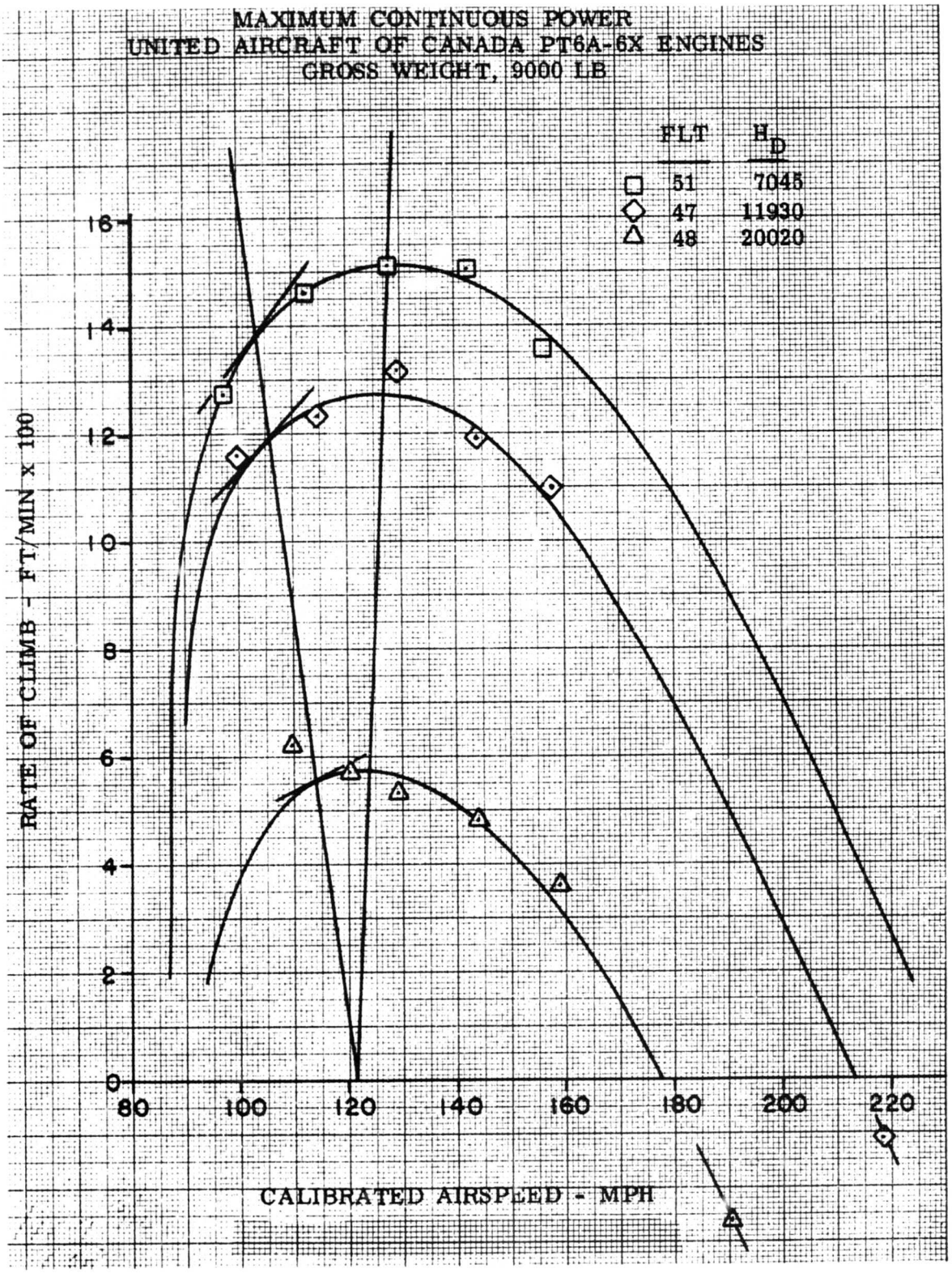


Figure 12. Normal Saw-Tooth Climb.

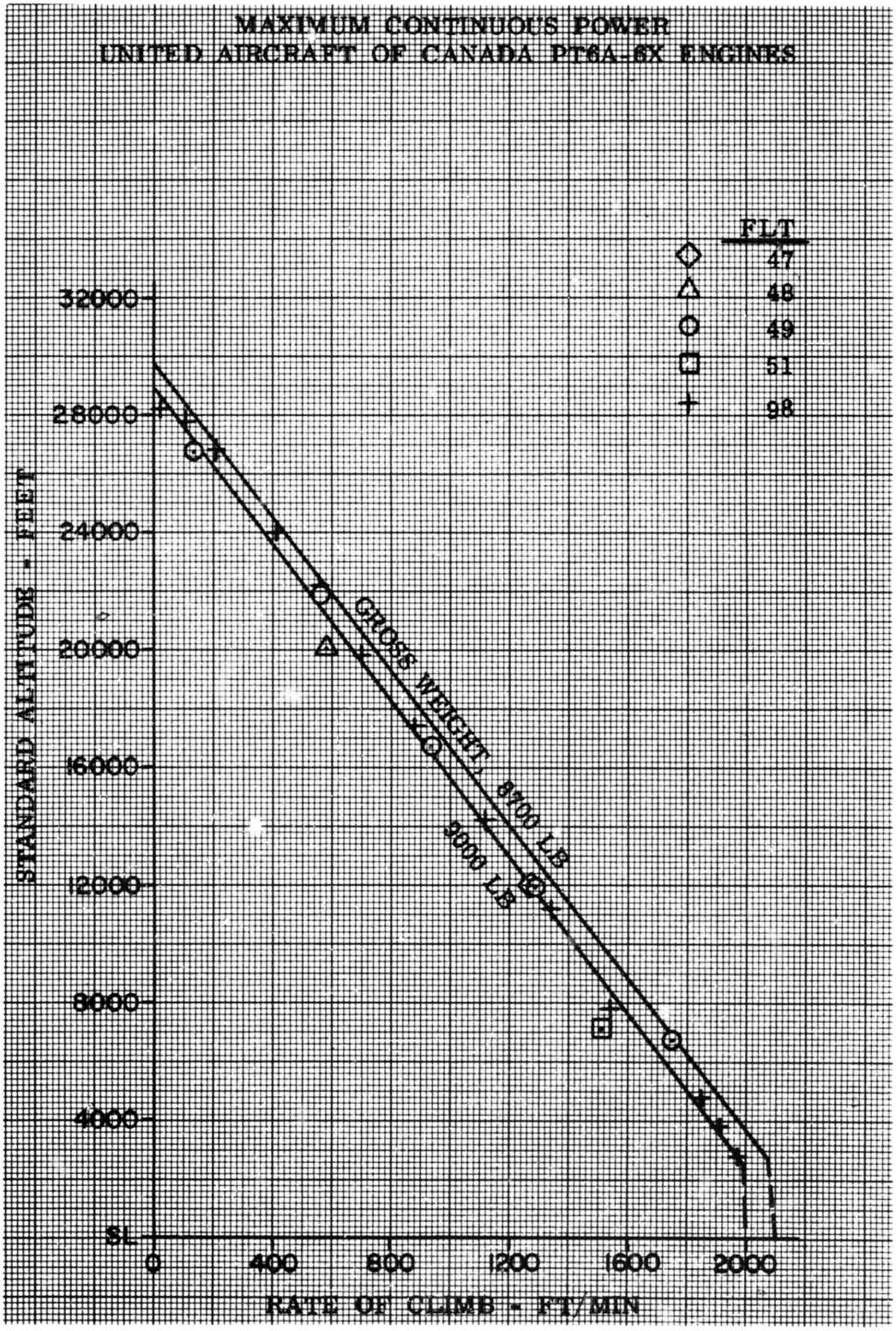


Figure 13. Normal Climb.

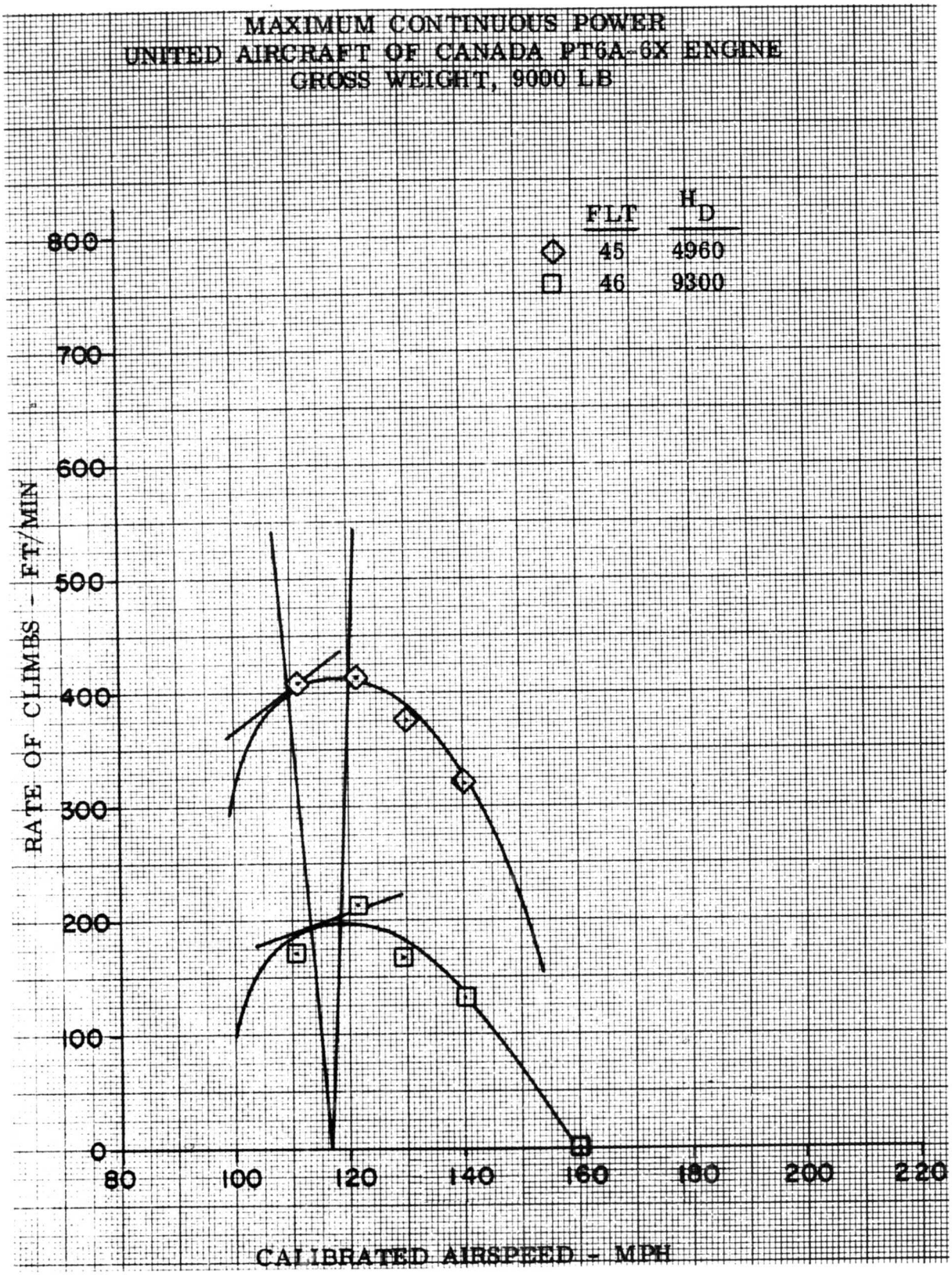


Figure 14. Single-Engine Saw-Tooth Climb.

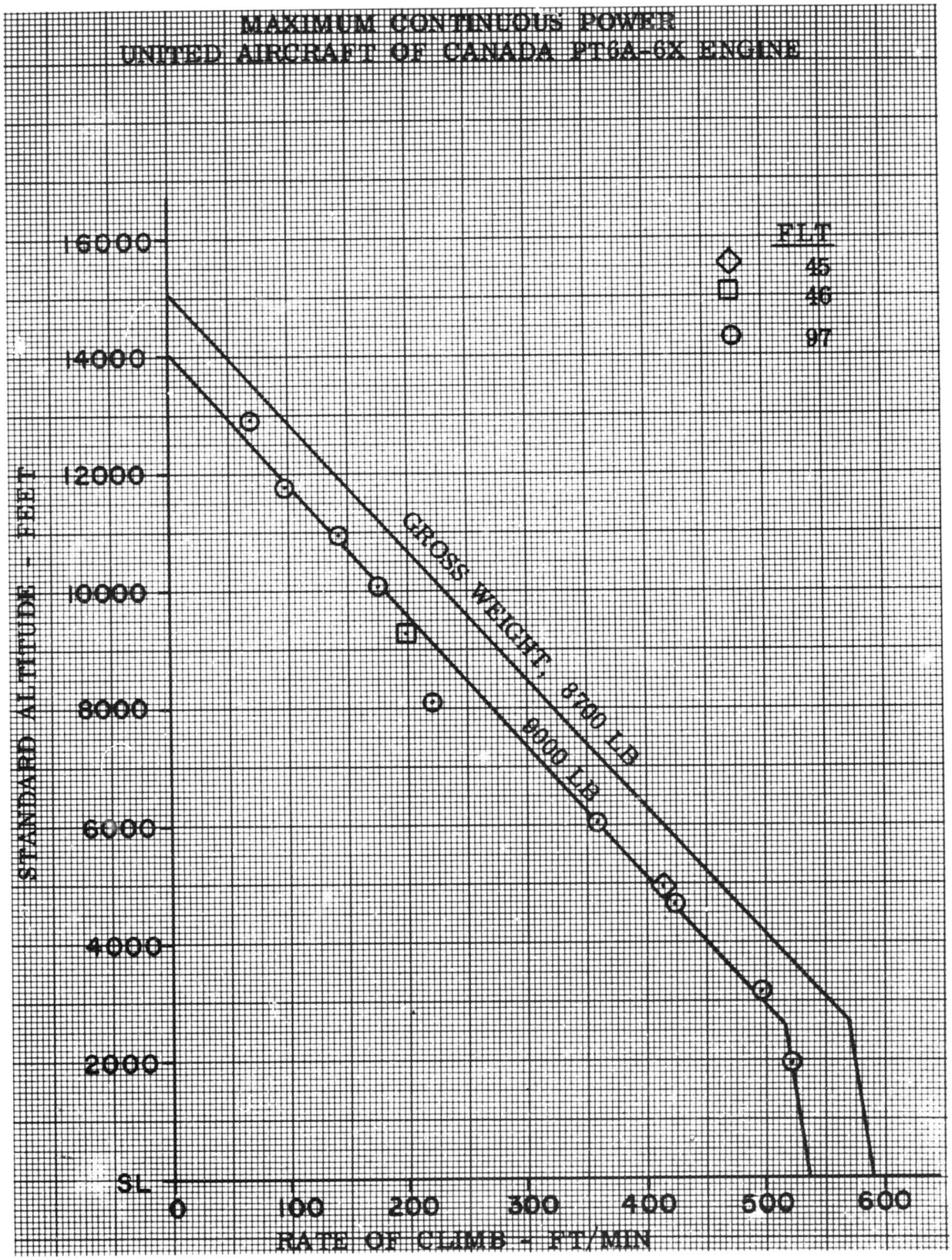


Figure 15. Single-Engine Climb Performance.

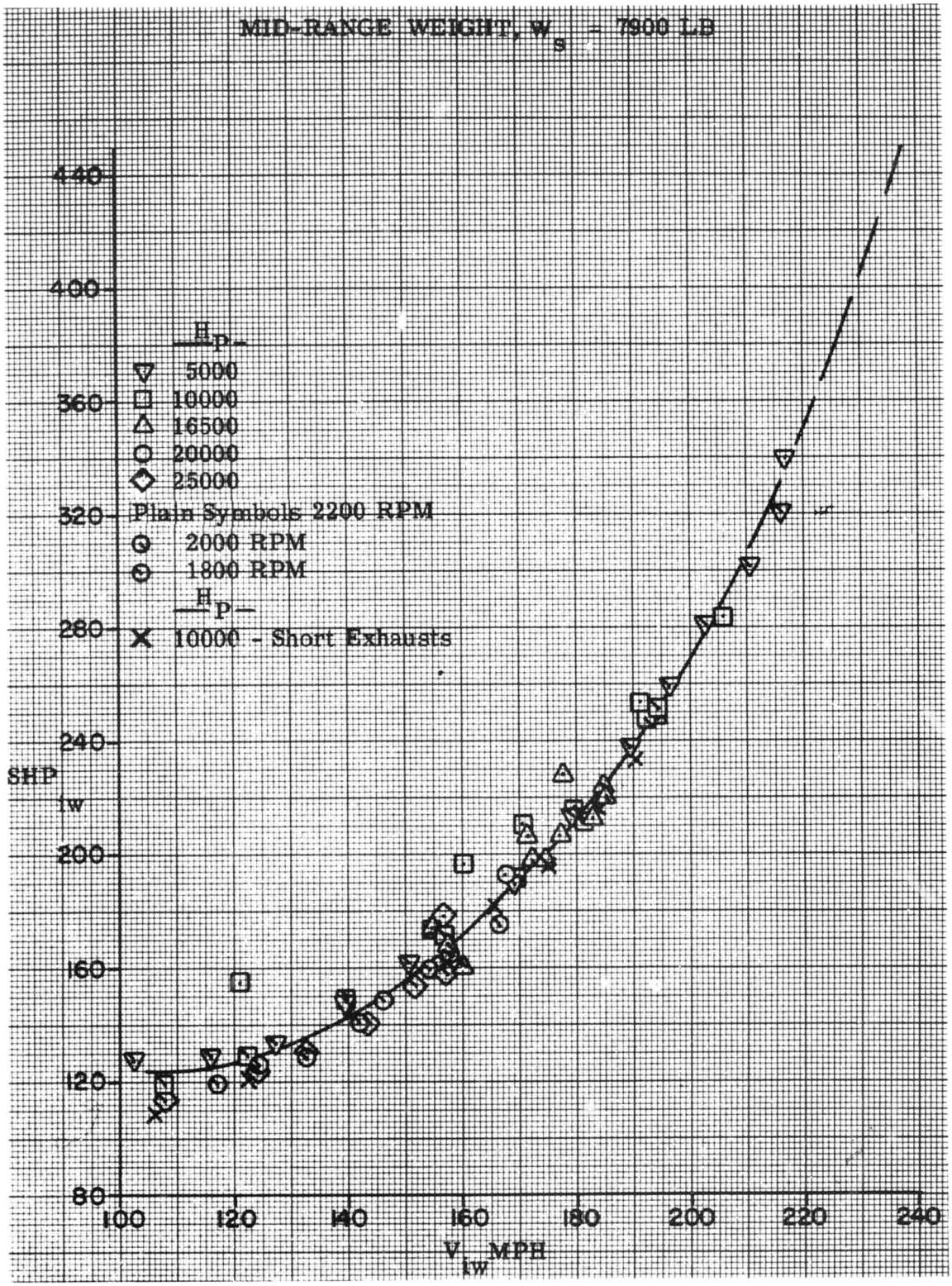


Figure 16. Universal Speed-Power - PT6A-6X Engines.

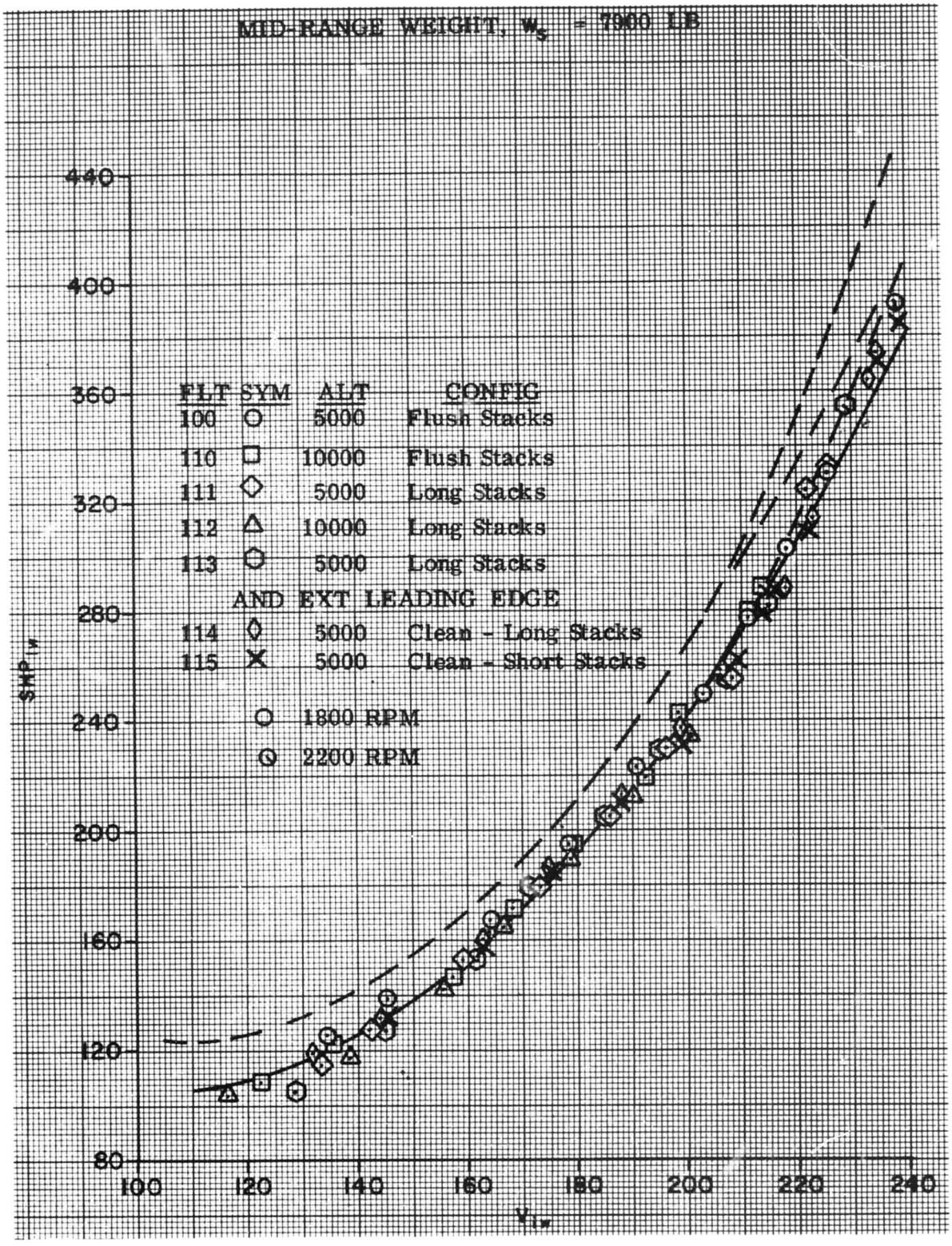


Figure 17. Universal Speed-Power - PT6A-6 Engines.

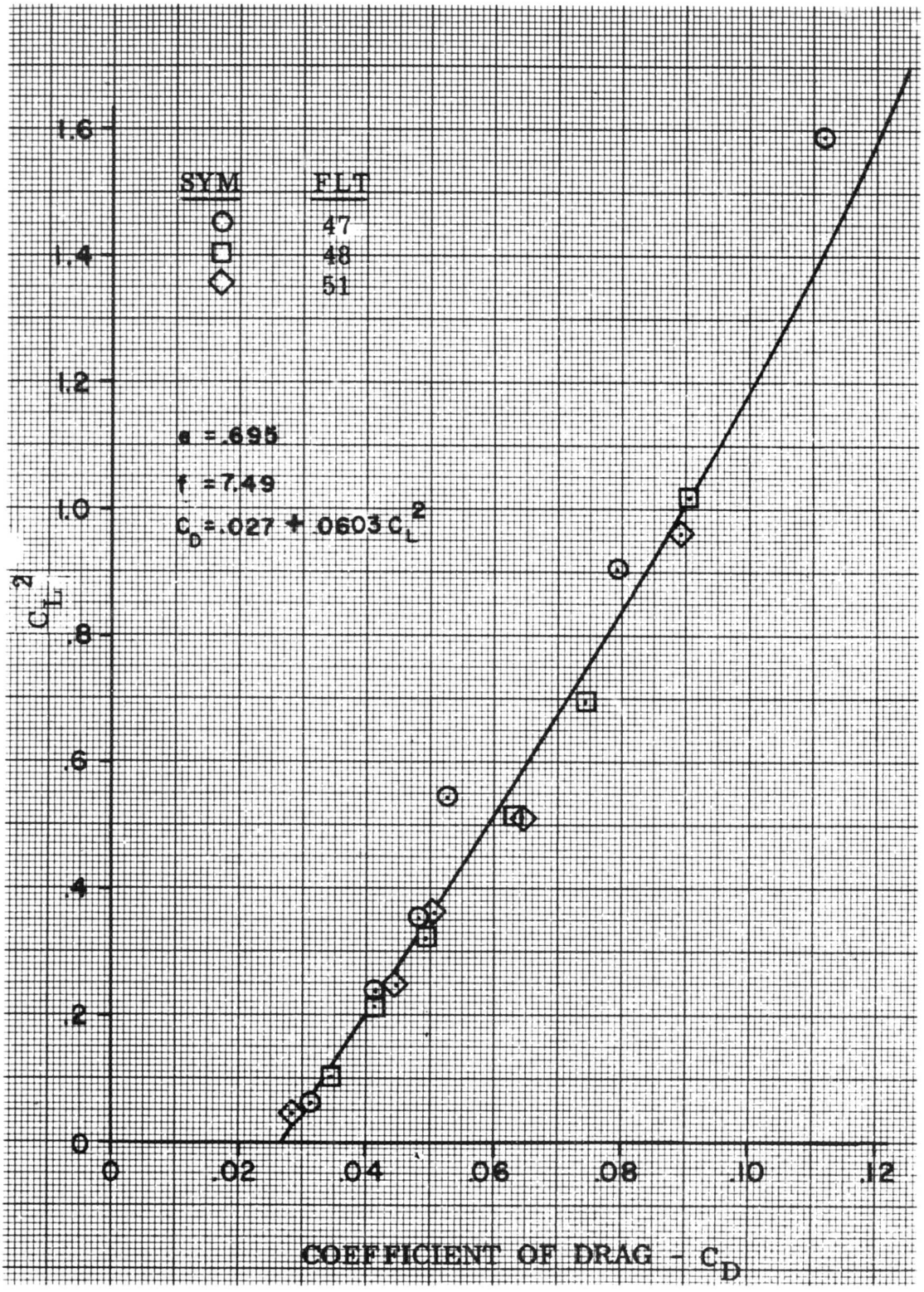


Figure 18. Drag Polar - Normal Climb.

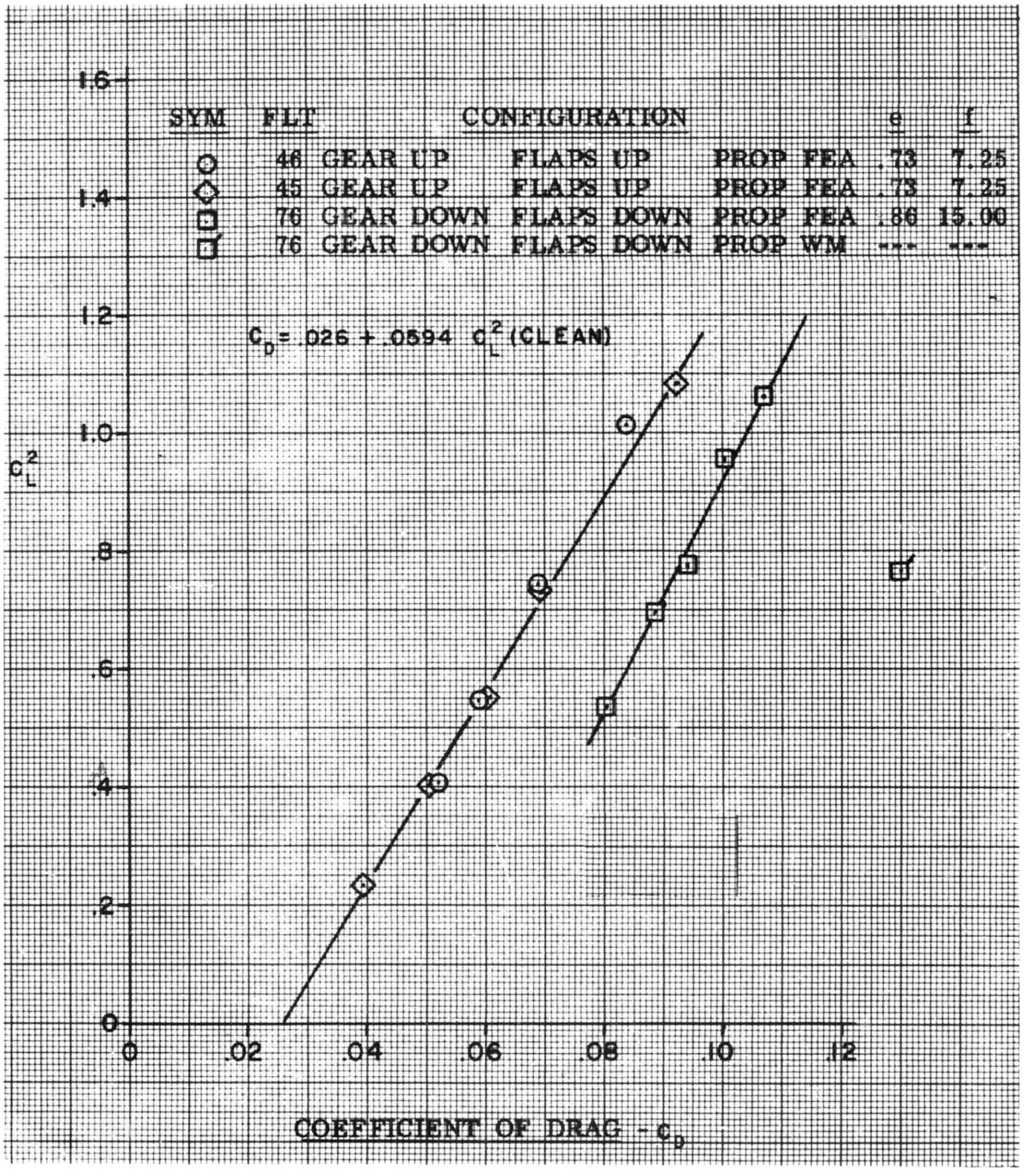


Figure 19. Drag Polar - Single-Engine Climb.

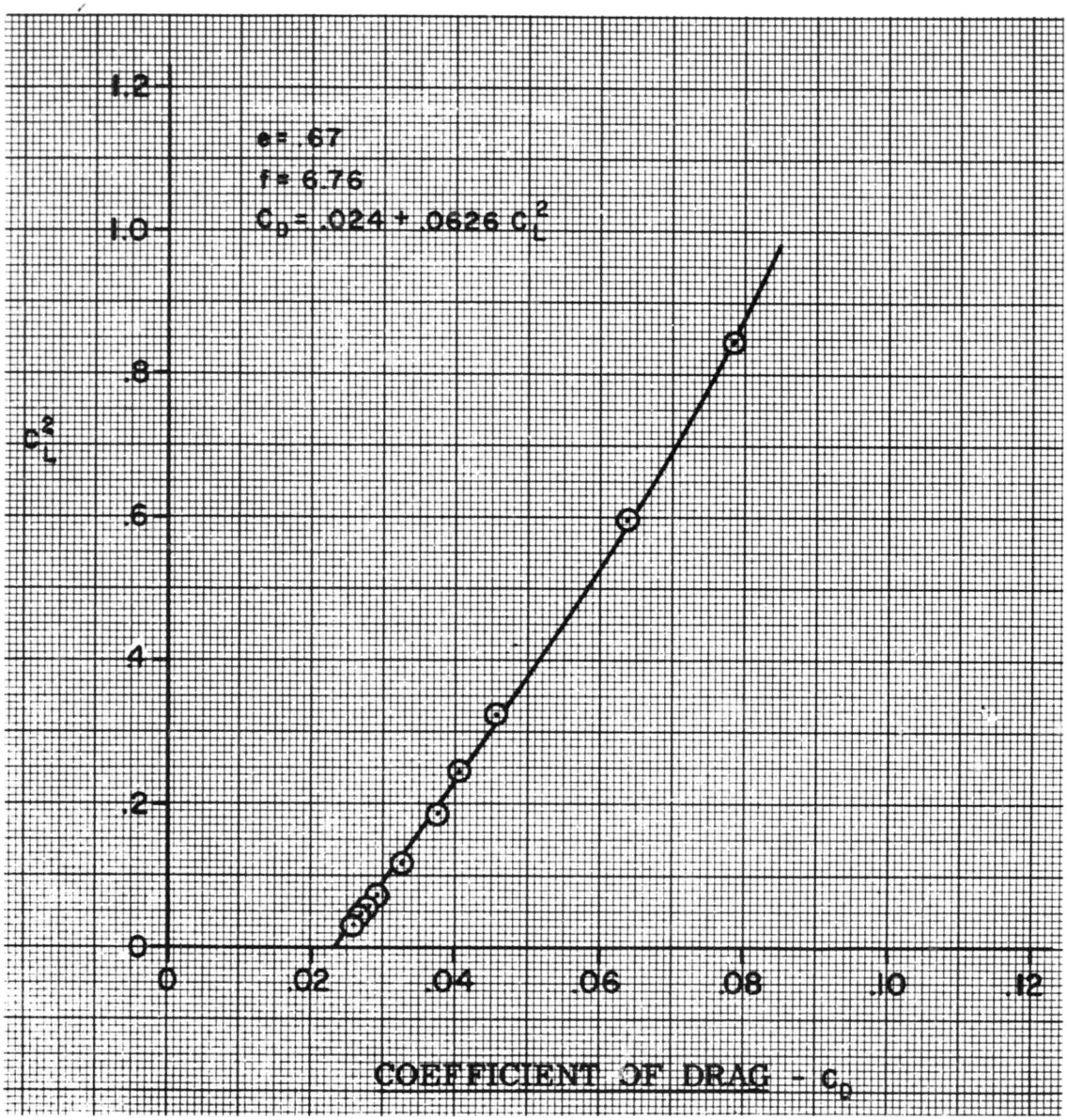


Figure 20. Drag Polar - Cruise Condition.

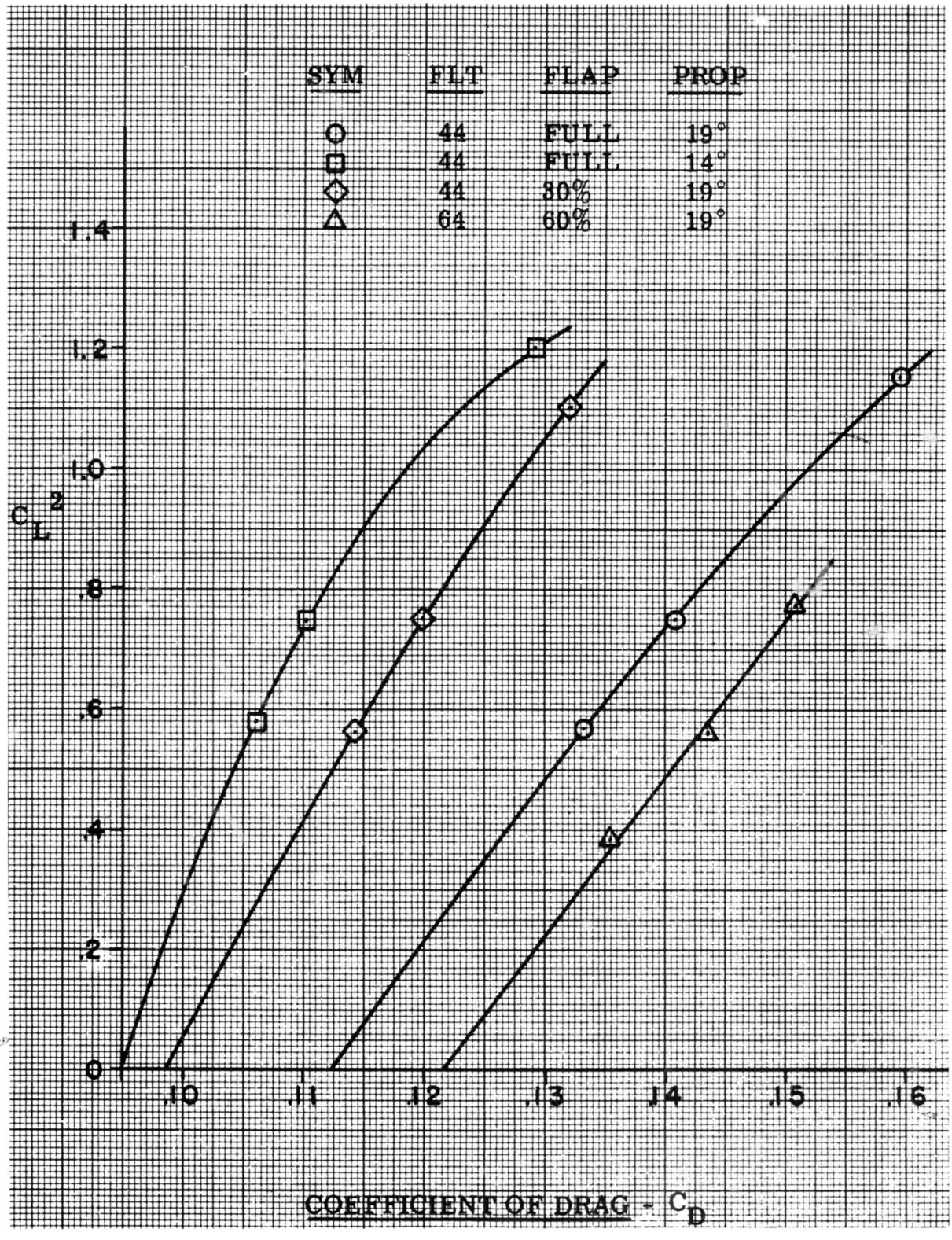


Figure 21. Drag Polar - Landing Configuration.

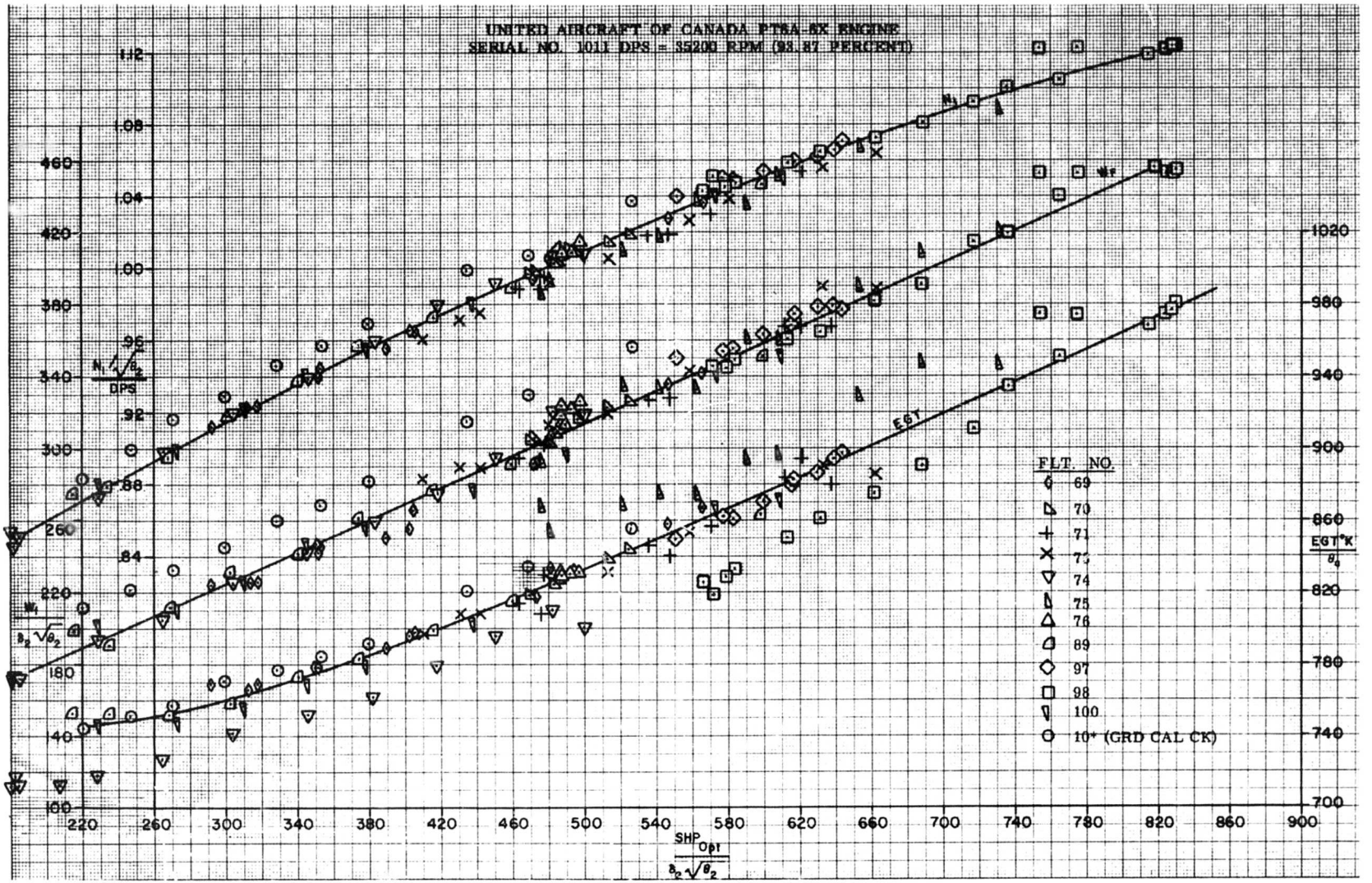


Figure 22. Engine Calibration and Performance - PT6A-6X Engines.

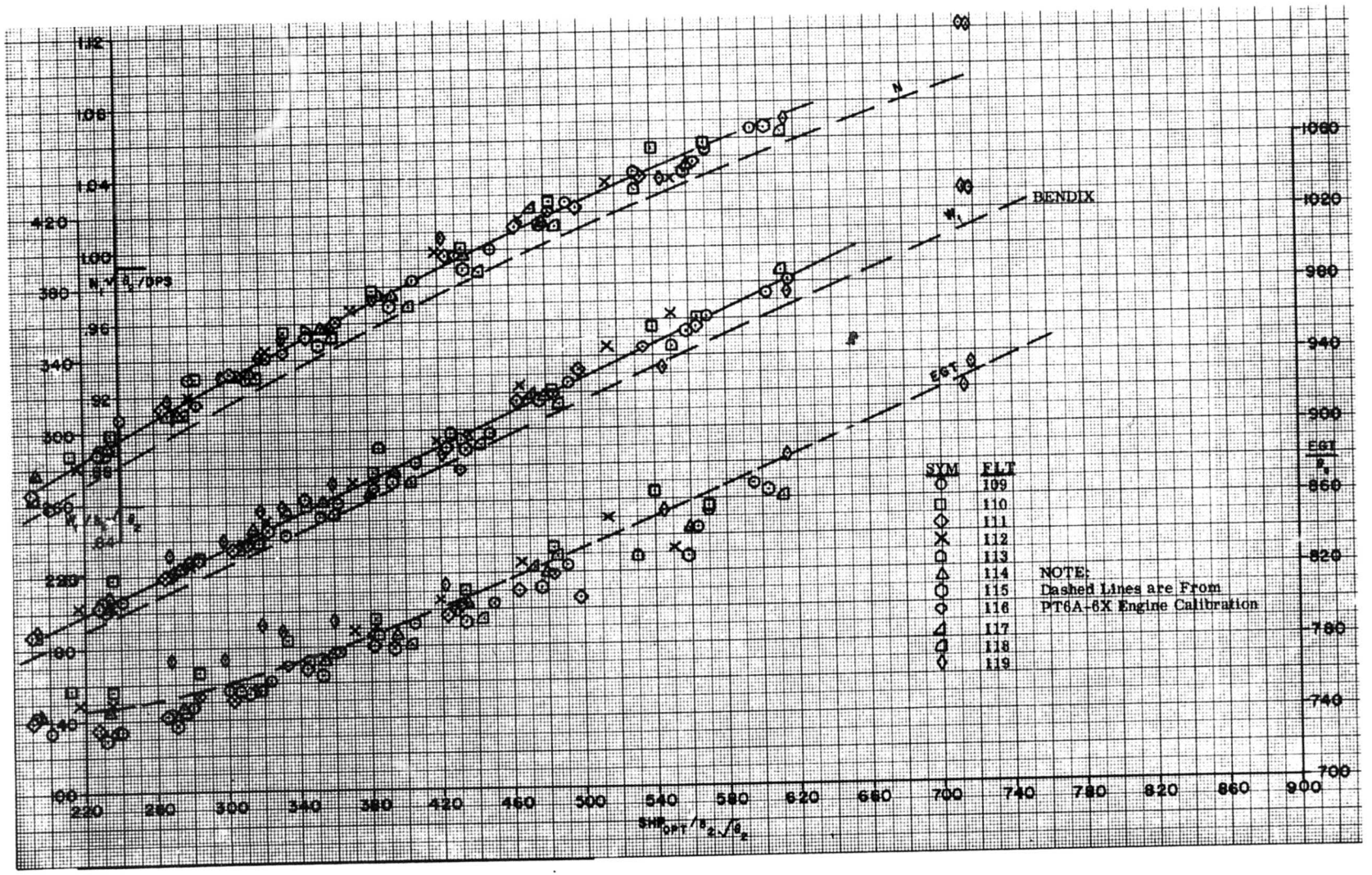


Figure 23. Engine Calibration and Performance - PT6A-6 Engines.

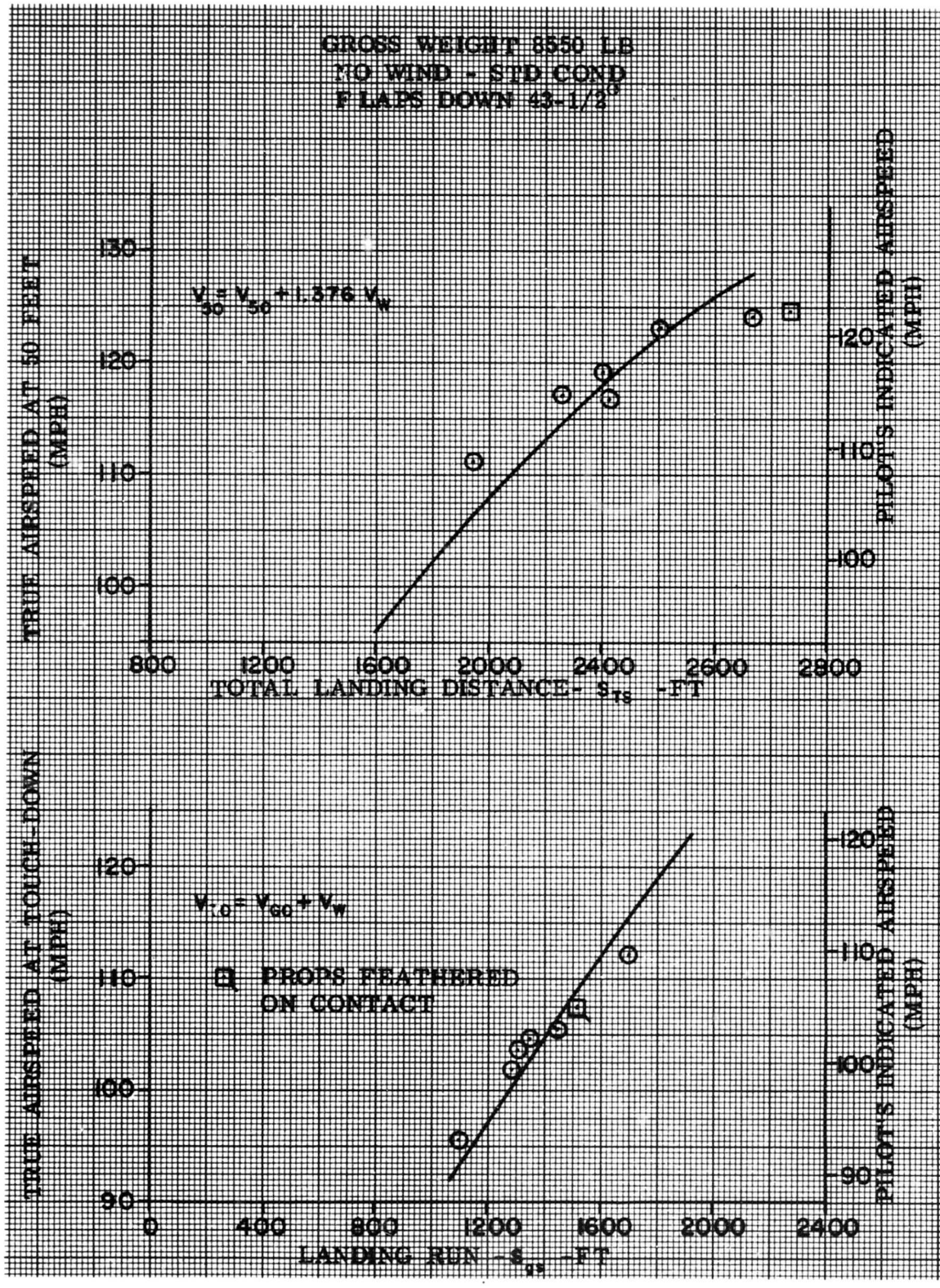


Figure 24. FAA Landing Distance.

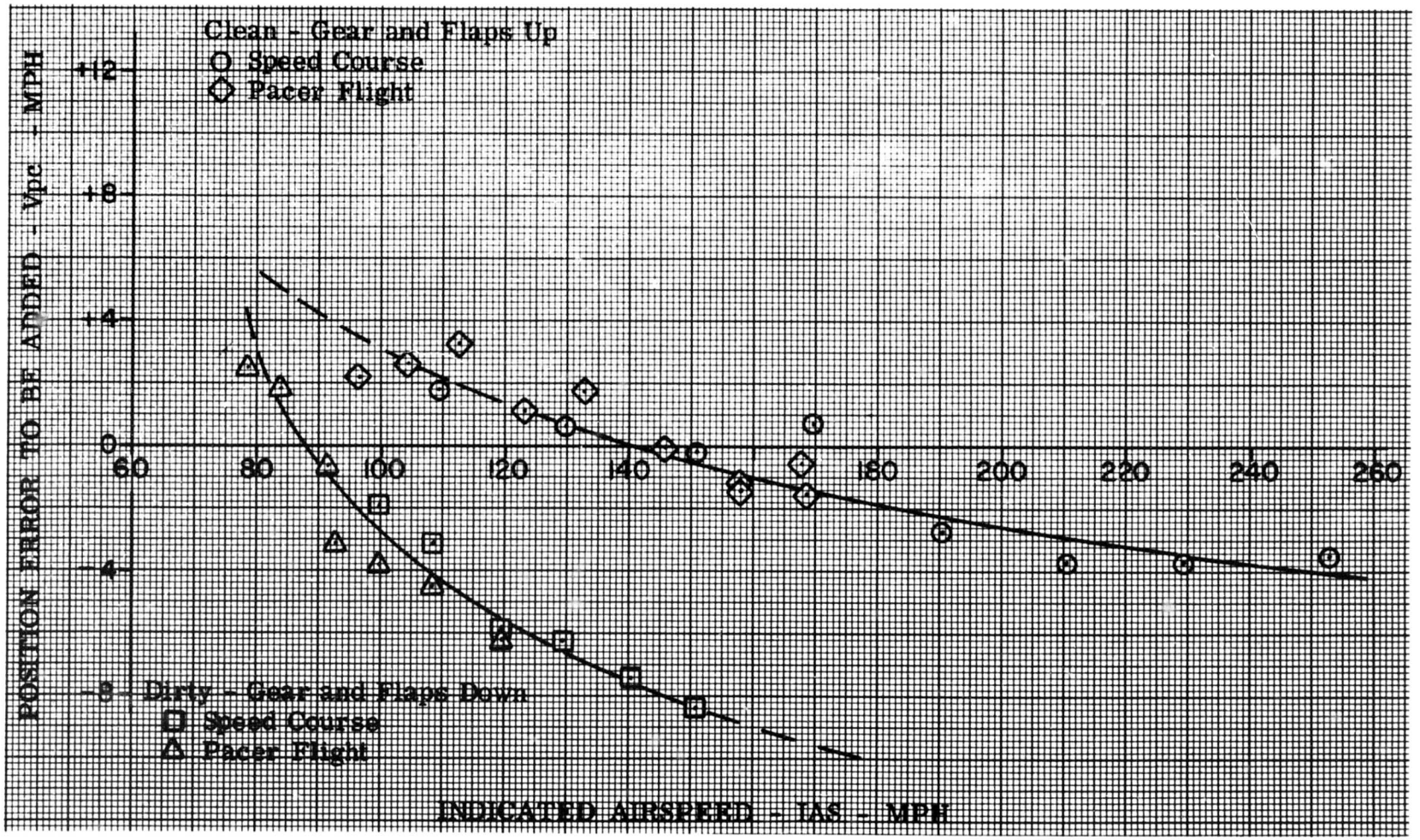


Figure 25. Airspeed Calibration - Ship's System.

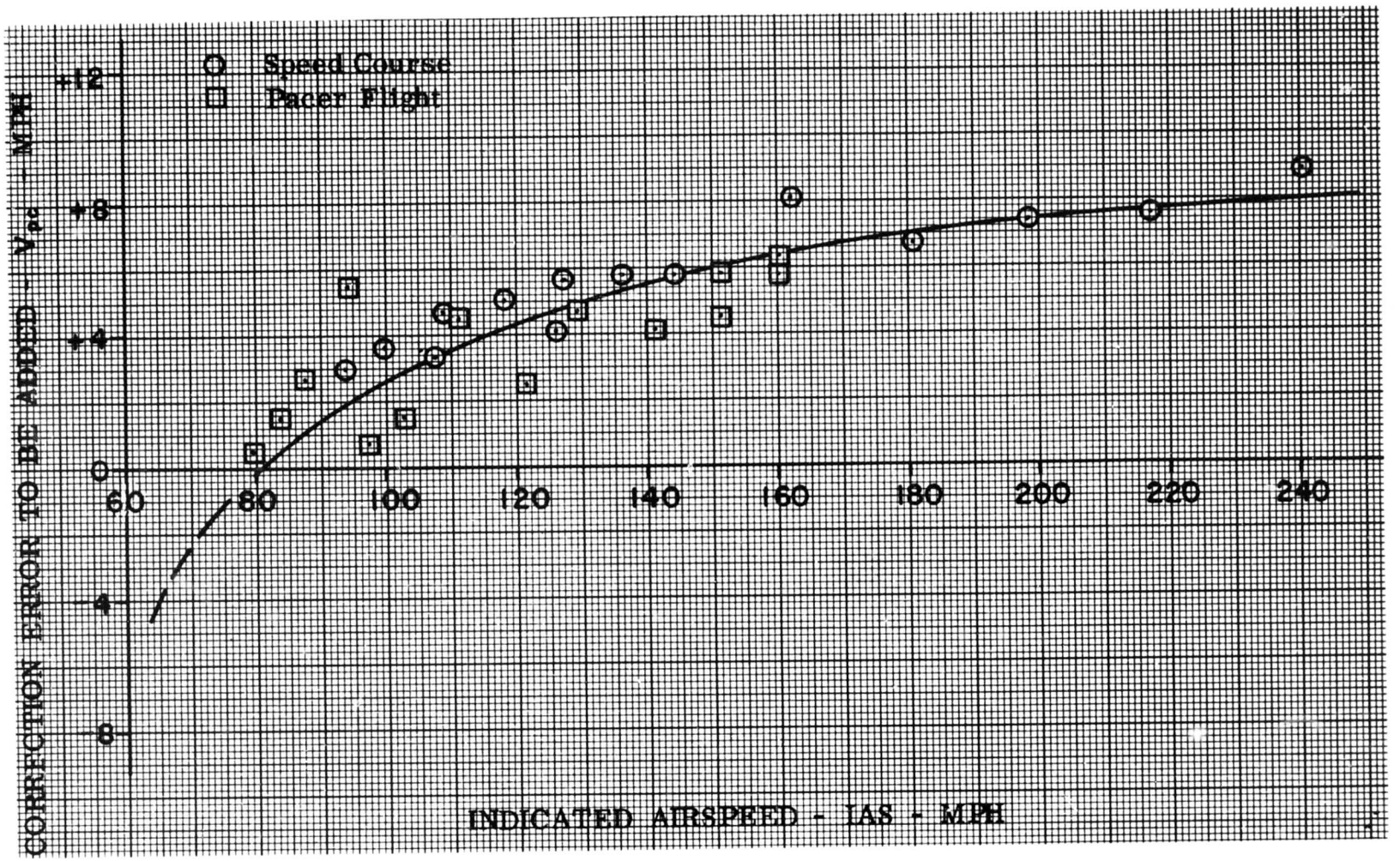


Figure 26. Airspeed Calibration - Boom.

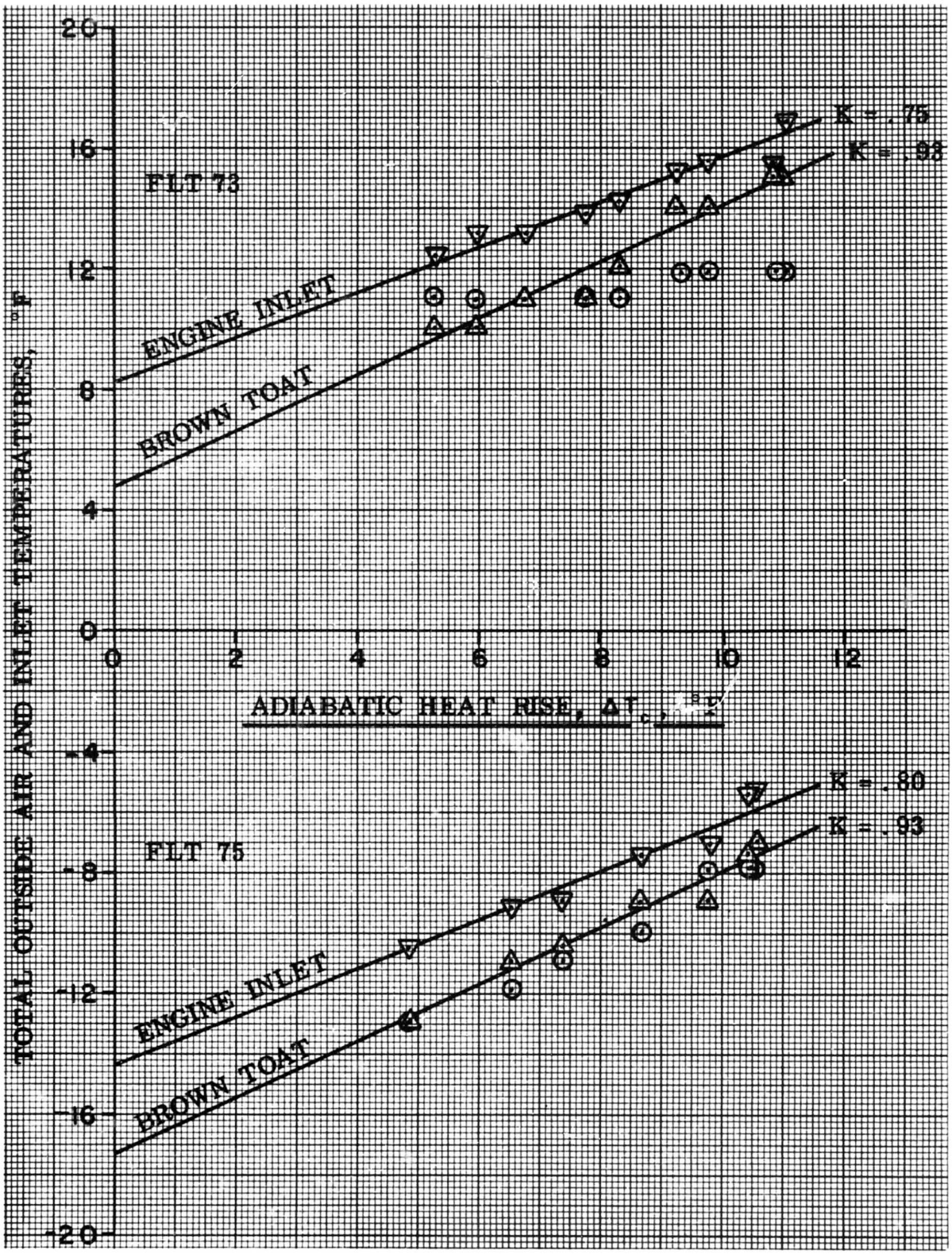


Figure 27. Temperature Probe Calibration.

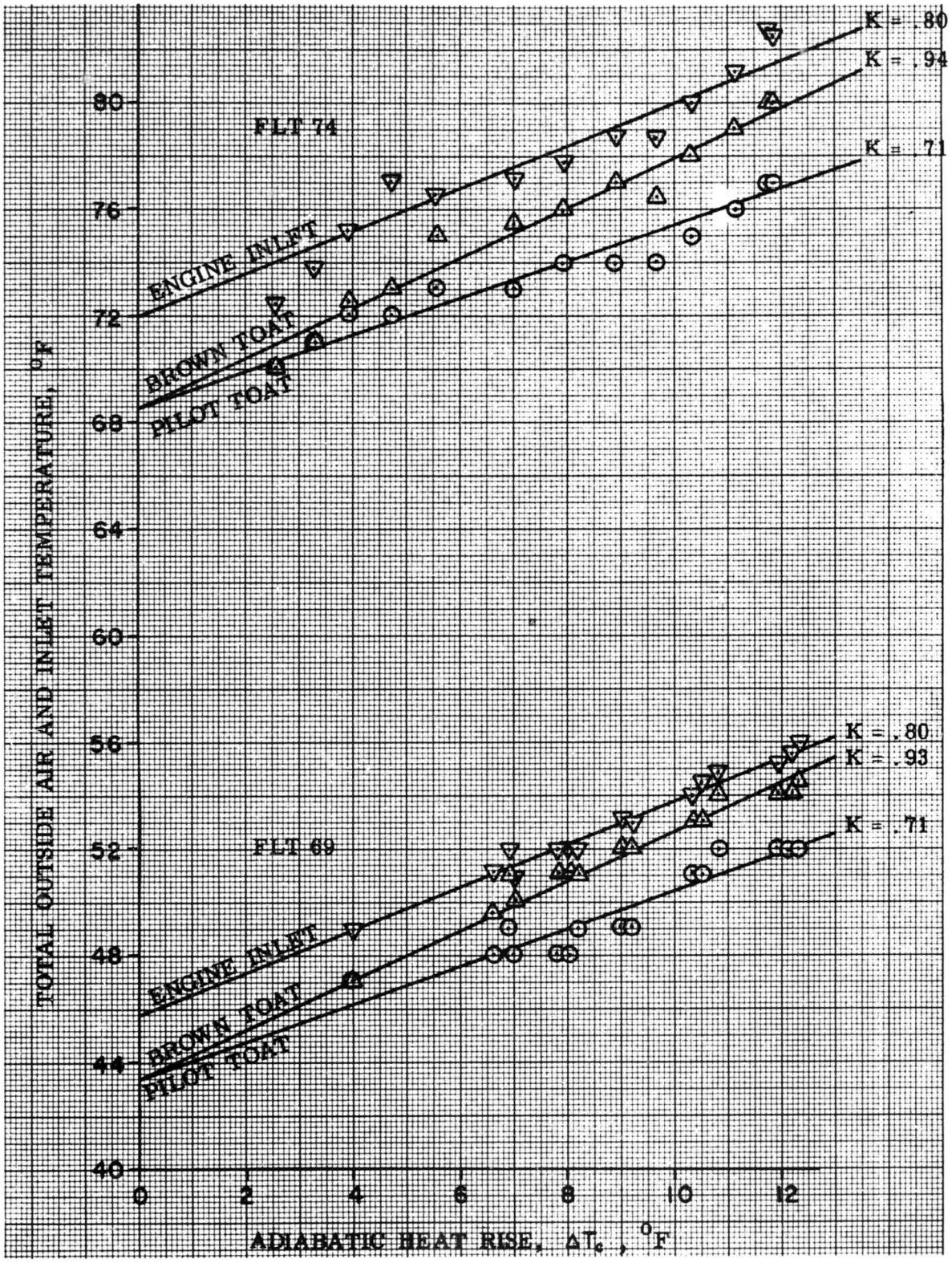


Figure 28. Temperature Probe Calibration.

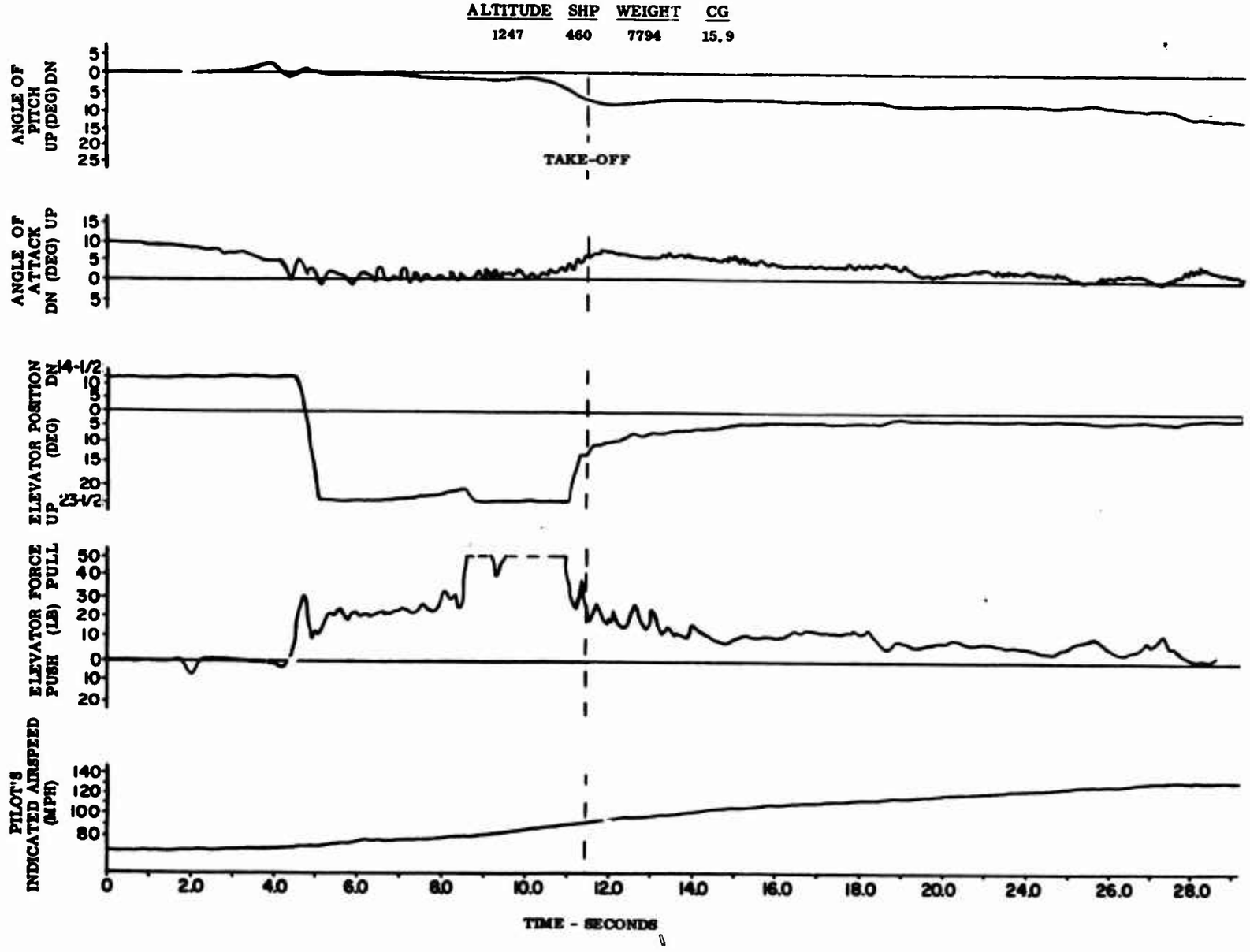


Figure 29. Take-Off Time History - Two-Engine.

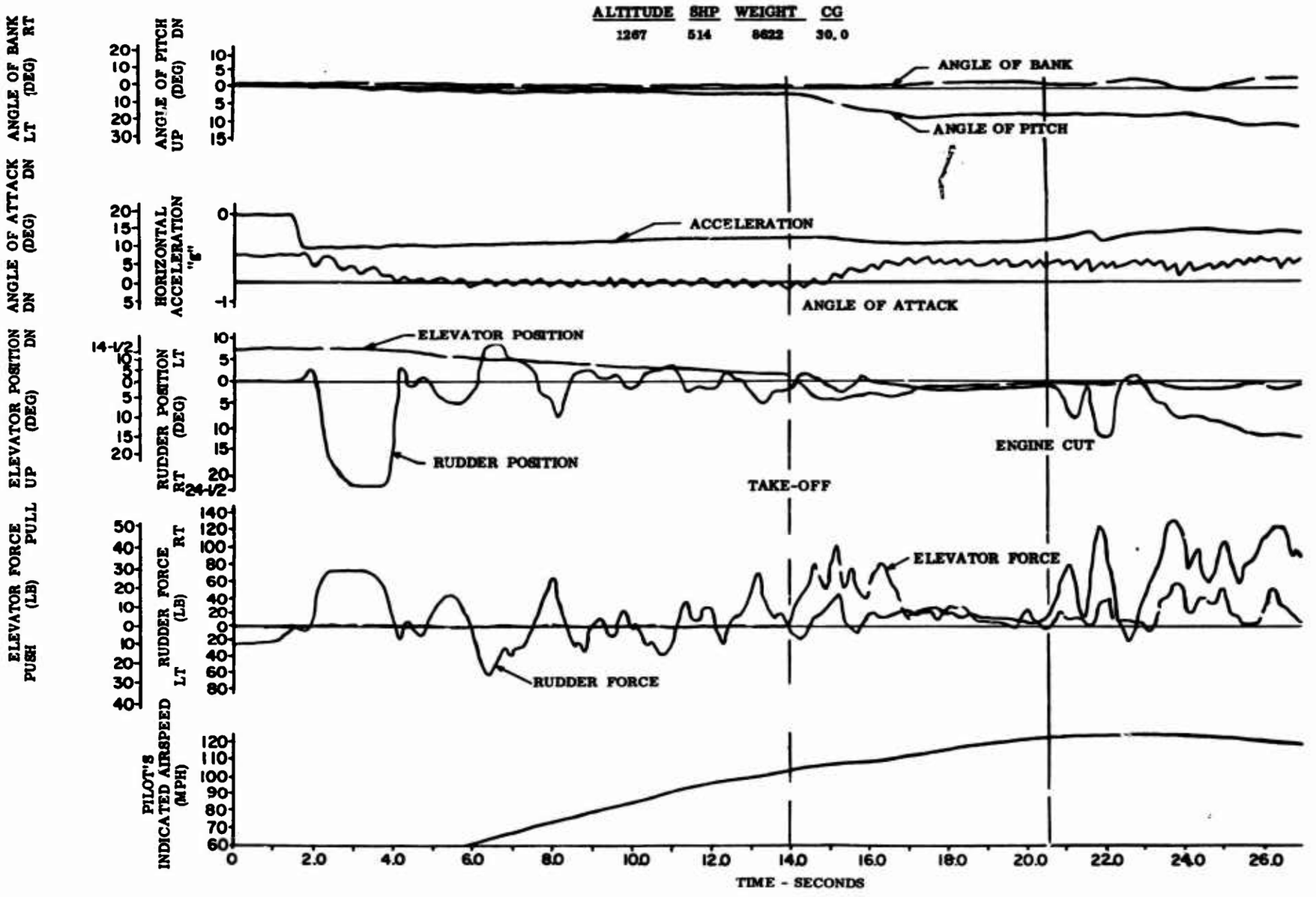


Figure 30. Take-Off Time History - Single-Engine.

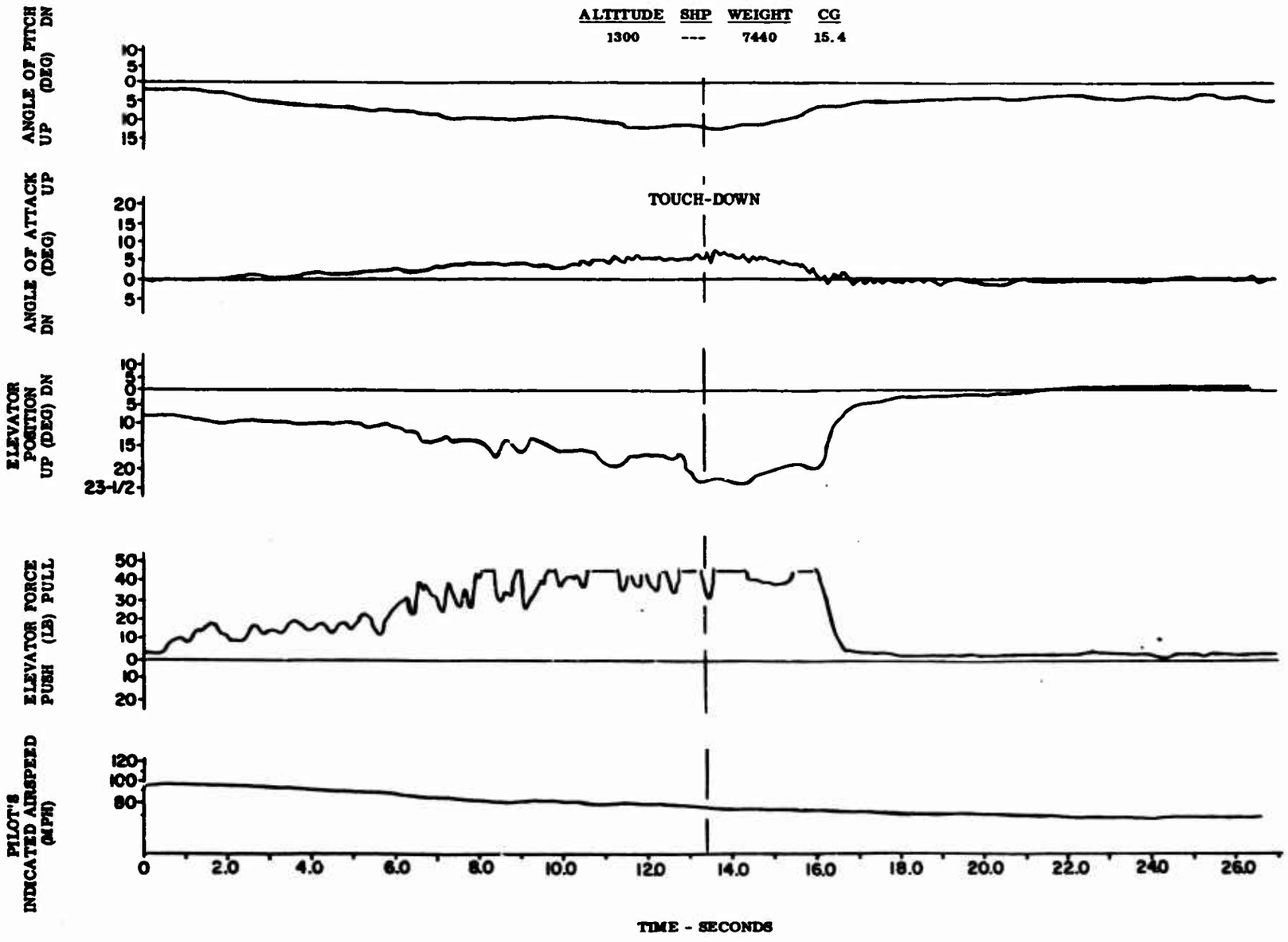


Figure 31. Landing Time History.

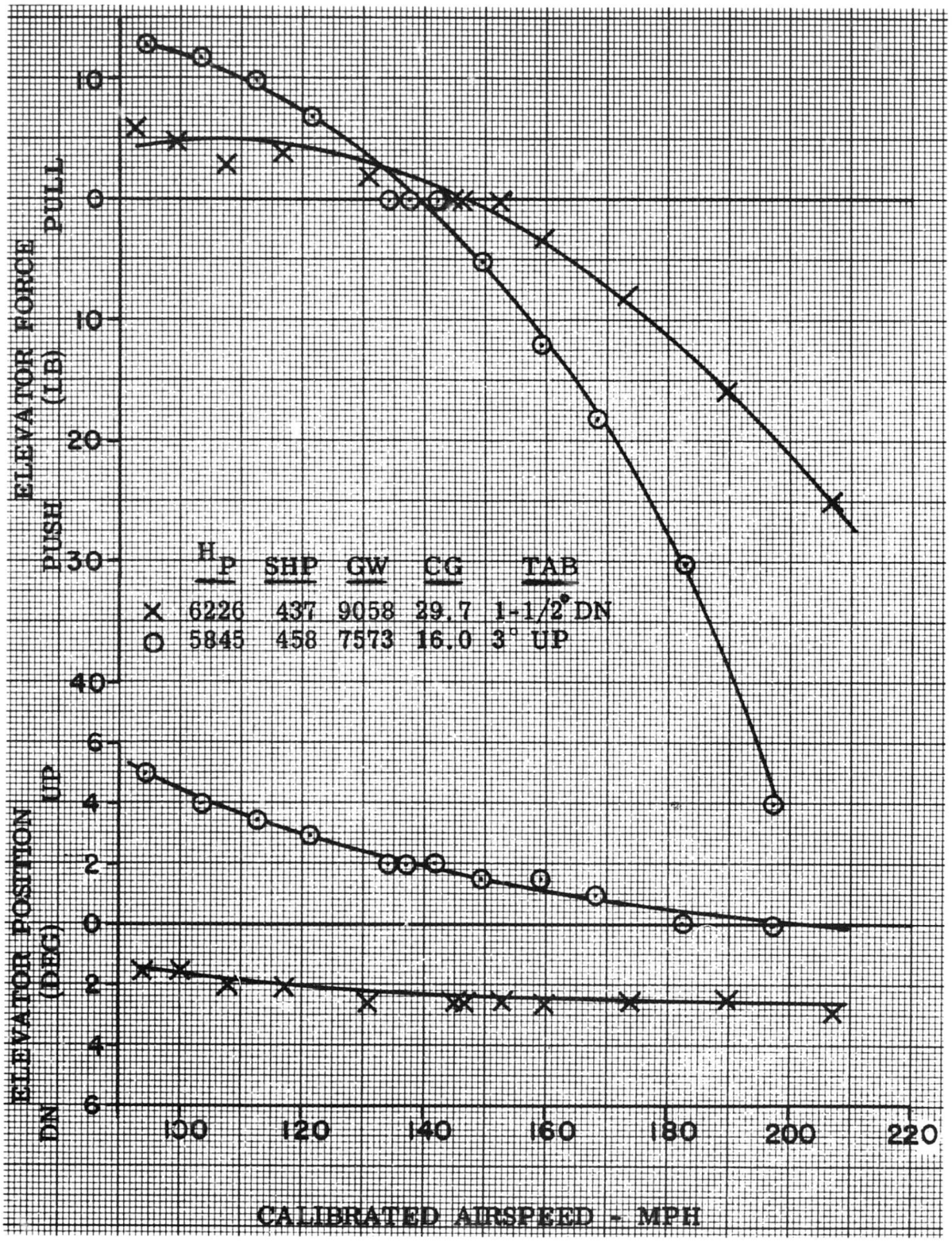


Figure 32. Static Longitudinal Stability - Climb Configuration.

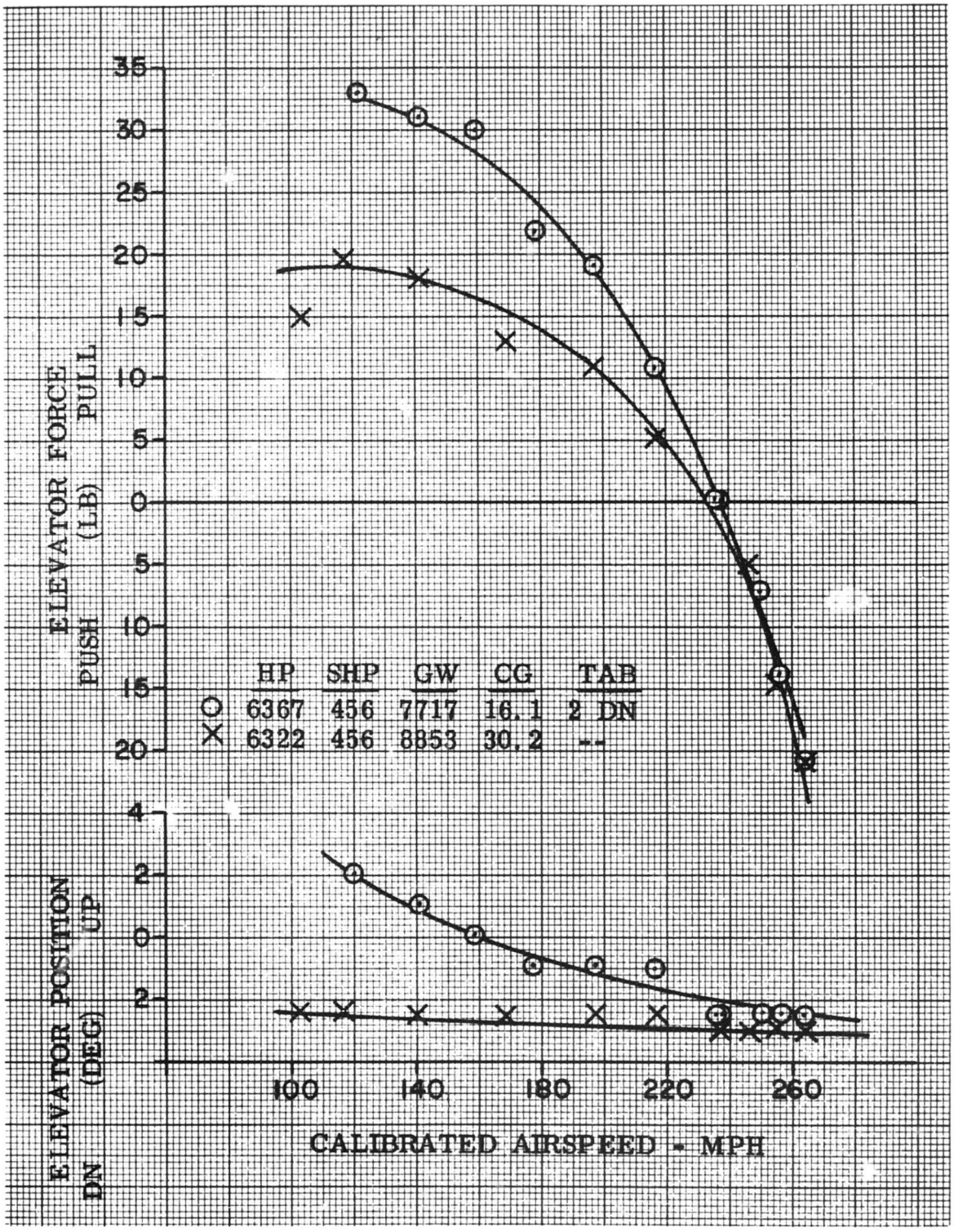


Figure 33. Static Longitudinal Stability - Cruise Configuration.

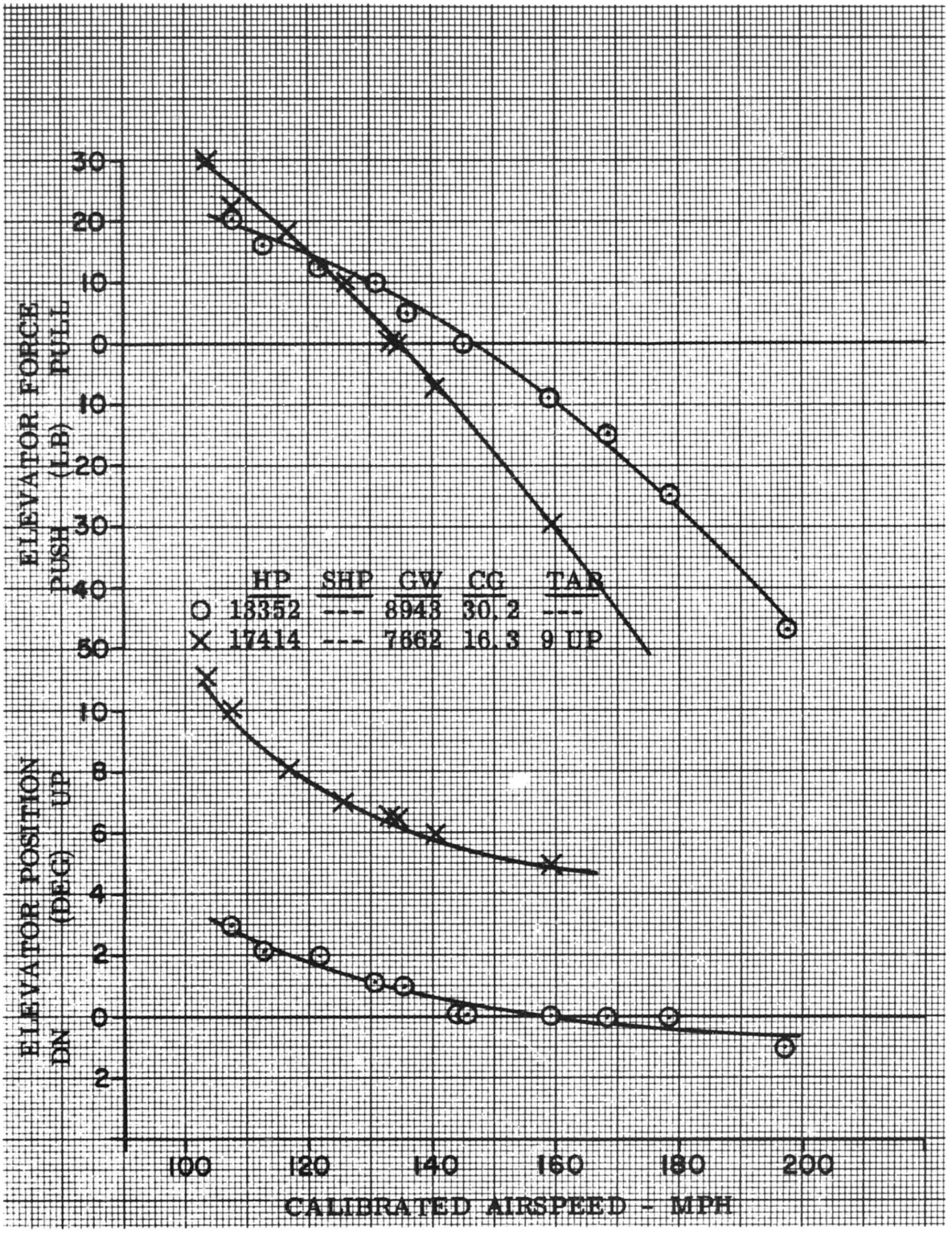


Figure 34. Static Longitudinal Stability - Glide Configuration.

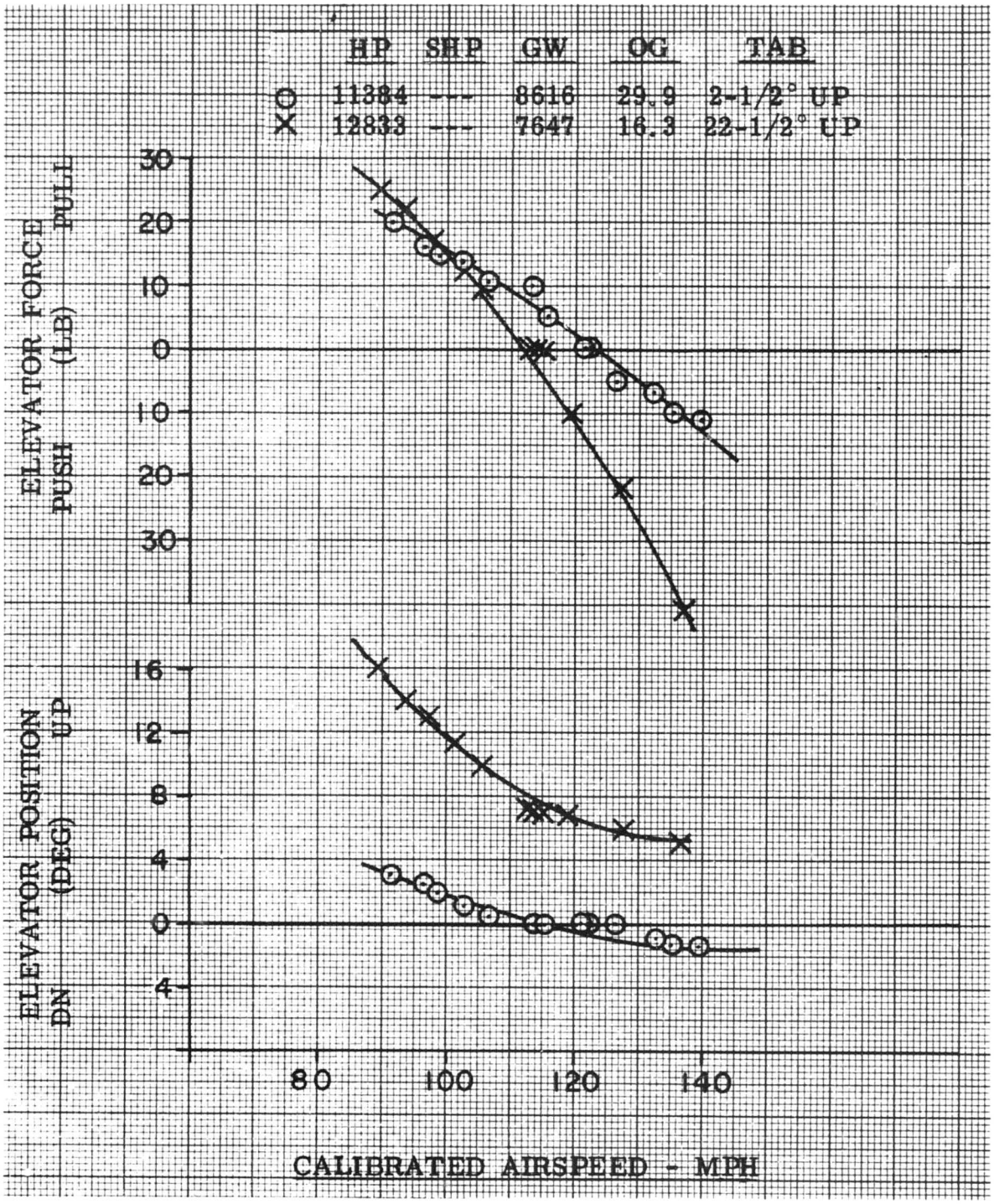


Figure 35. Static Longitudinal Stability - Landing Configuration.

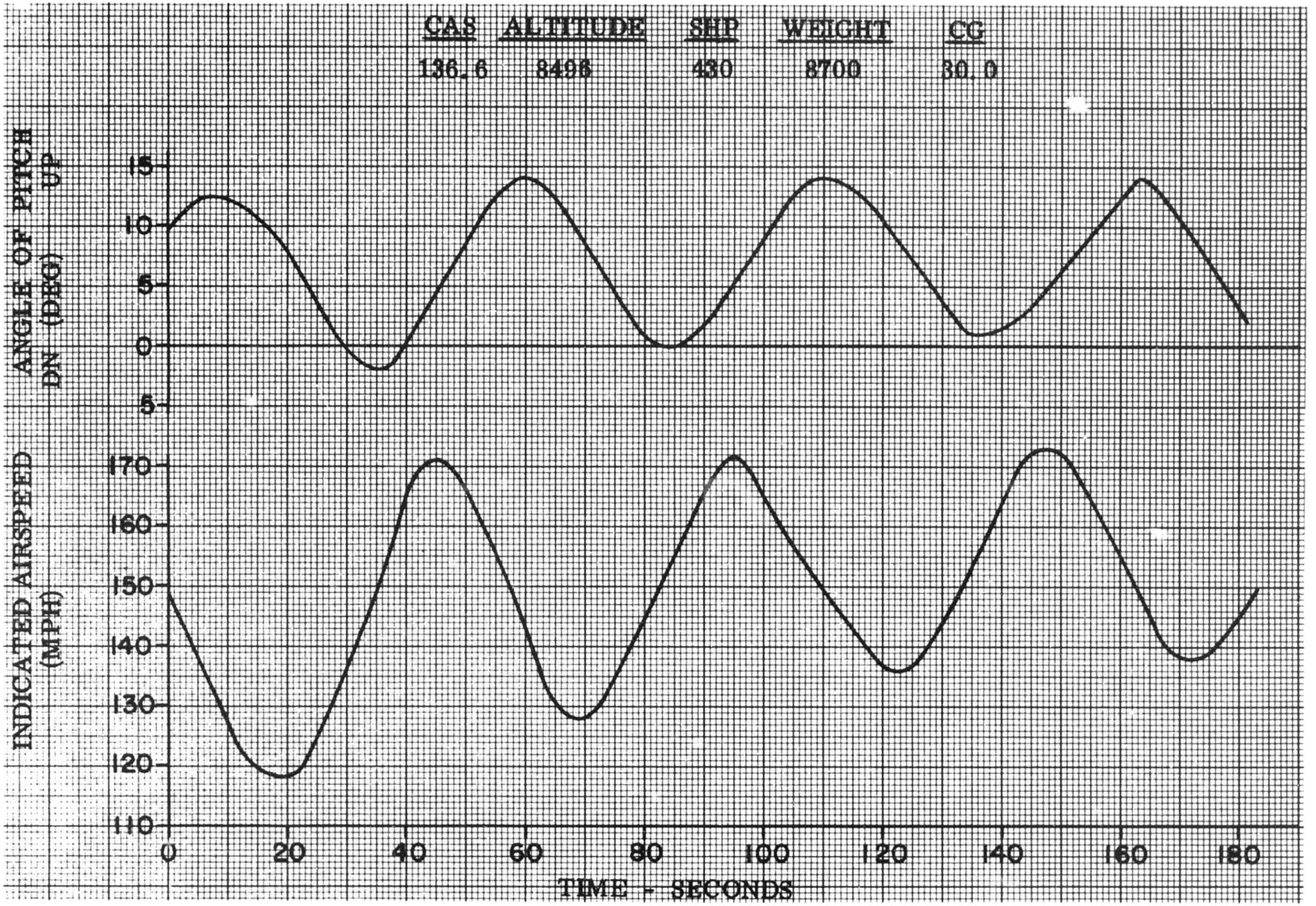


Figure 36. Dynamic Longitudinal Stability - Phugoid - Climb Configuration.

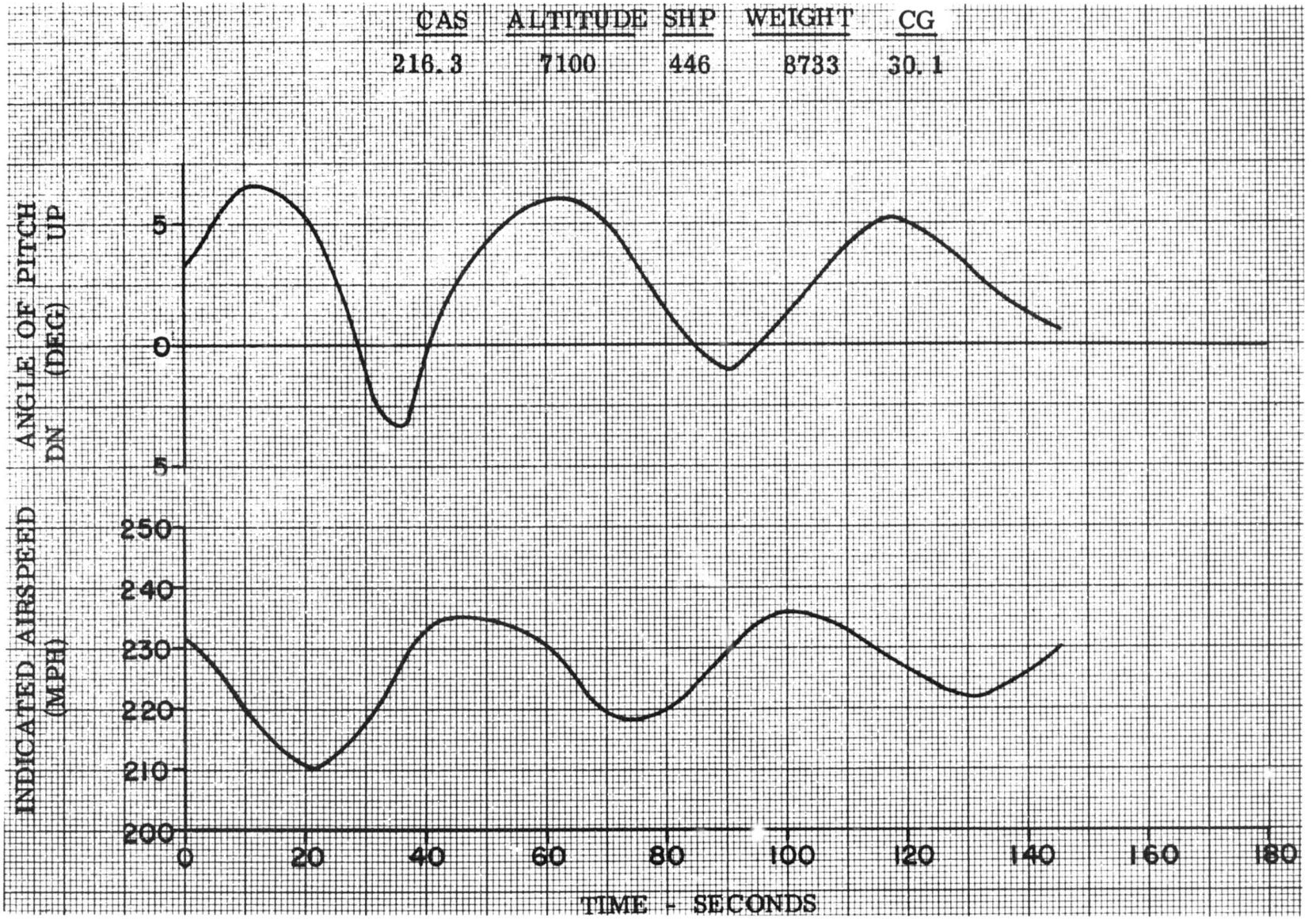


Figure 37. Dynamic Longitudinal Stability - Phugoid - Cruise Configuration.

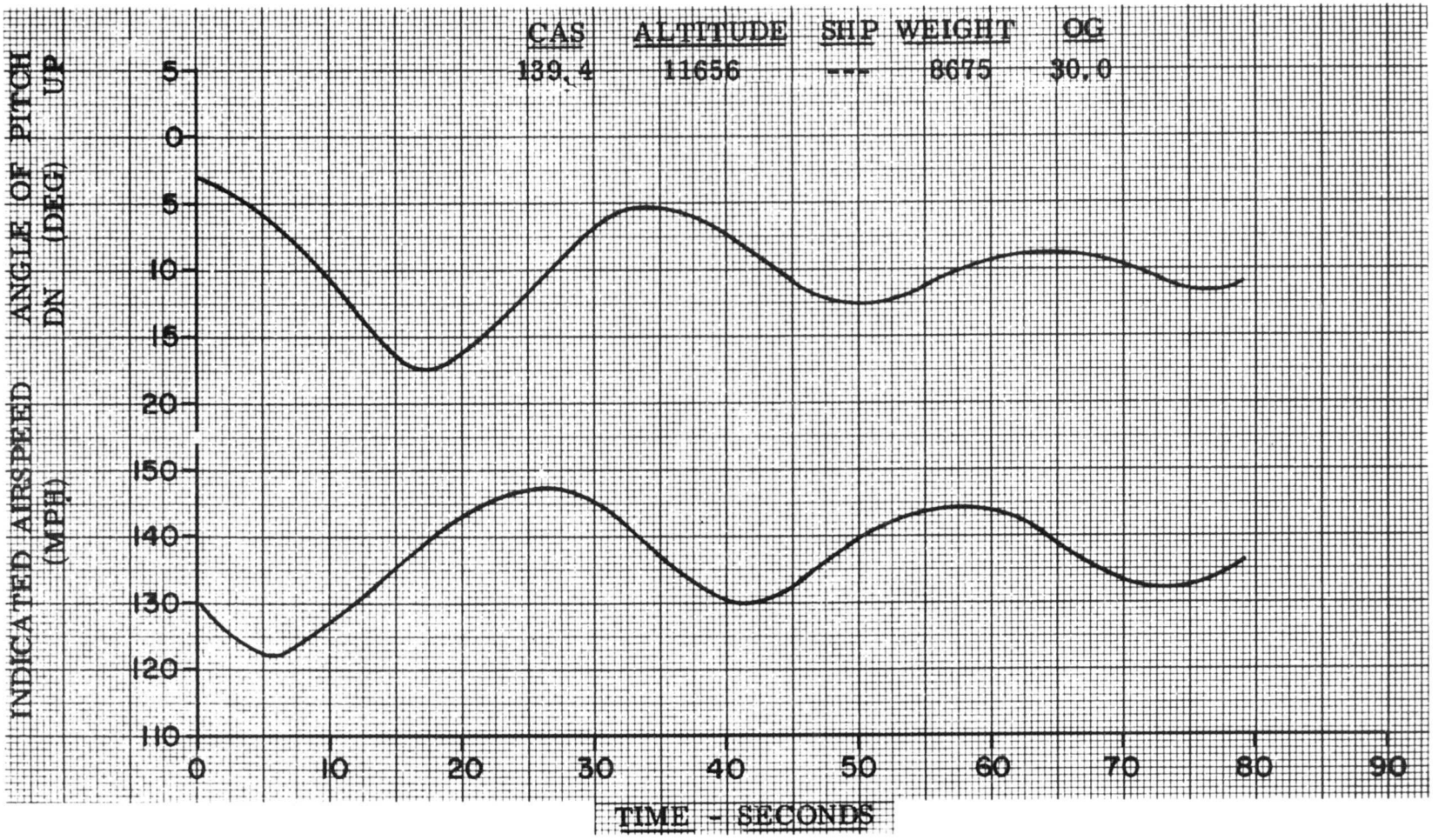


Figure 38. Dynamic Longitudinal Stability - Phugoid - Landing Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG
120	3802	450	8781	30.2

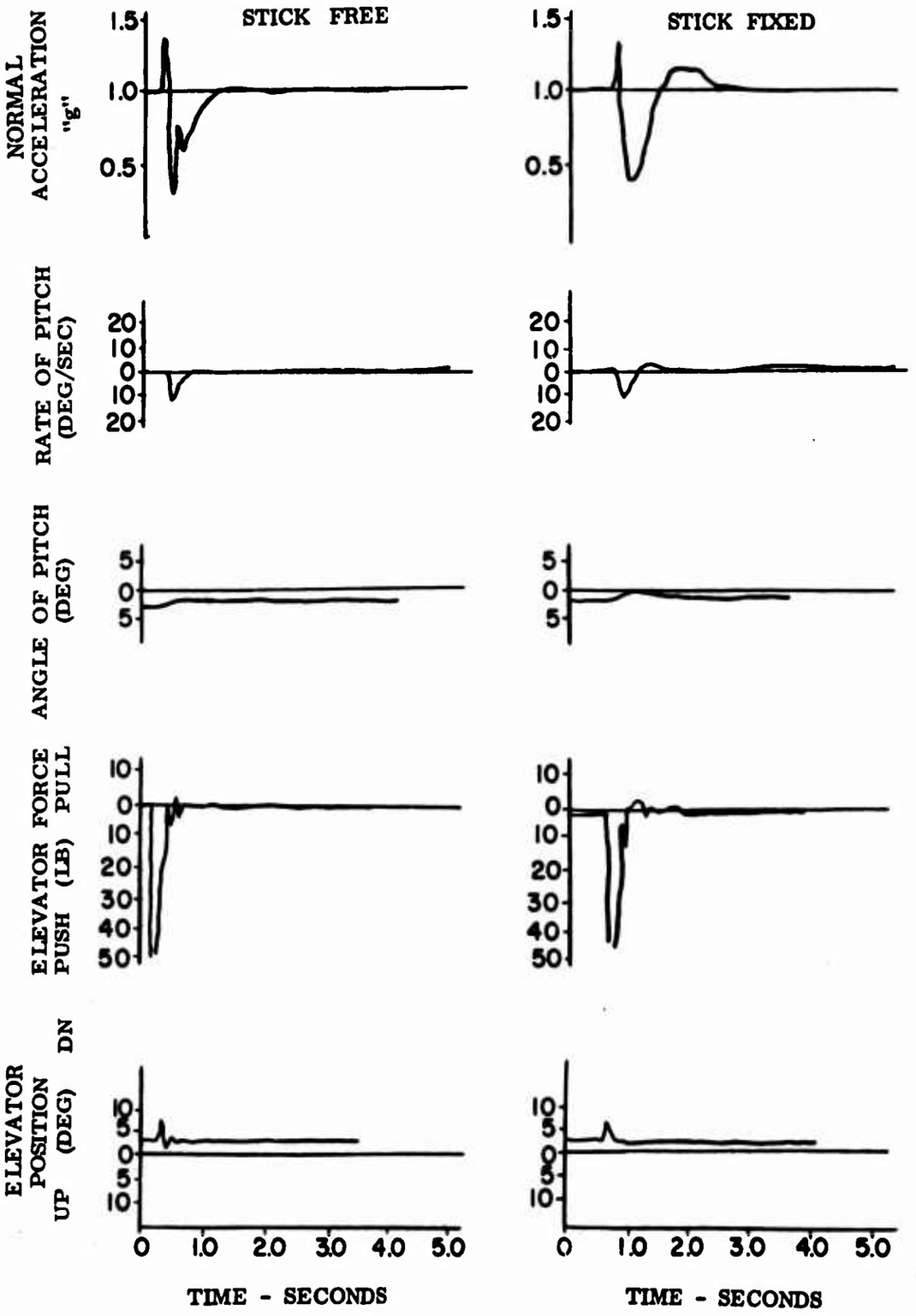


Figure 39. Dynamic Longitudinal Stability - Cruise Configuration.

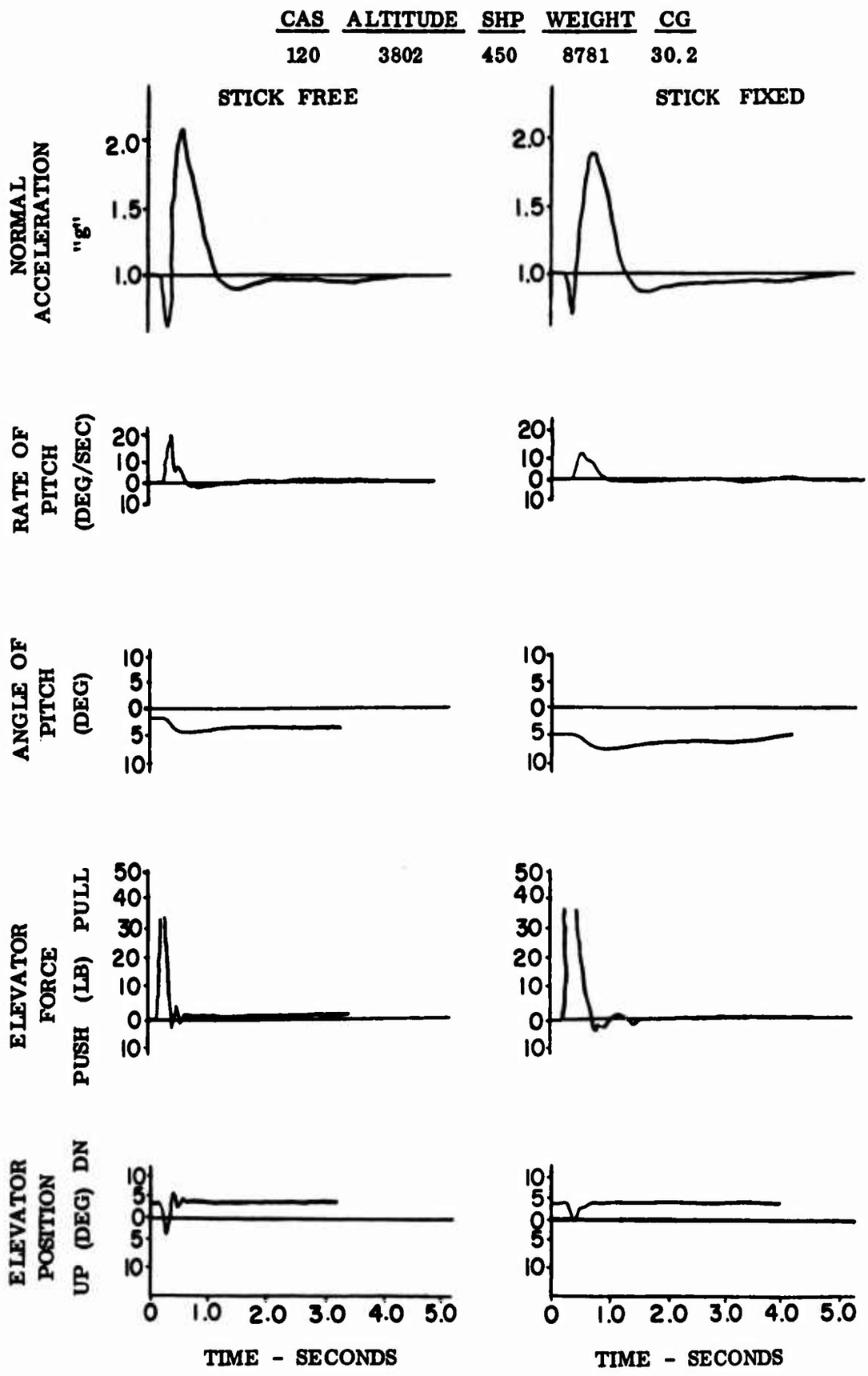


Figure 40. Dynamic Longitudinal Stability - Cruise Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG
145.9	9928	---	8773	30.2

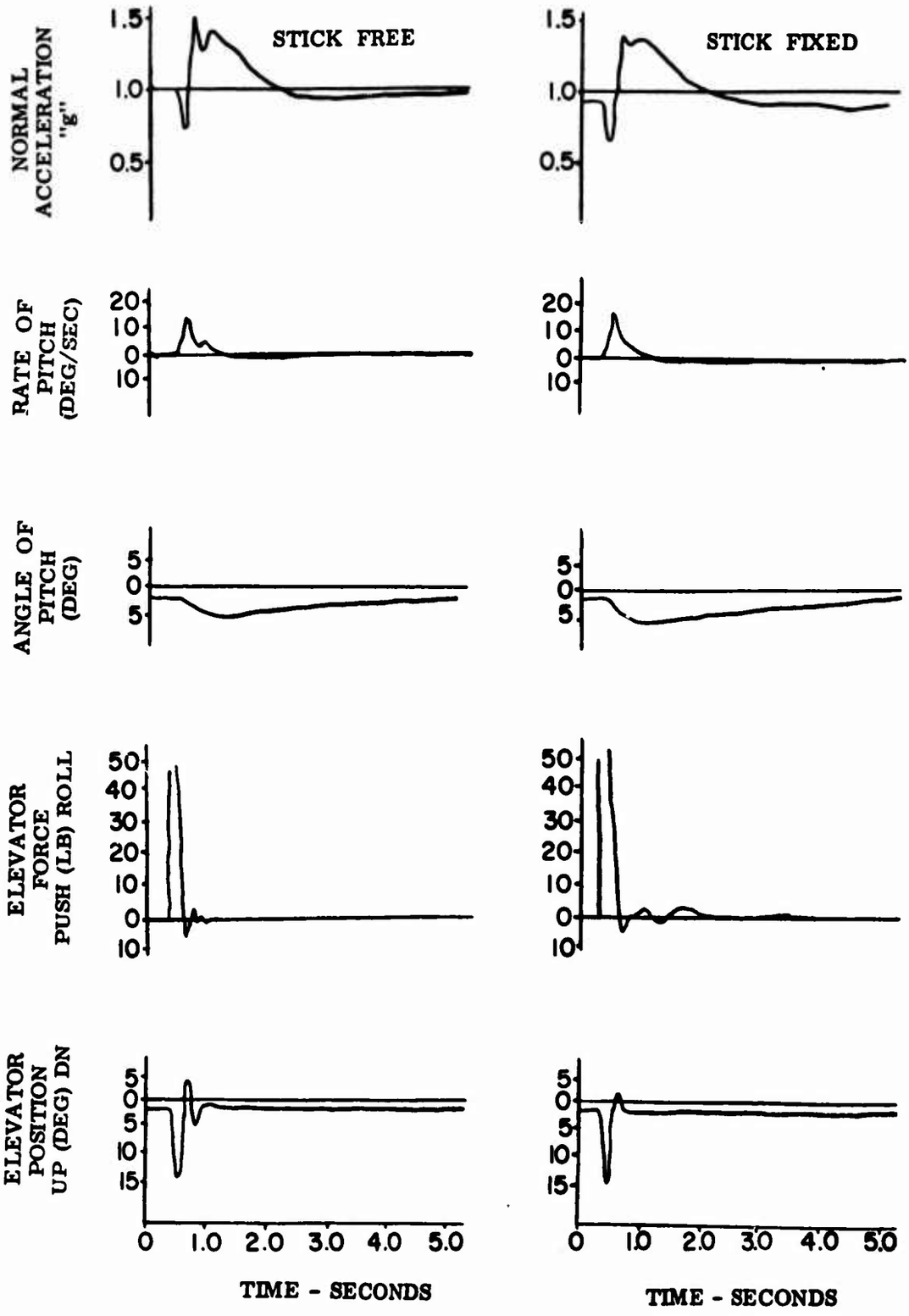


Figure 41. Dynamic Longitudinal Stability - Glide Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG
145.9	9928	---	8773	30.2

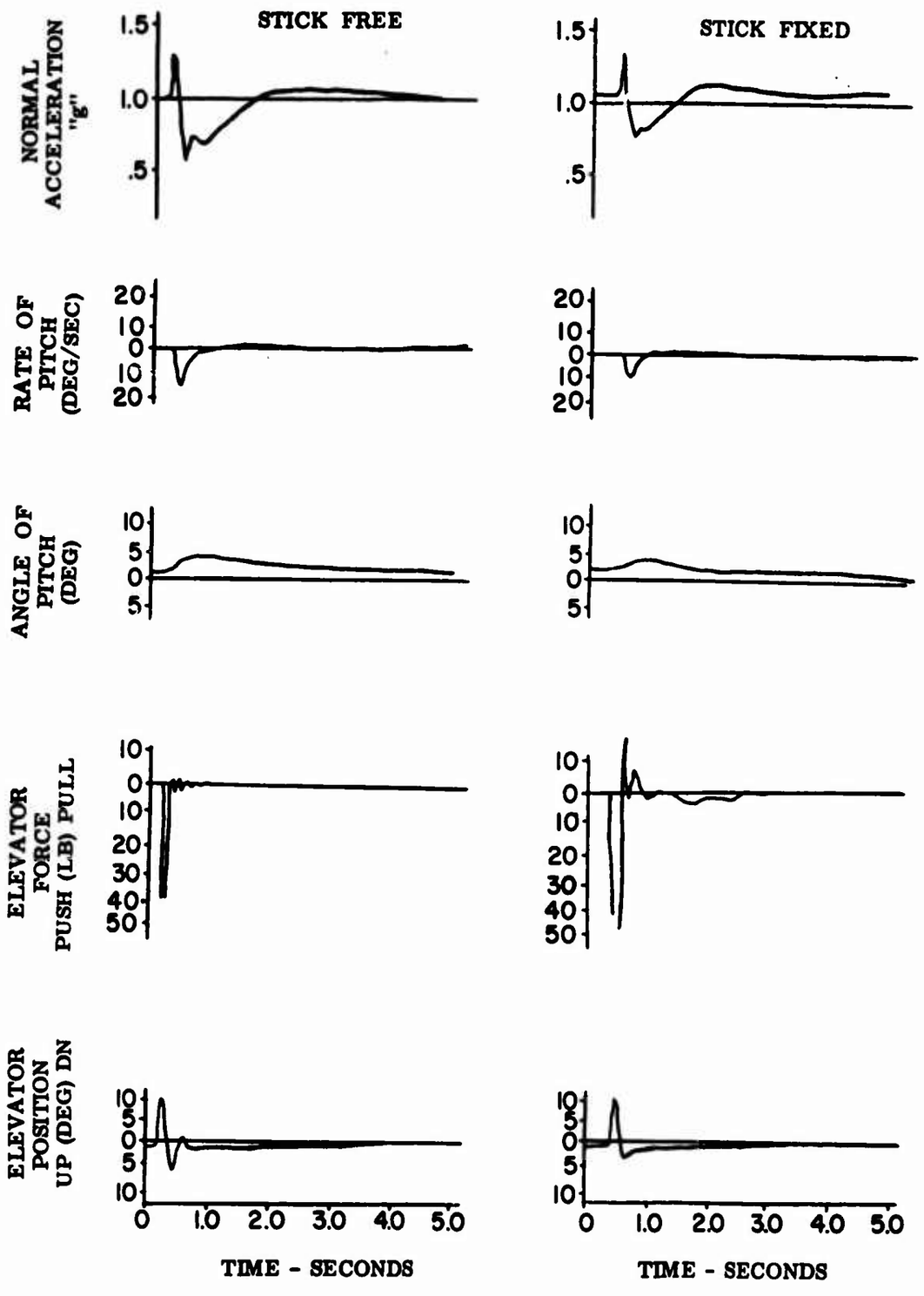


Figure 42. Dynamic Longitudinal Stability - Glide Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG
138.9	7704	---	8743	30.1

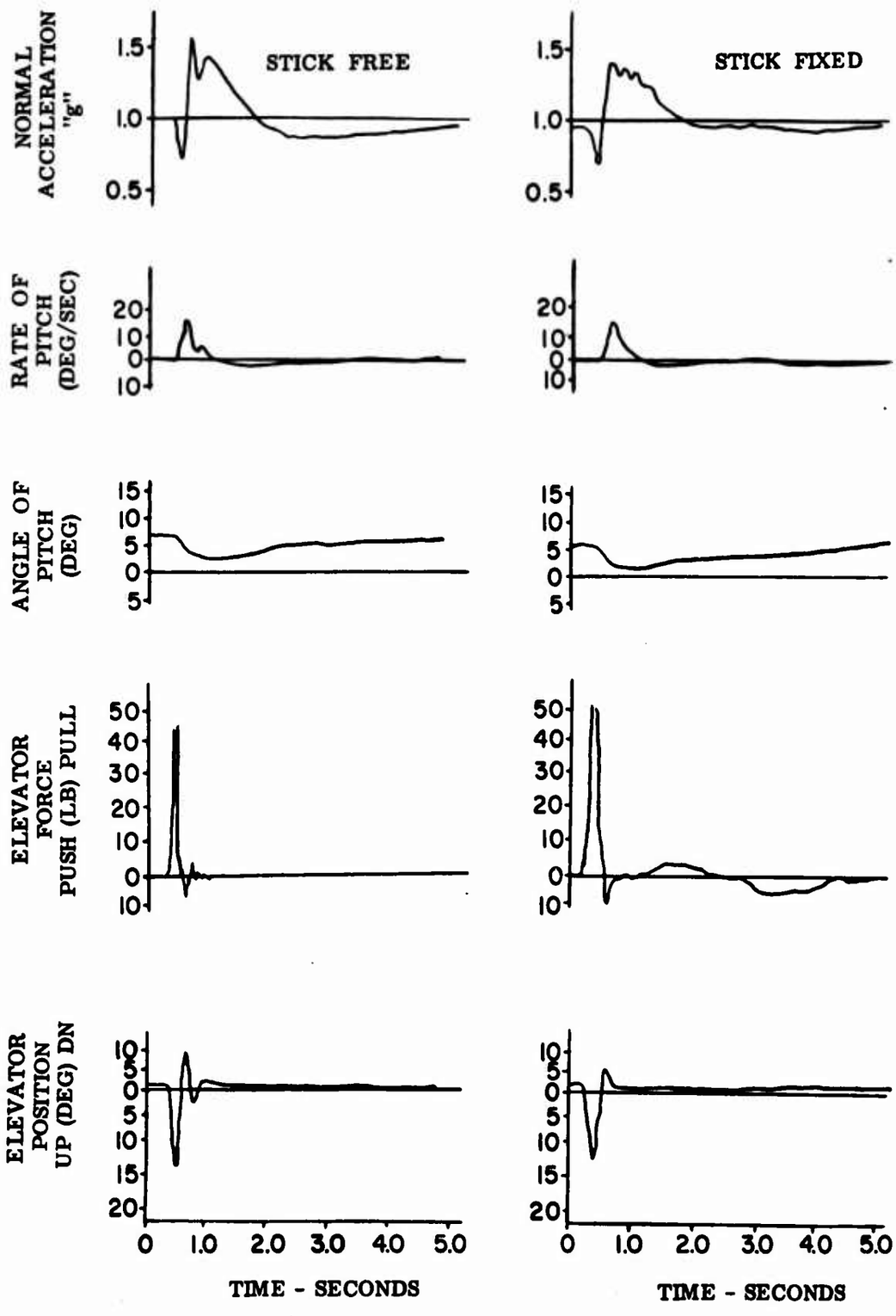


Figure 43. Dynamic Longitudinal Stability - Landing Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG
138.9	7704	---	8743	30.1

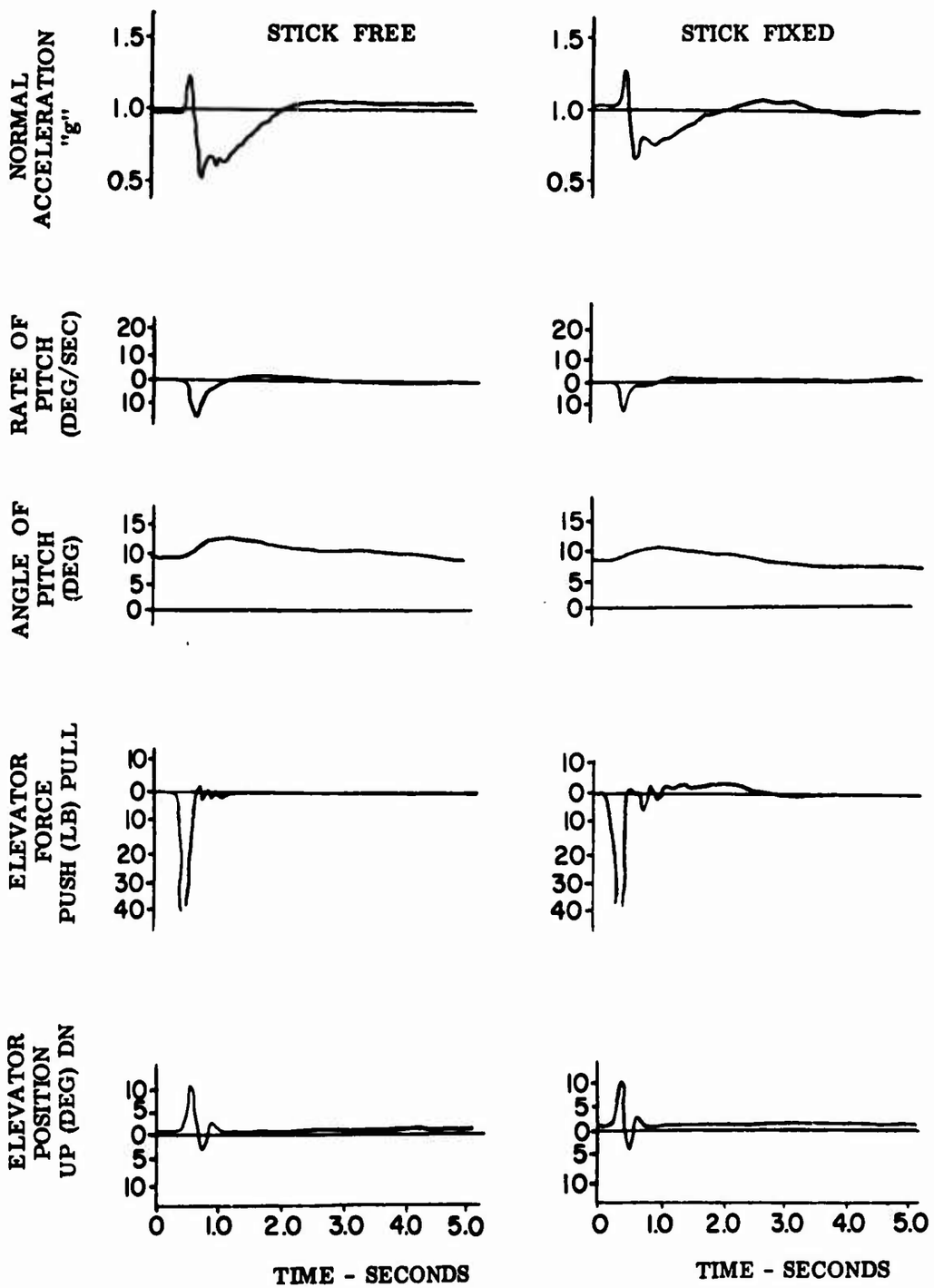


Figure 44. Dynamic Longitudinal Stability - Landing Configuration.

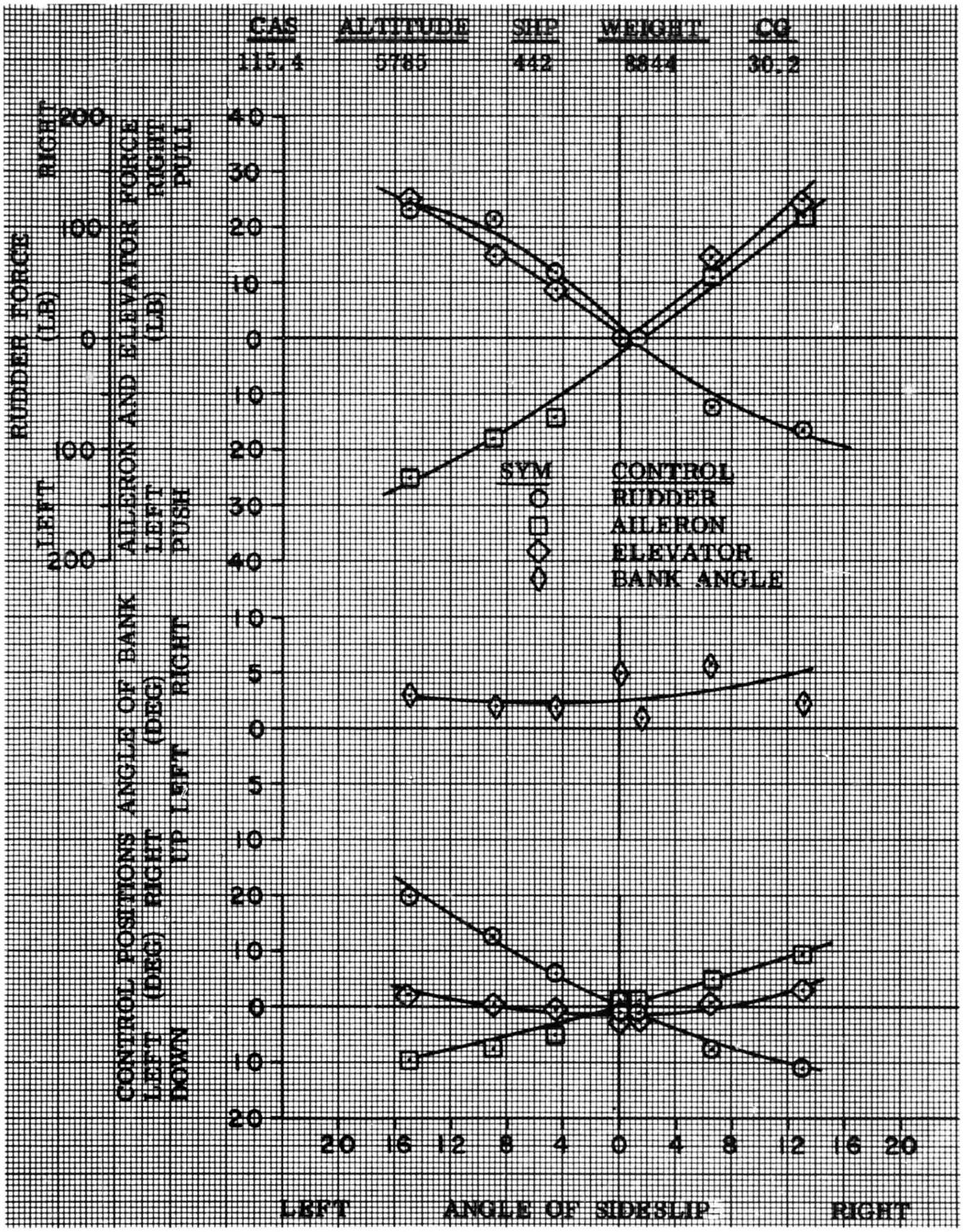


Figure 45. Static Directional Stability - Skid Characteristics - Climb Configuration.

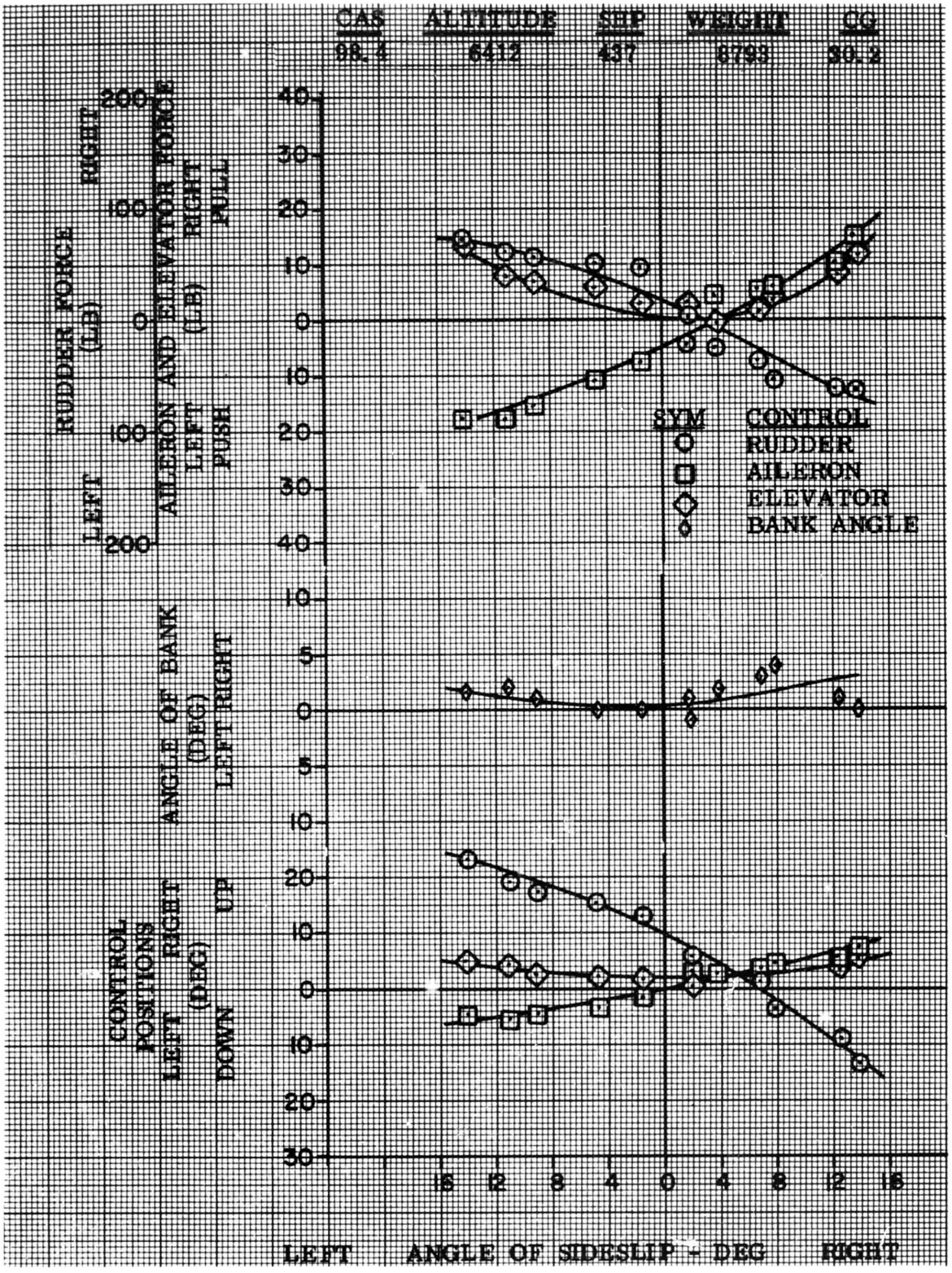


Figure 46. Static Directional Stability - Skid Characteristics - Wave-Off Configuration.

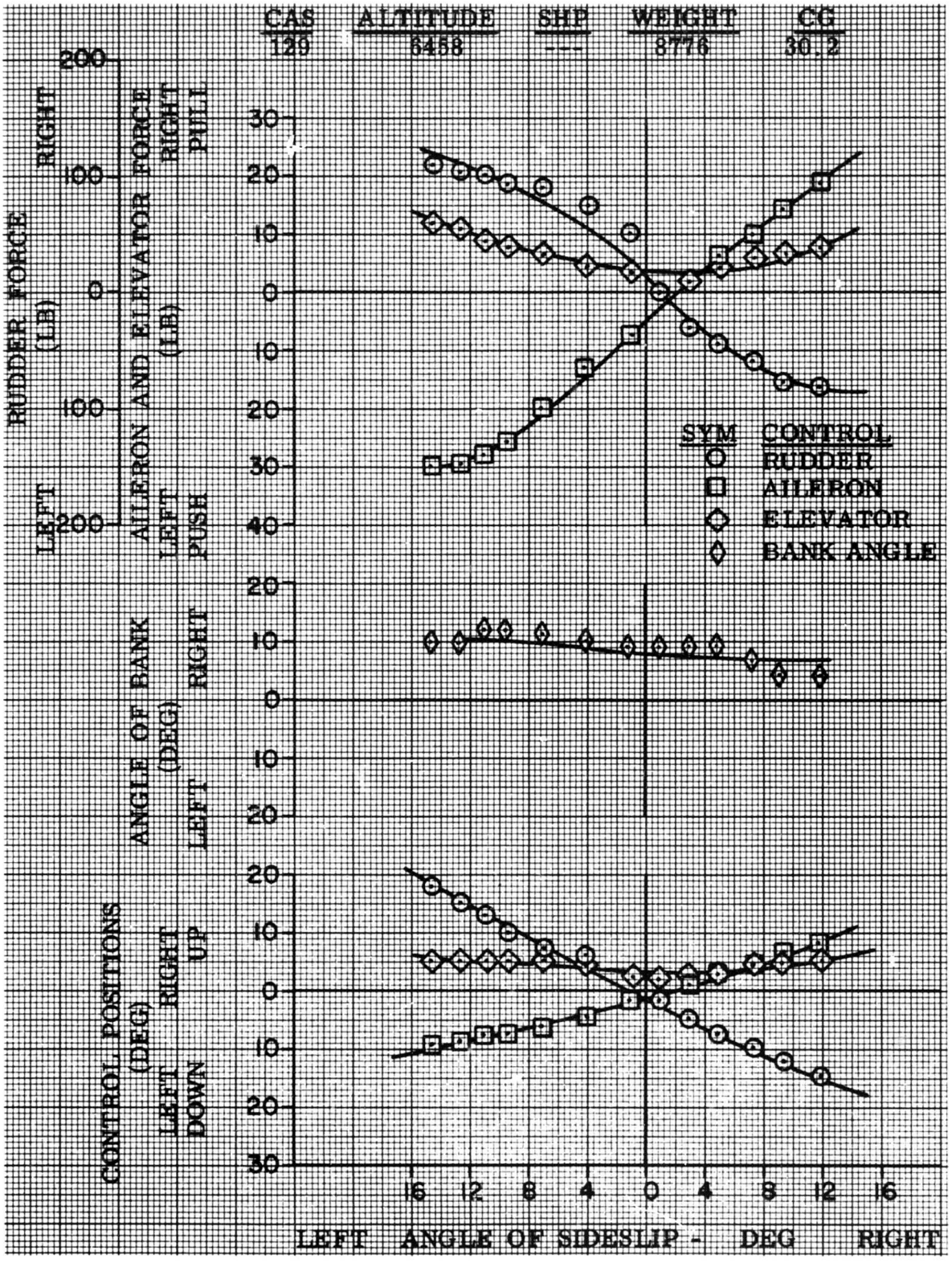


Figure 47. Static Directional Stability - Skid Characteristics - Glide Configuration.

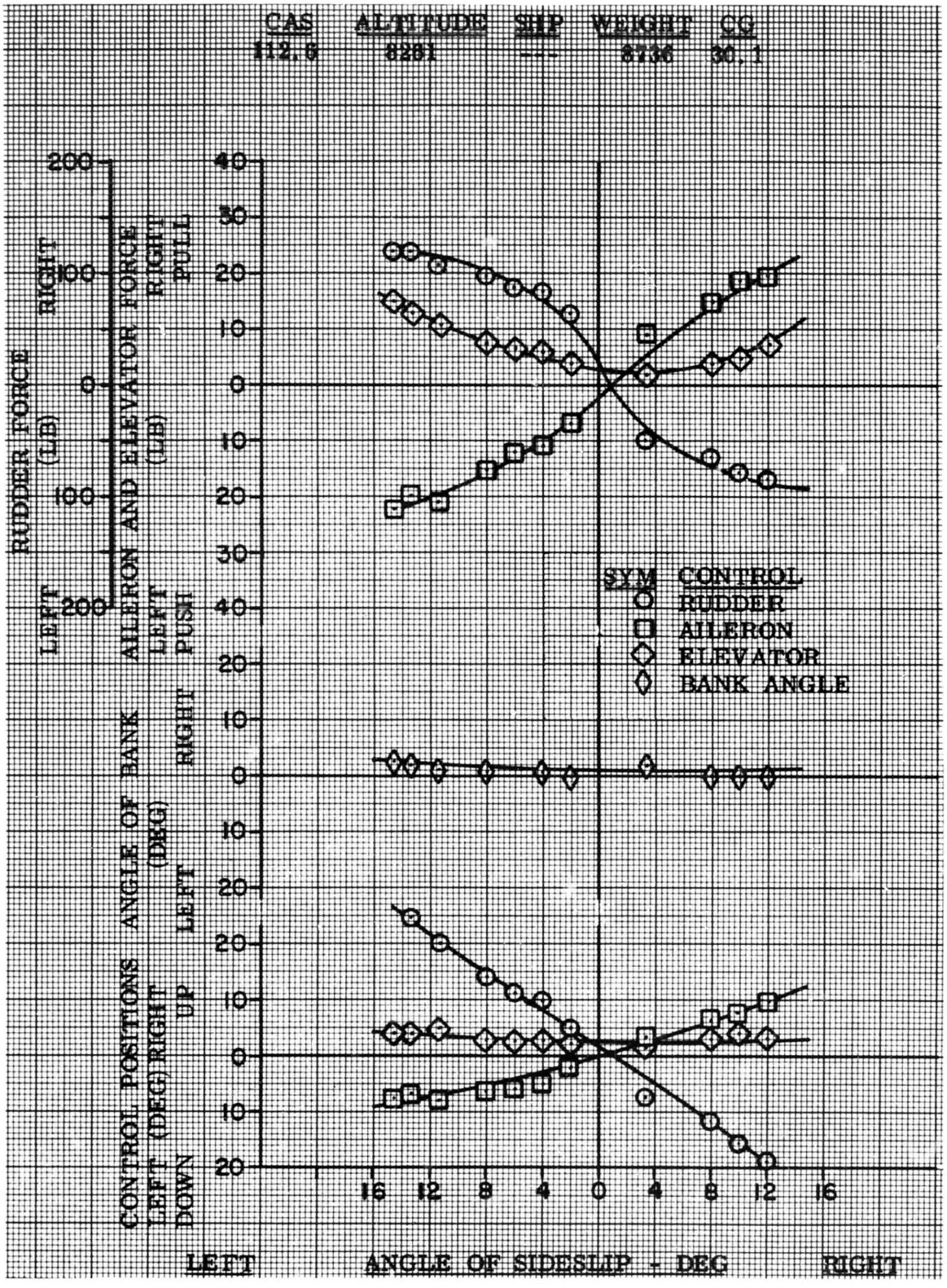


Figure 48. Static Directional Stability - Skid Characteristics - Landing Configuration.

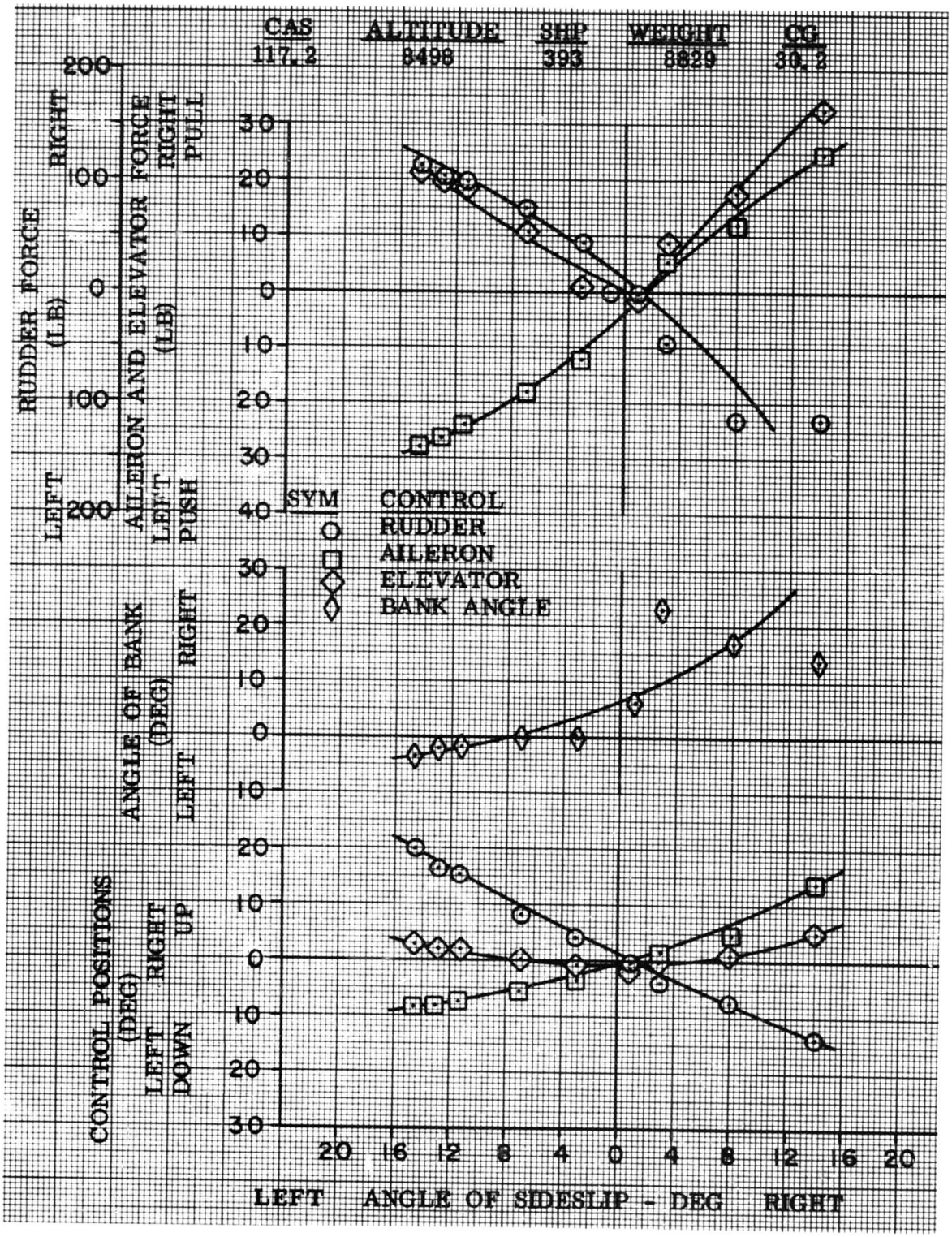


Figure 49. Static Directional Stability - Sideslip Characteristics - Climb Configuration.

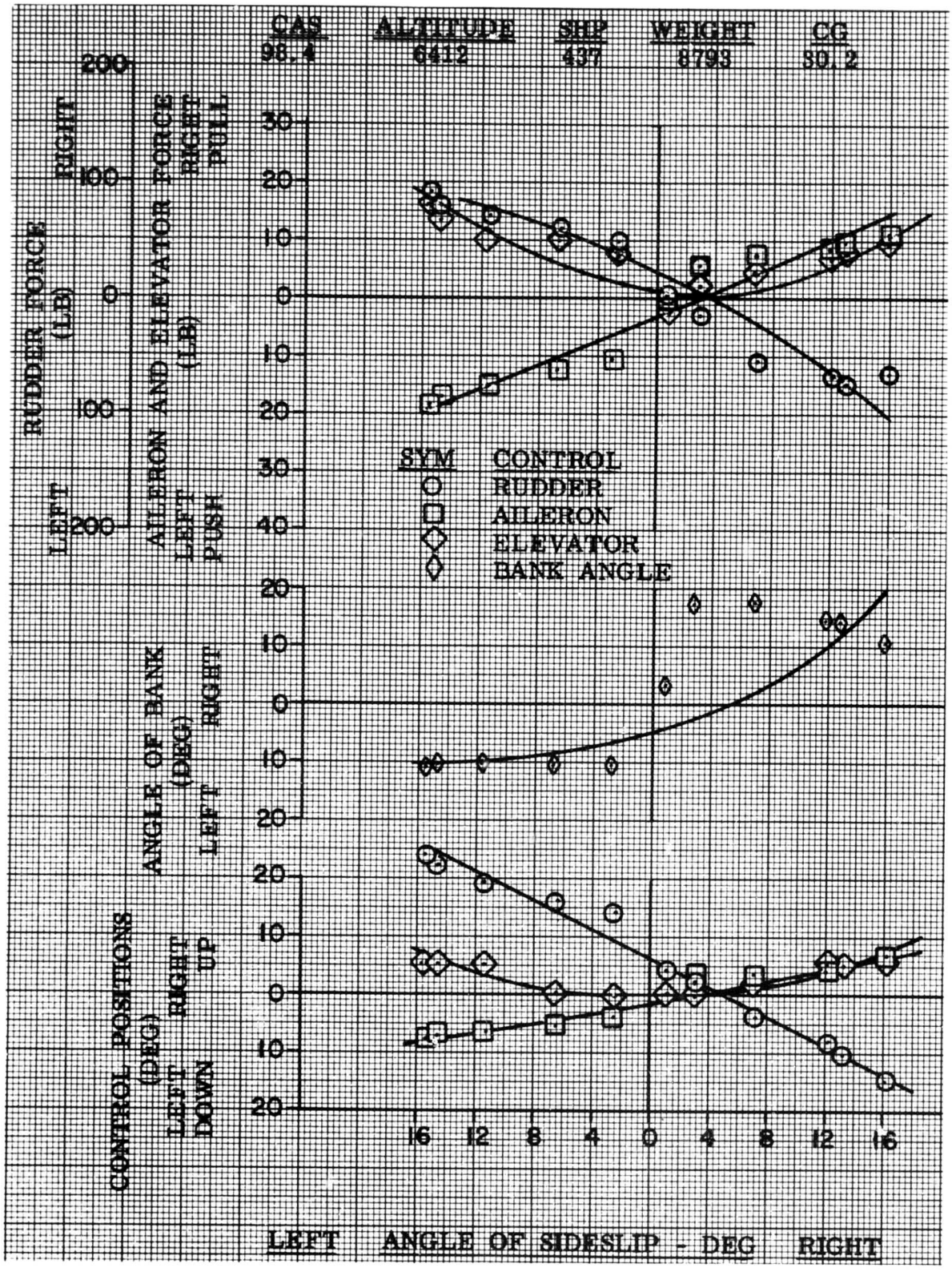


Figure 50. Static Directional Stability - Sideslip Characteristics - Wave-Off Configuration.

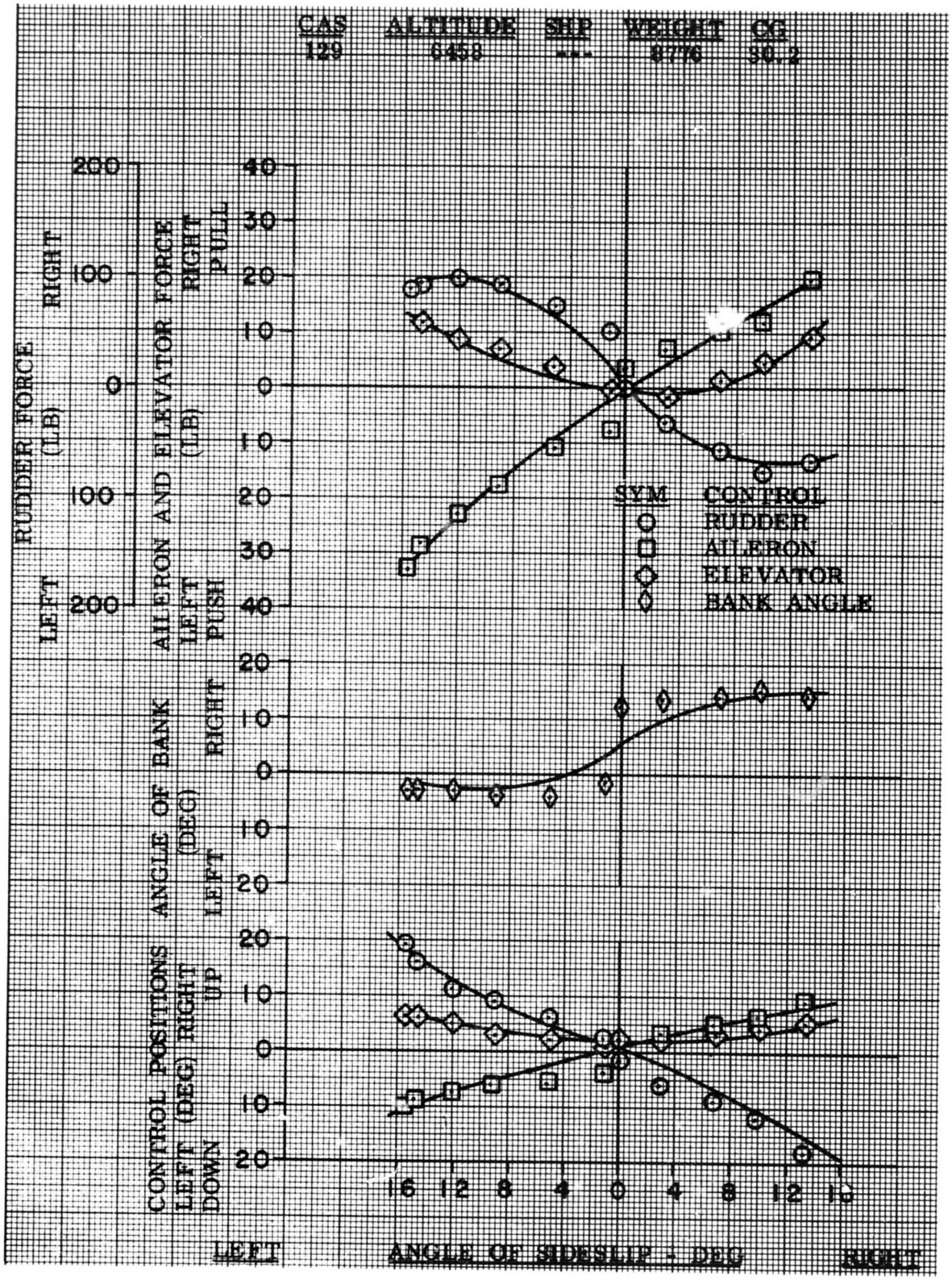


Figure 51. Static Directional Stability - Sideslip Characteristics - Glide Configuration.

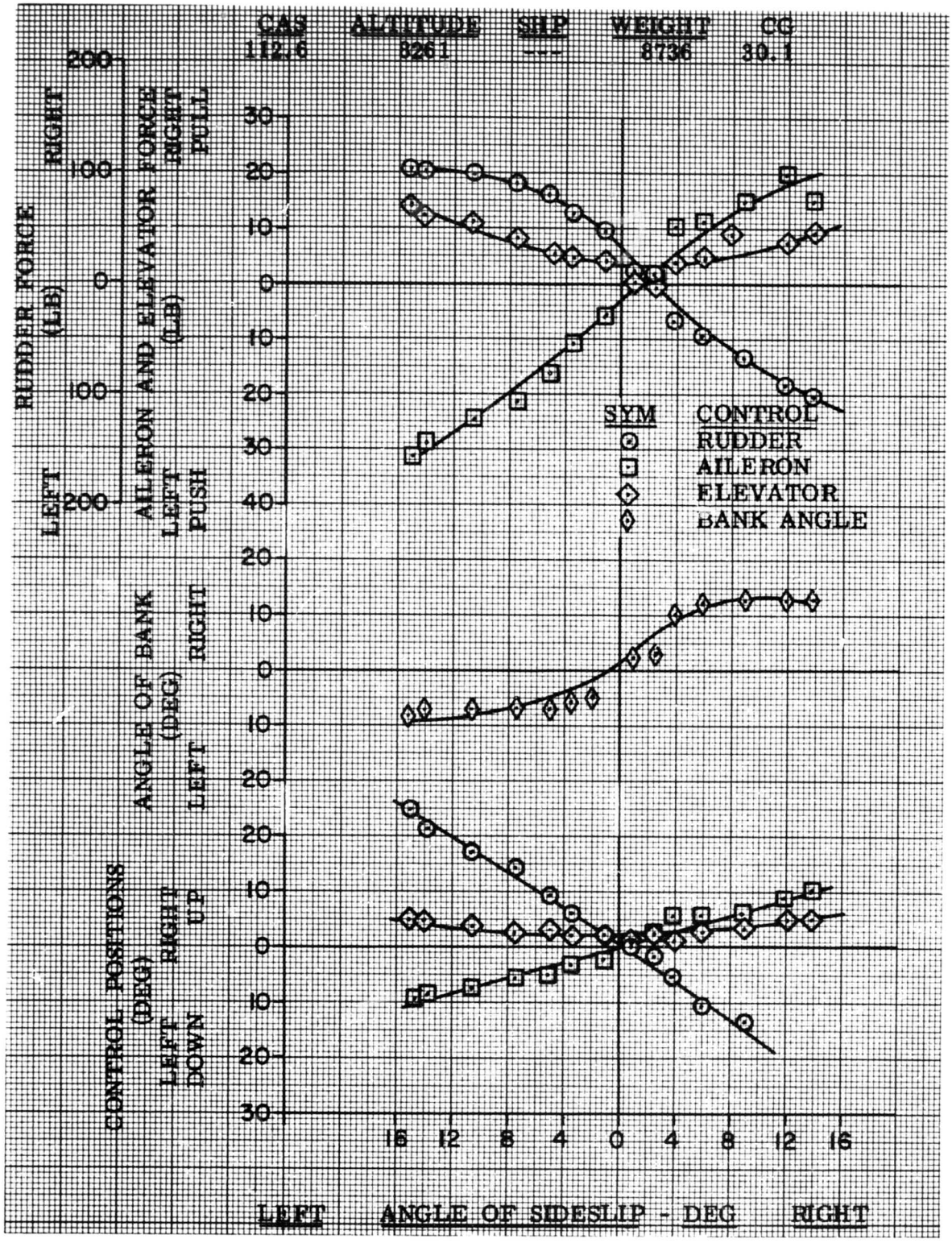


Figure 52. Static Directional Stability - Sideslip Characteristics - Landing Configuration.

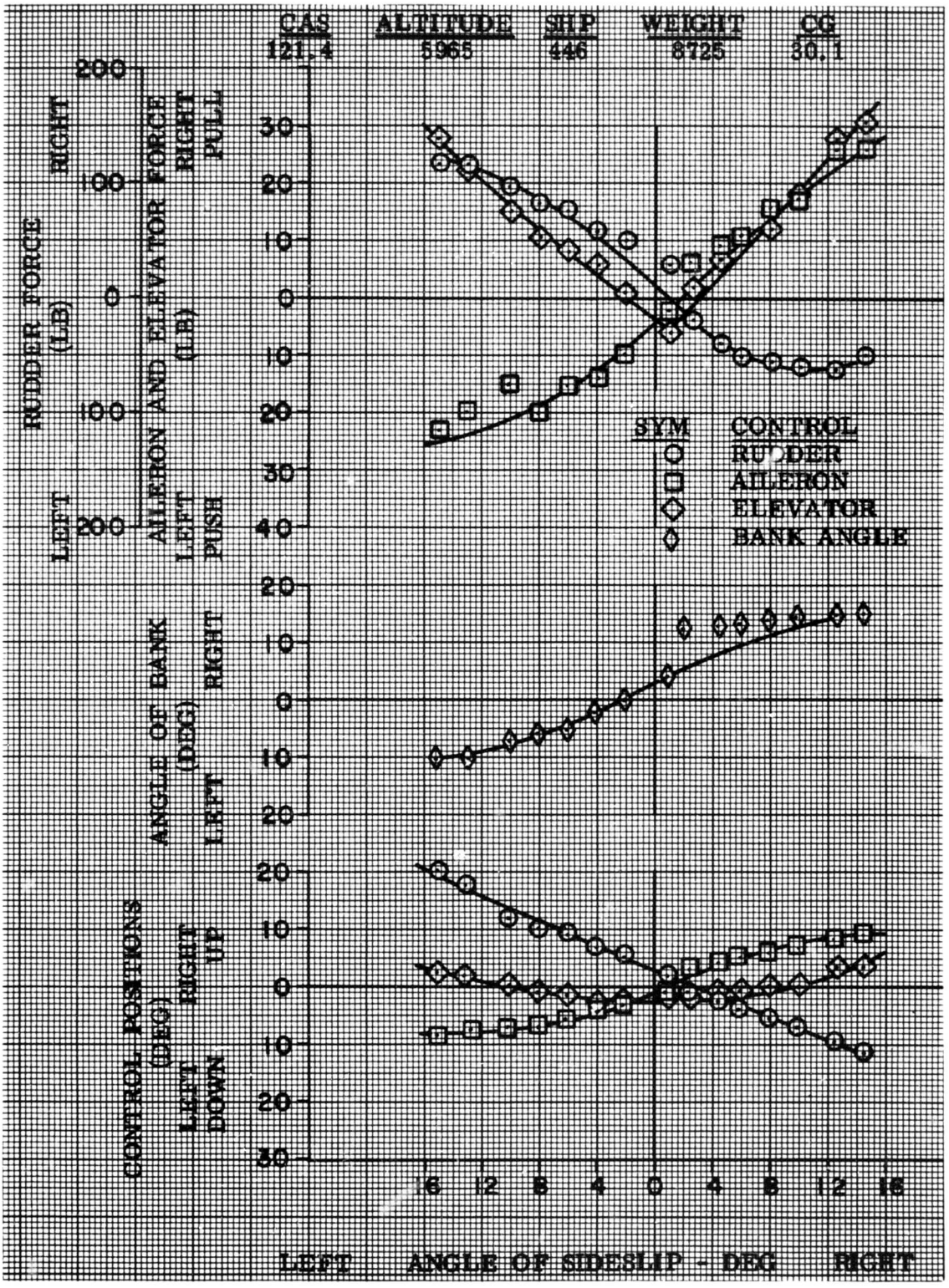


Figure 53. Static Directional Stability - Rudder Lock - Climb Configuration.

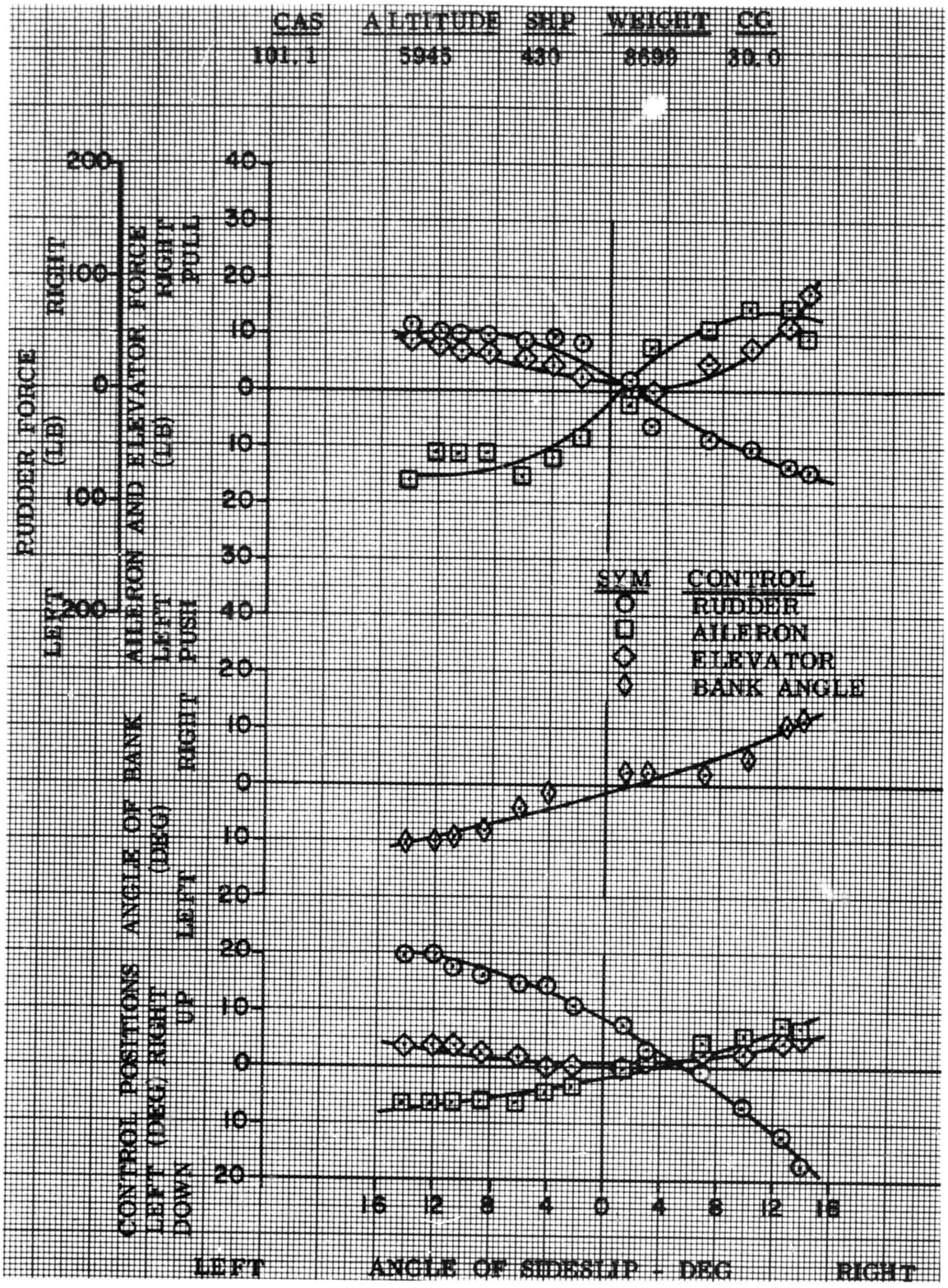


Figure 54. Static Directional Stability - Rudder Lock - Wave-Off Configuration.

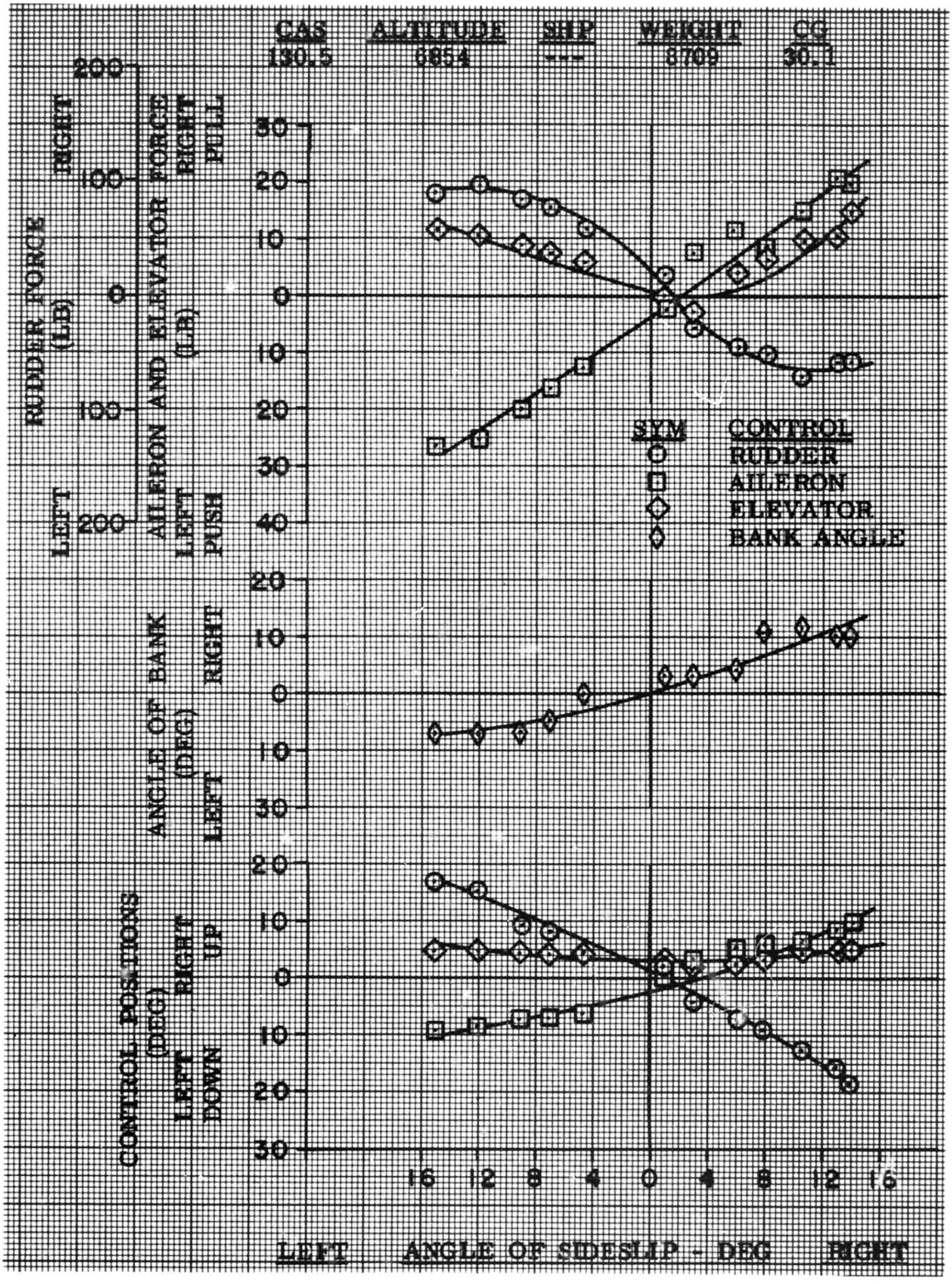


Figure 55. Static Directional Stability - Rudder Lock - Glide Configuration.

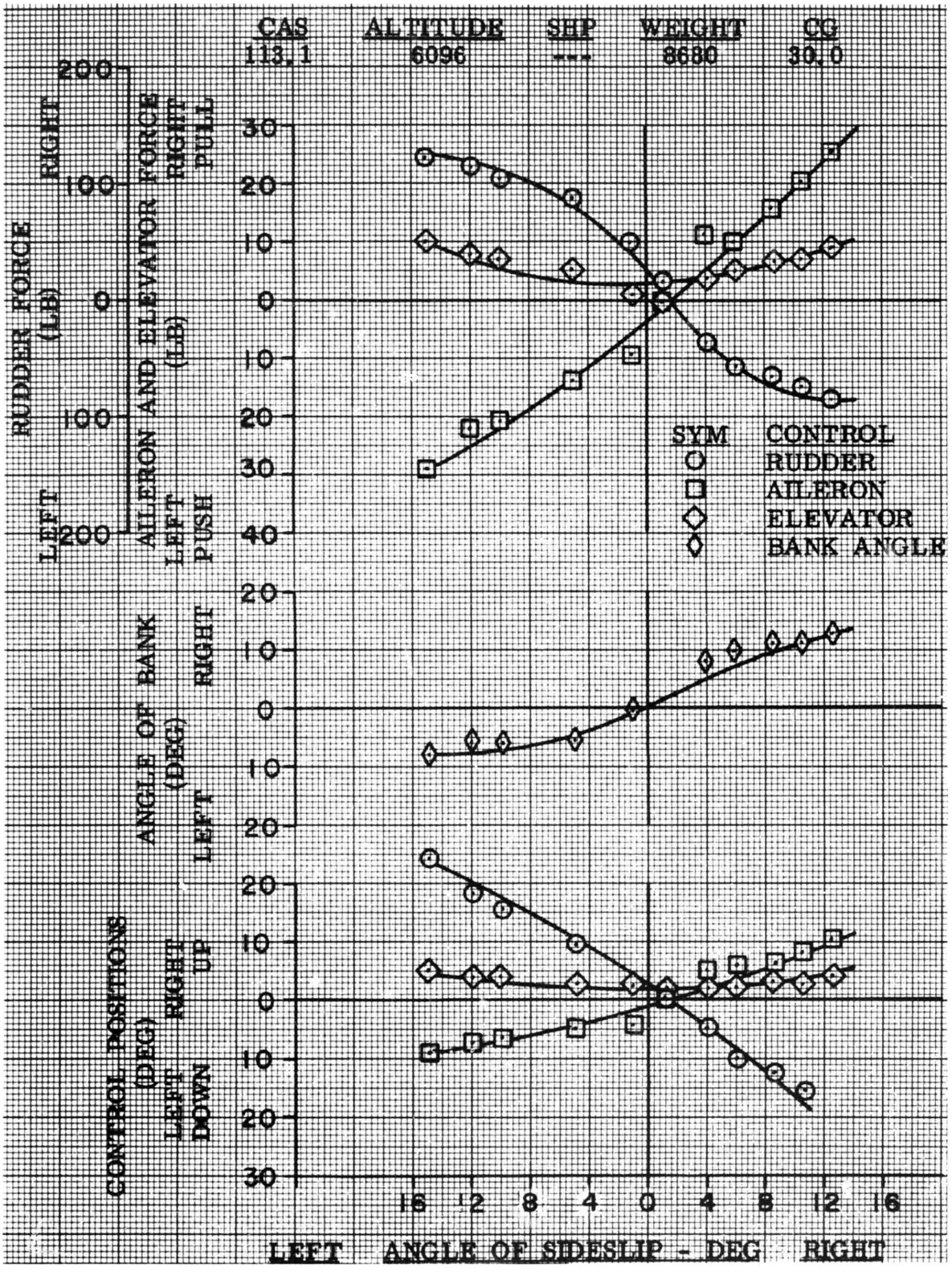


Figure 56. Static Directional Stability - Rudder Lock - Landing Configuration.

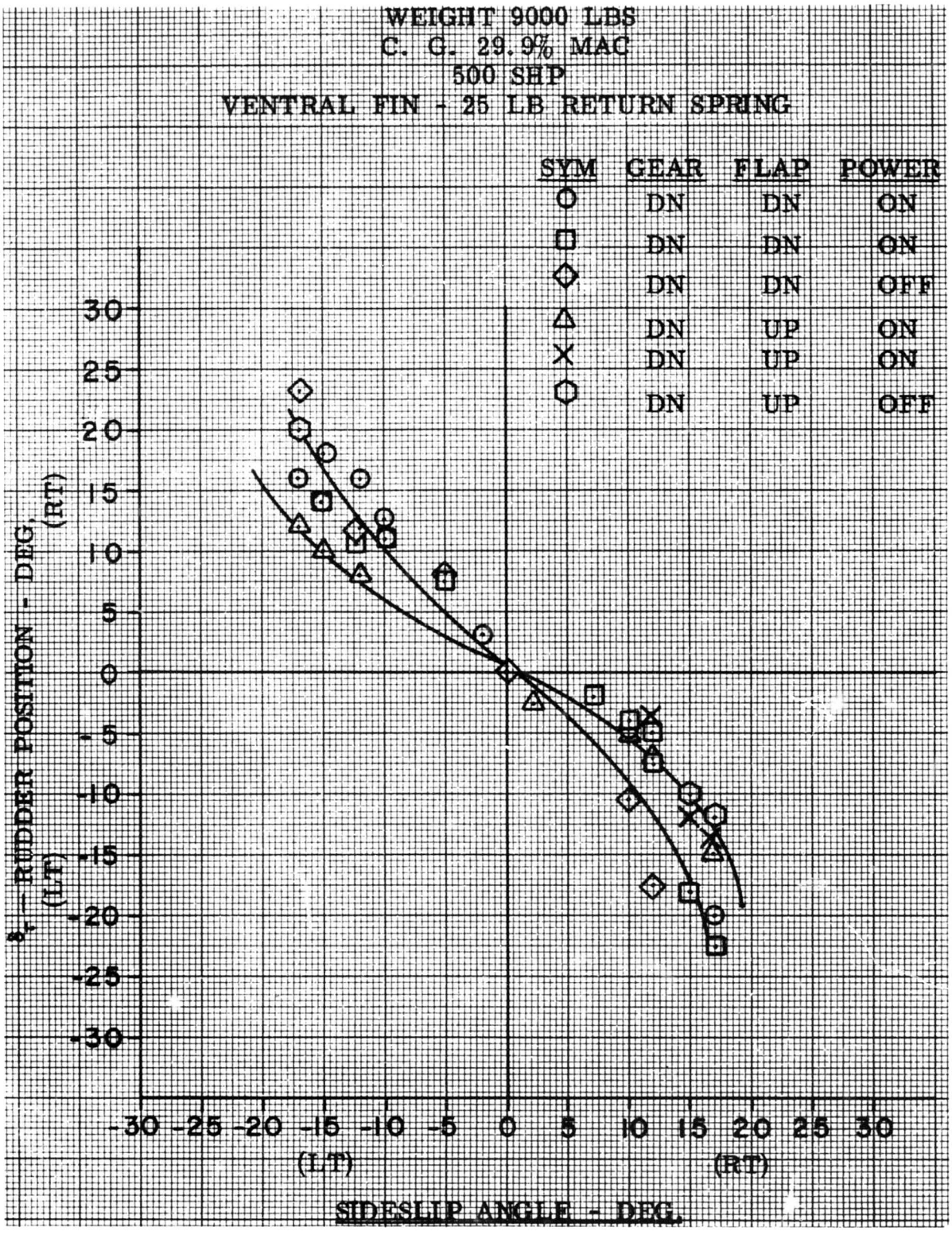


Figure 57. Effect of Ventral Fin.

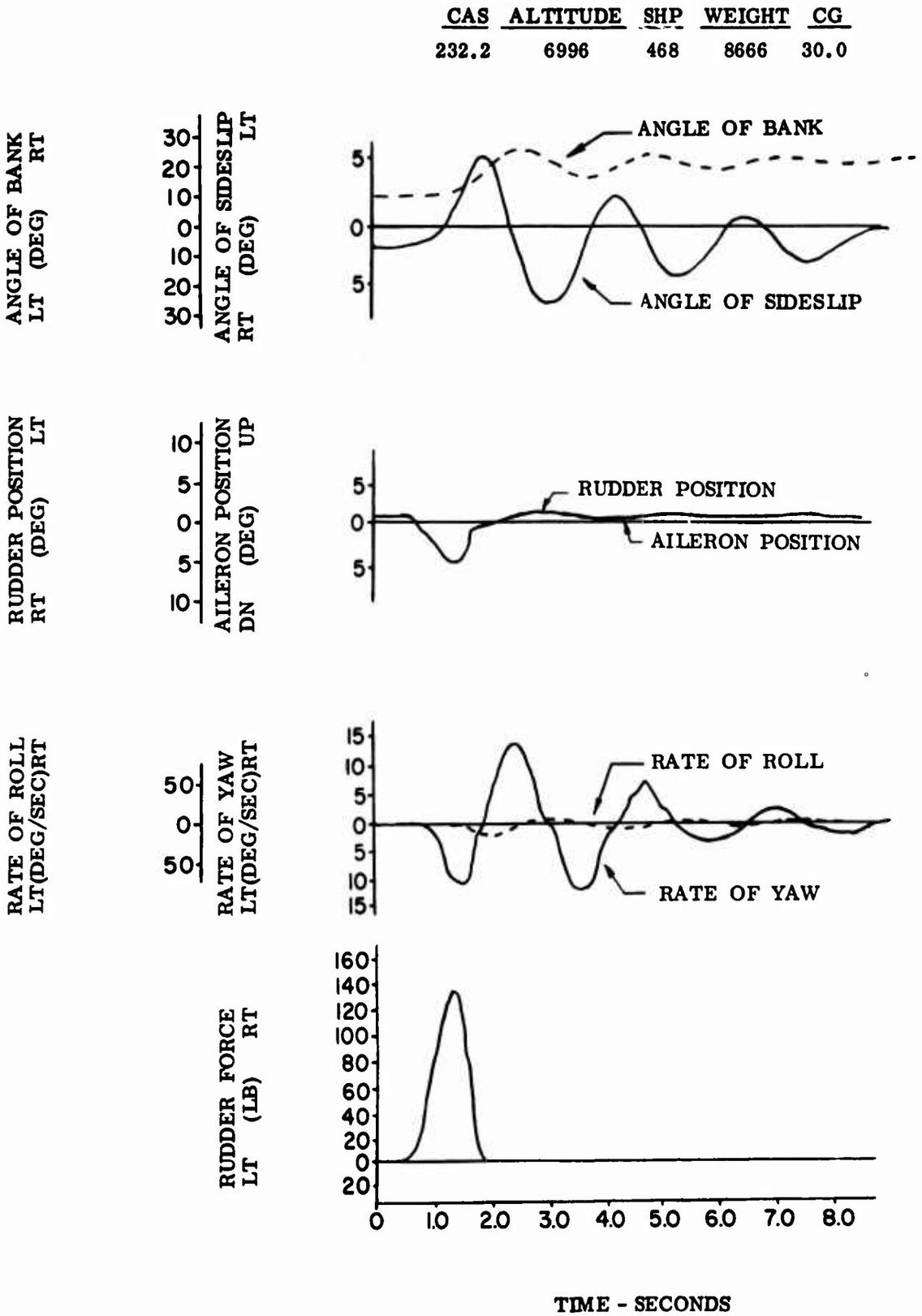


Figure 58. Dynamic Lateral-Directional Stability - Cruise Configuration - Rudder Kick.

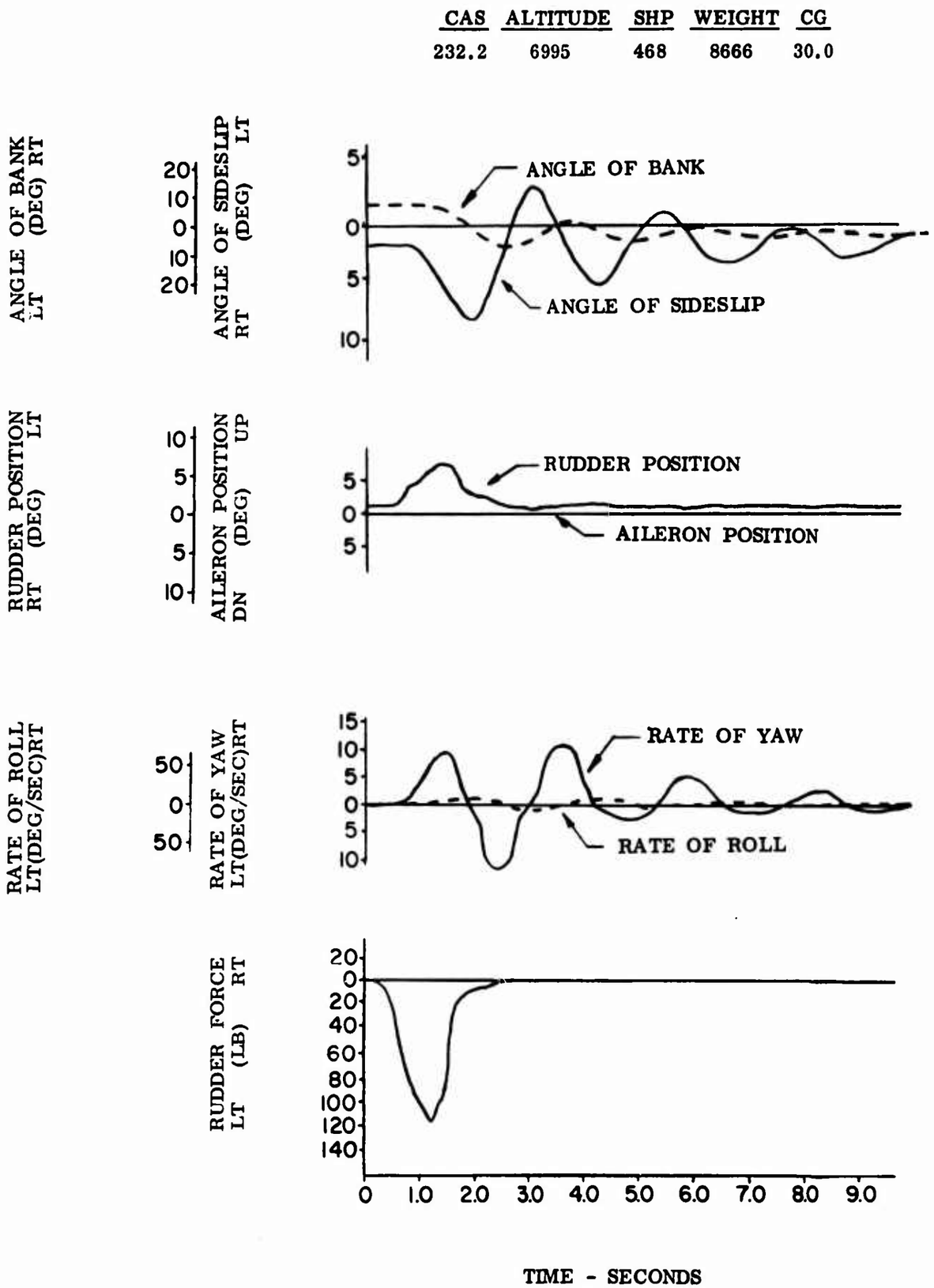


Figure 59. Dynamic Lateral-Directional Stability - Cruise Configuration - Rudder Kick.

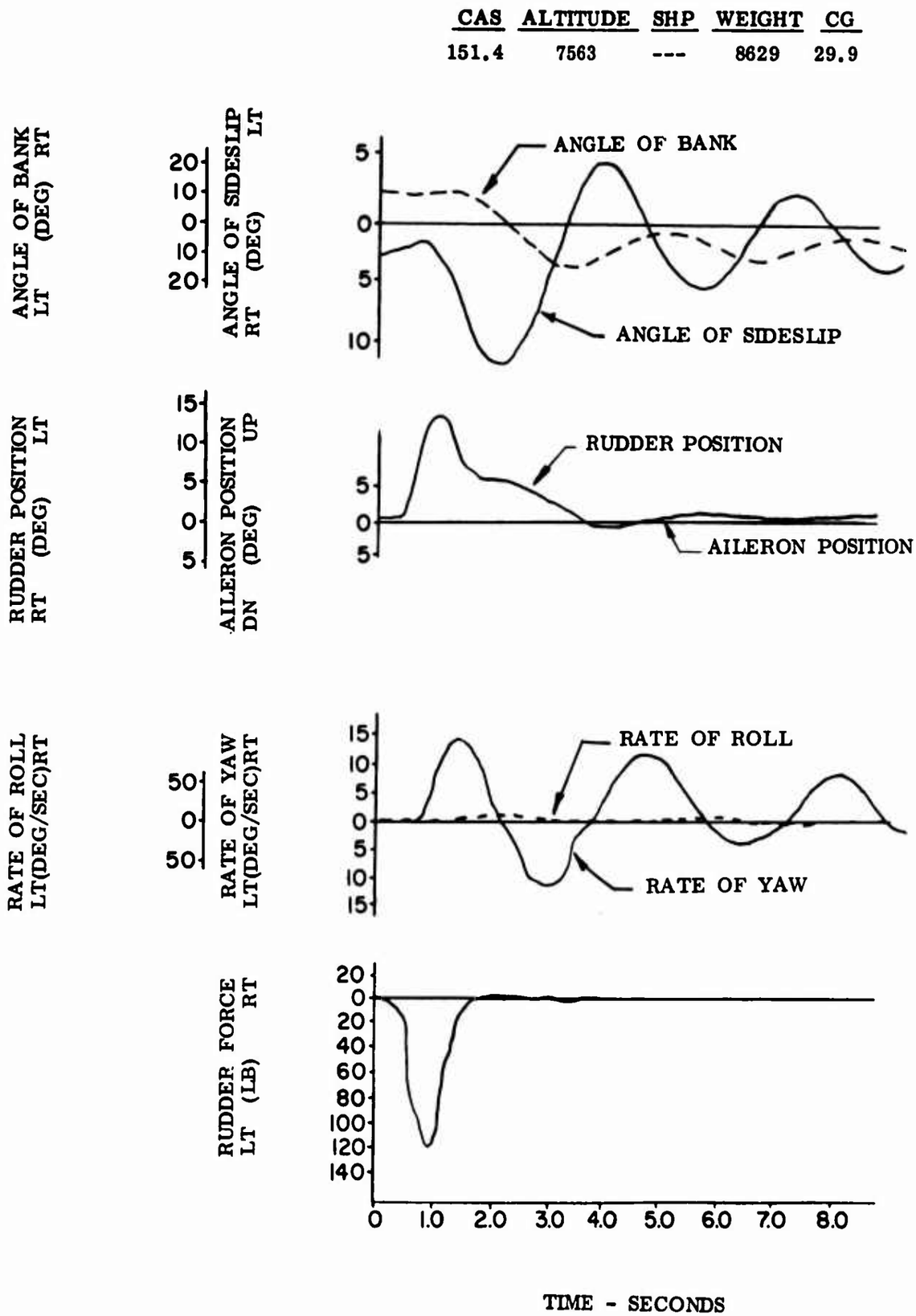


Figure 60. Dynamic Lateral-Directional Stability - Landing Configuration - Rudder Kick.

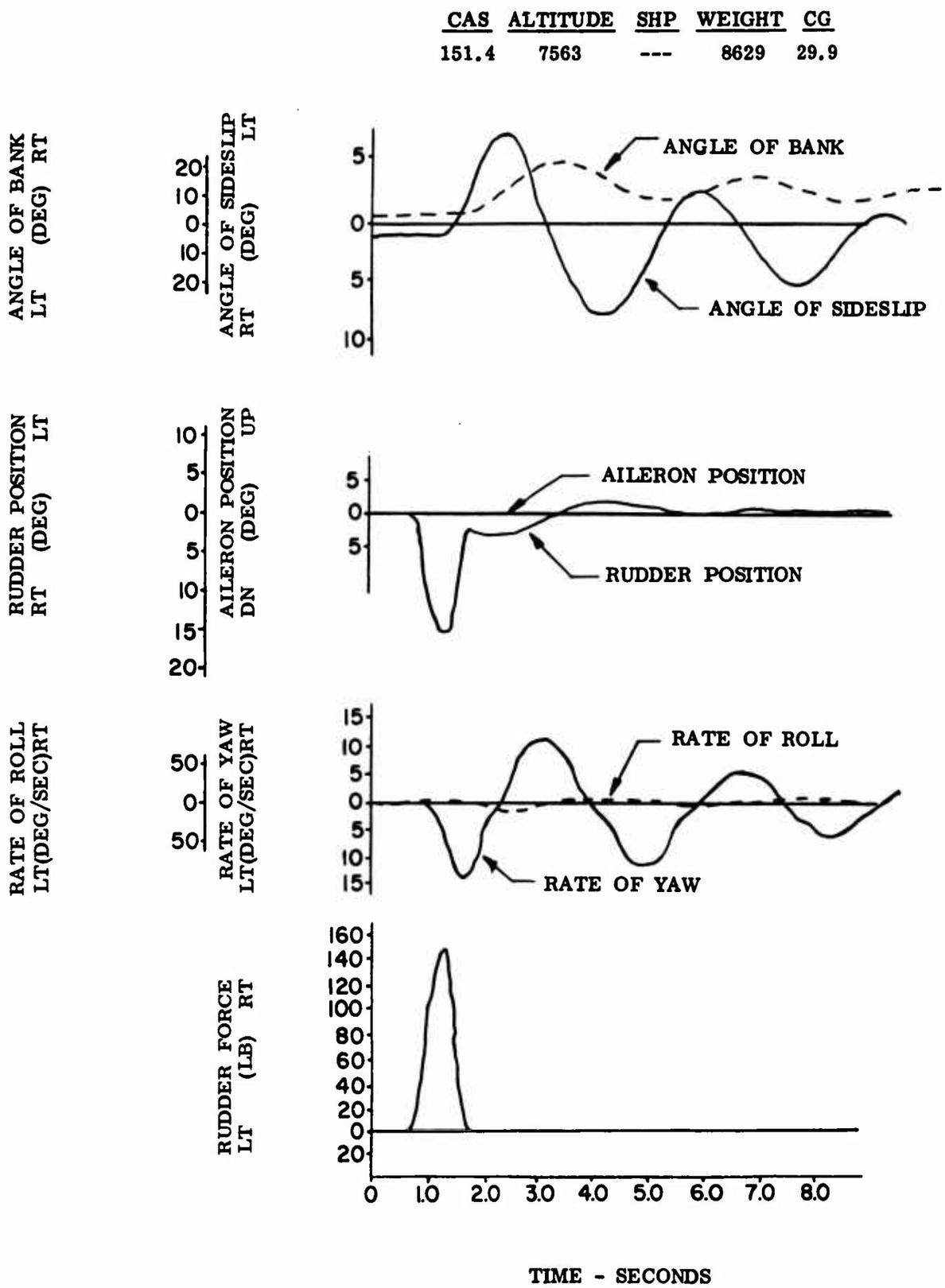


Figure 61. Dynamic Lateral-Directional Stability - Landing Configuration - Rudder Kick.

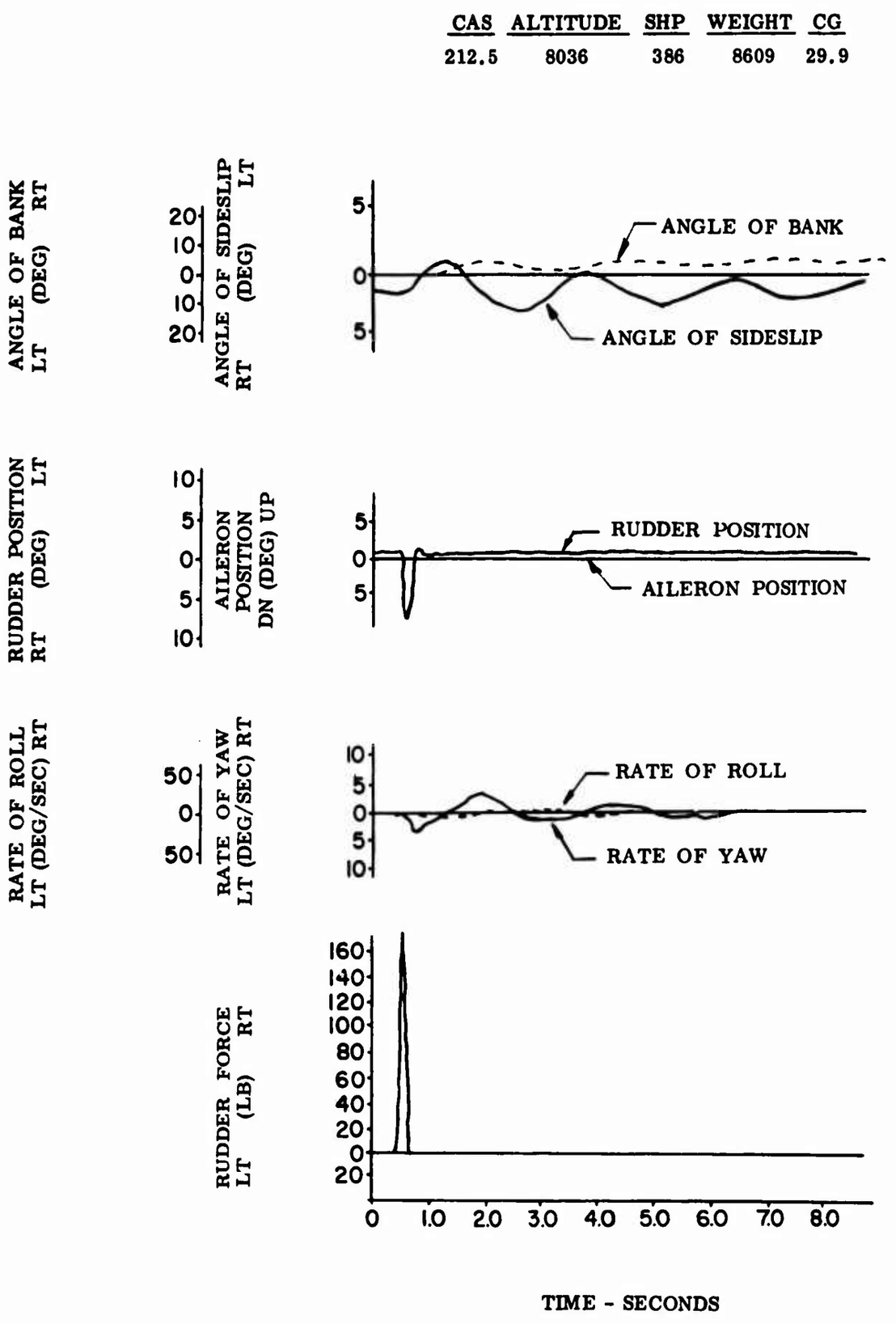


Figure 62. Dynamic Lateral-Directional Stability - Cruise Configuration - Rudder Bump.

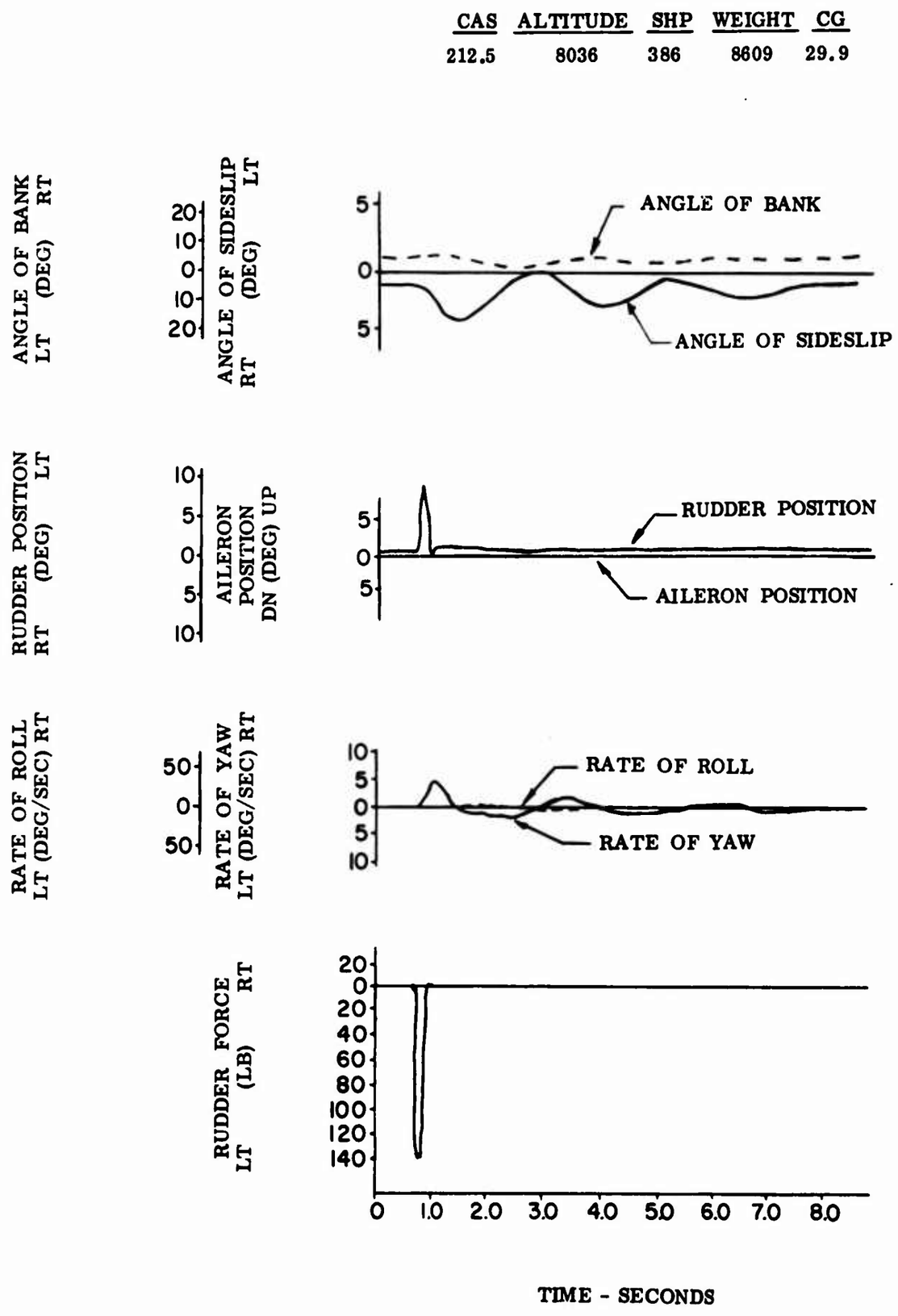


Figure 63. Dynamic Lateral-Directional Stability - Cruise Configuration - Rudder Bump.

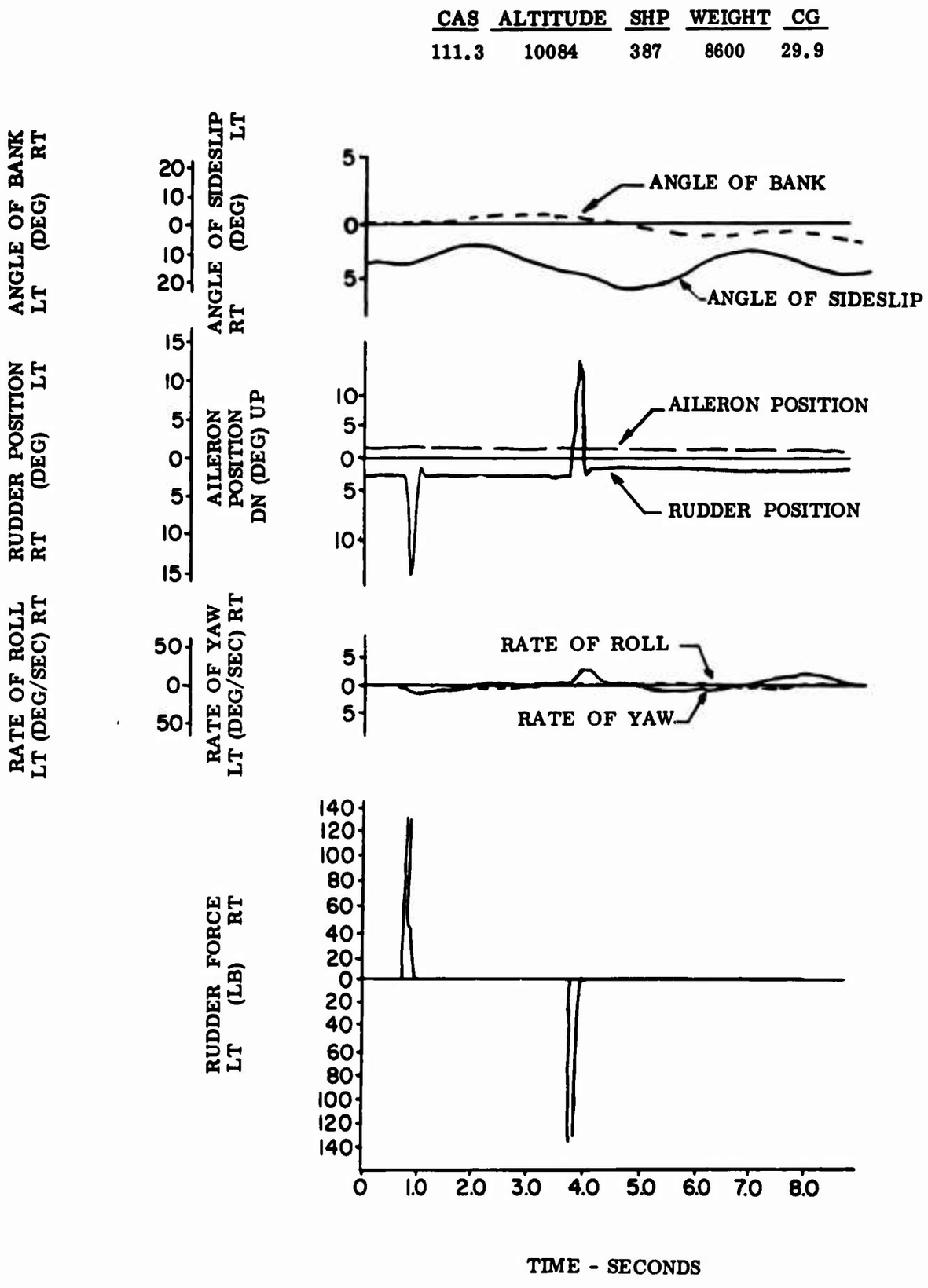


Figure 64. Dynamic Lateral-Directional Stability - Wave-Off Configuration - Rudder Bump.

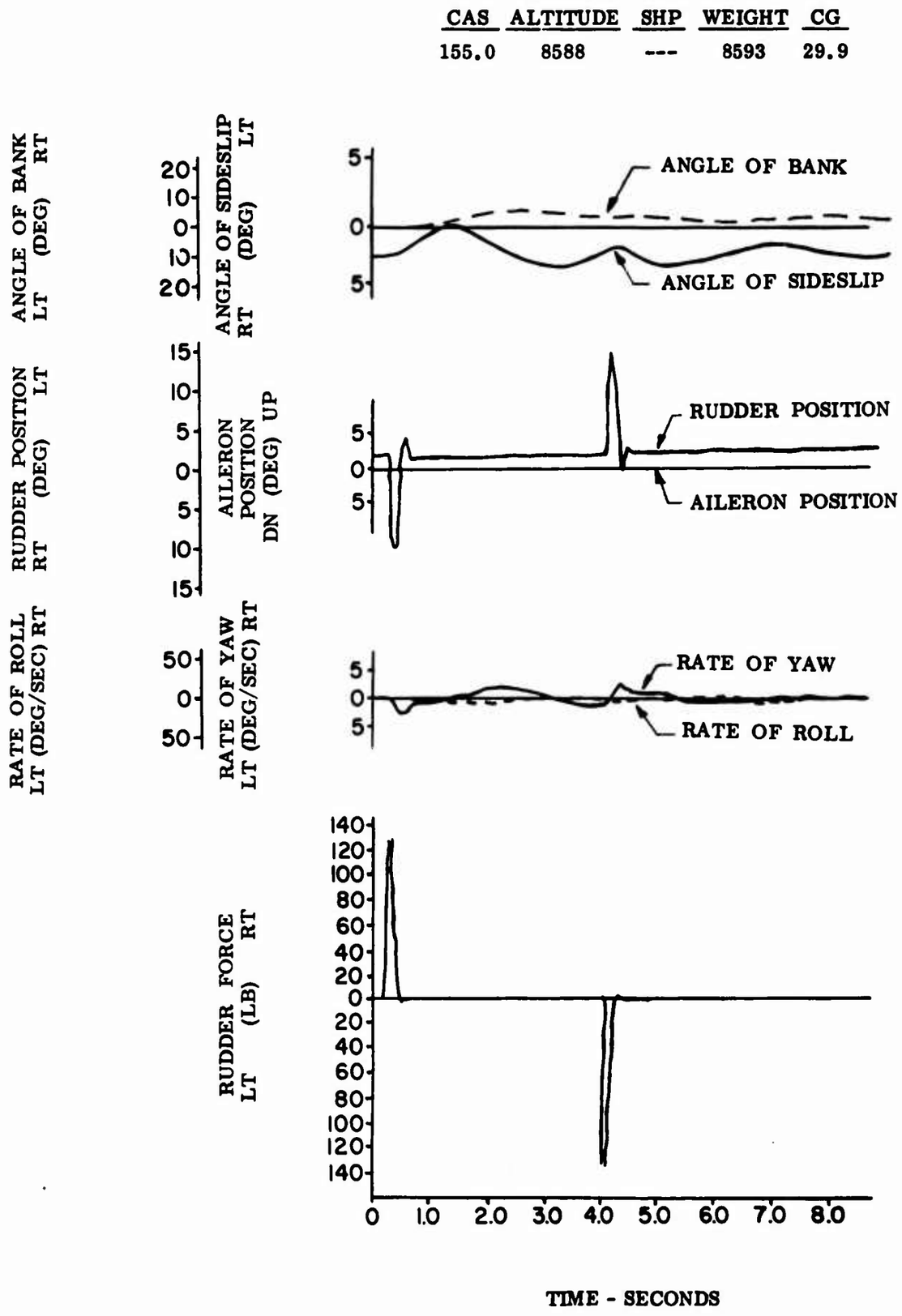


Figure 65. Dynamic Lateral-Directional Stability - Glide Configuration - Rudder Bump.

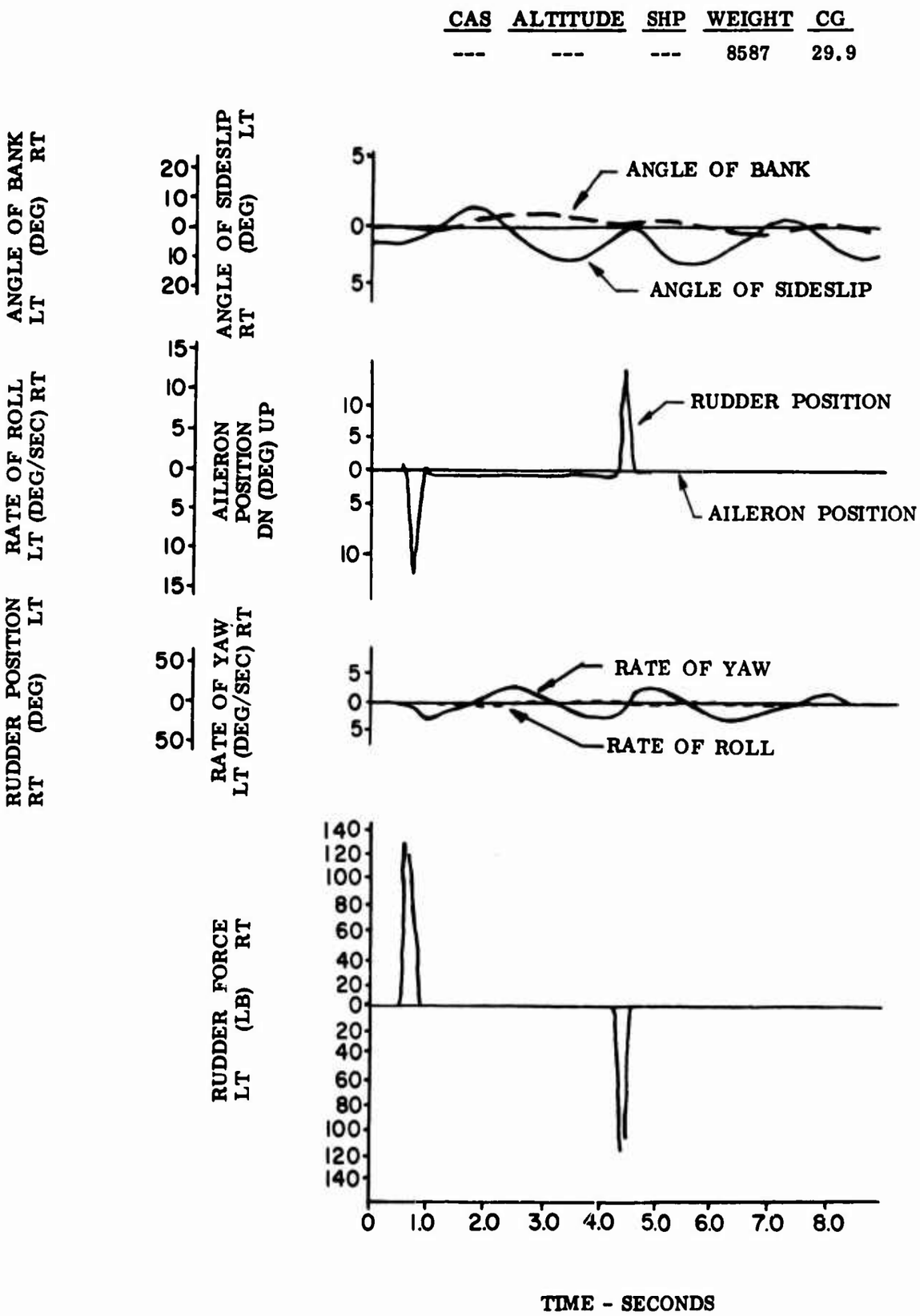


Figure 66. Dynamic Lateral-Directional Stability - Landing Configuration - Rudder Bump.

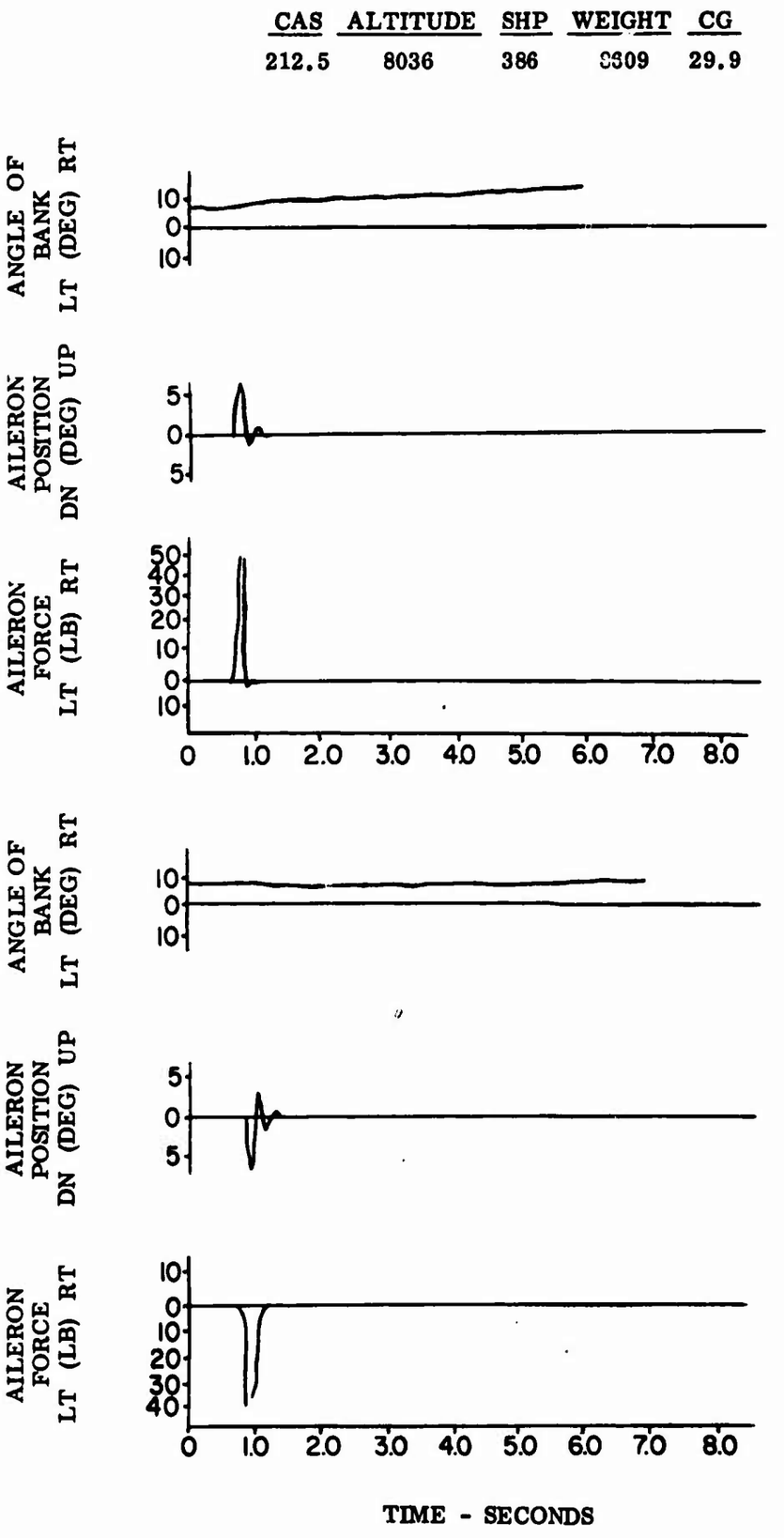


Figure 67. Dynamic Lateral-Directional Stability - Cruise Configuration - Aileron Bump.

CAS	ALTITUDE	SHP	WEIGHT	CG
111.3	10084	387	8600	29.9

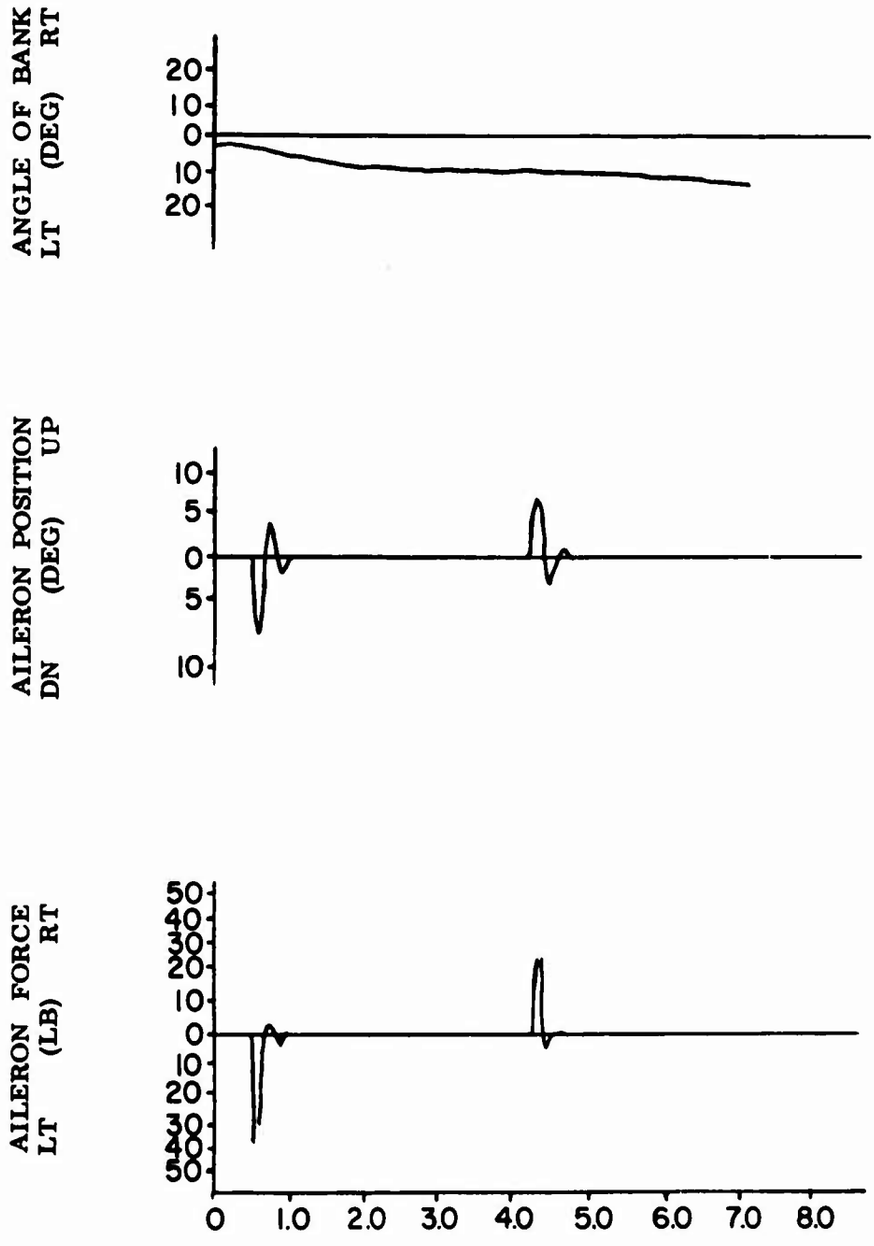


Figure 68. Dynamic Lateral-Directional Stability - Wave-Off Configuration - Aileron Bump.

CAS	ALTITUDE	SHP	WEIGHT	CG
155	8588	---	8593	29.9

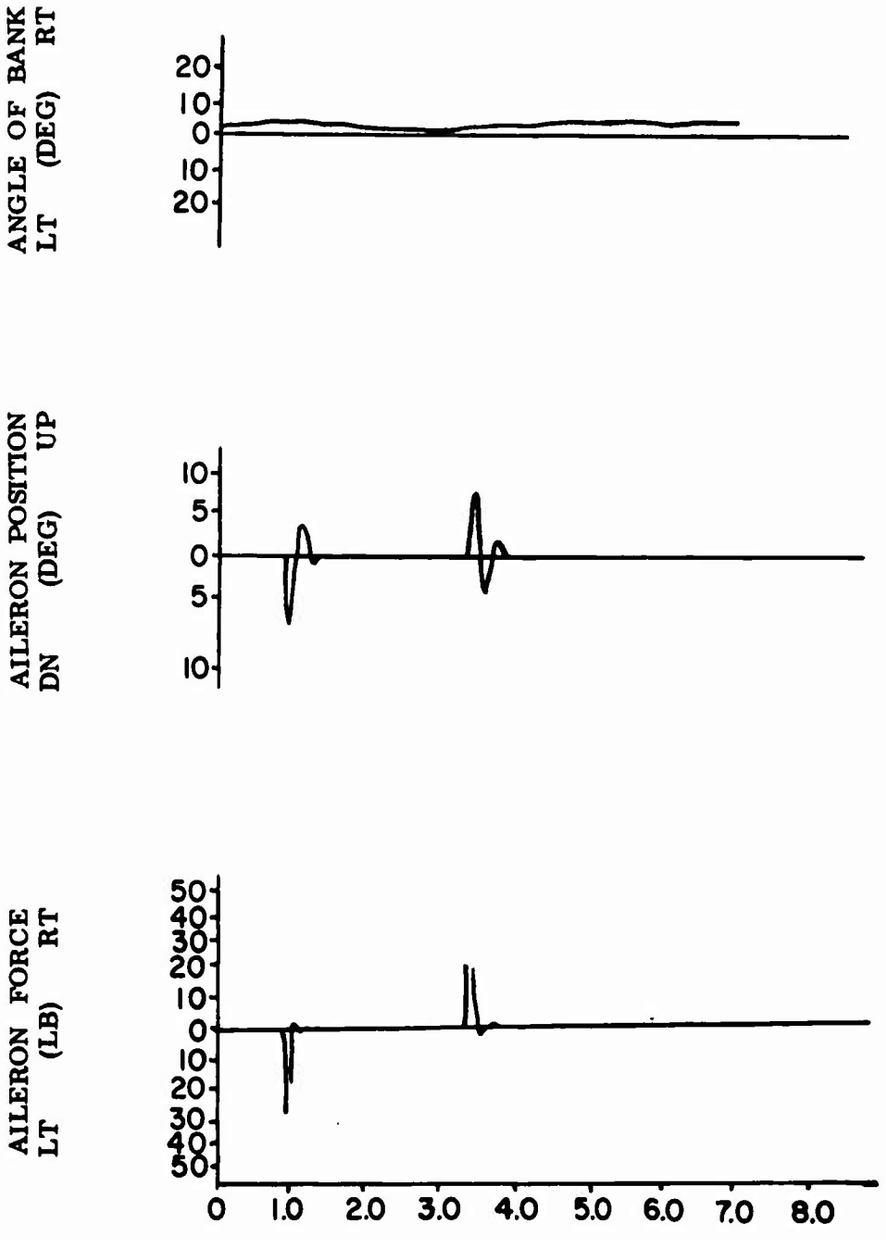


Figure 69. Dynamic Lateral-Directional Stability - Glide Configuration - Aileron Bump.

CAS	ALTITUDE	SHP	WEIGHT	CG
---	---	---	8587	29.9

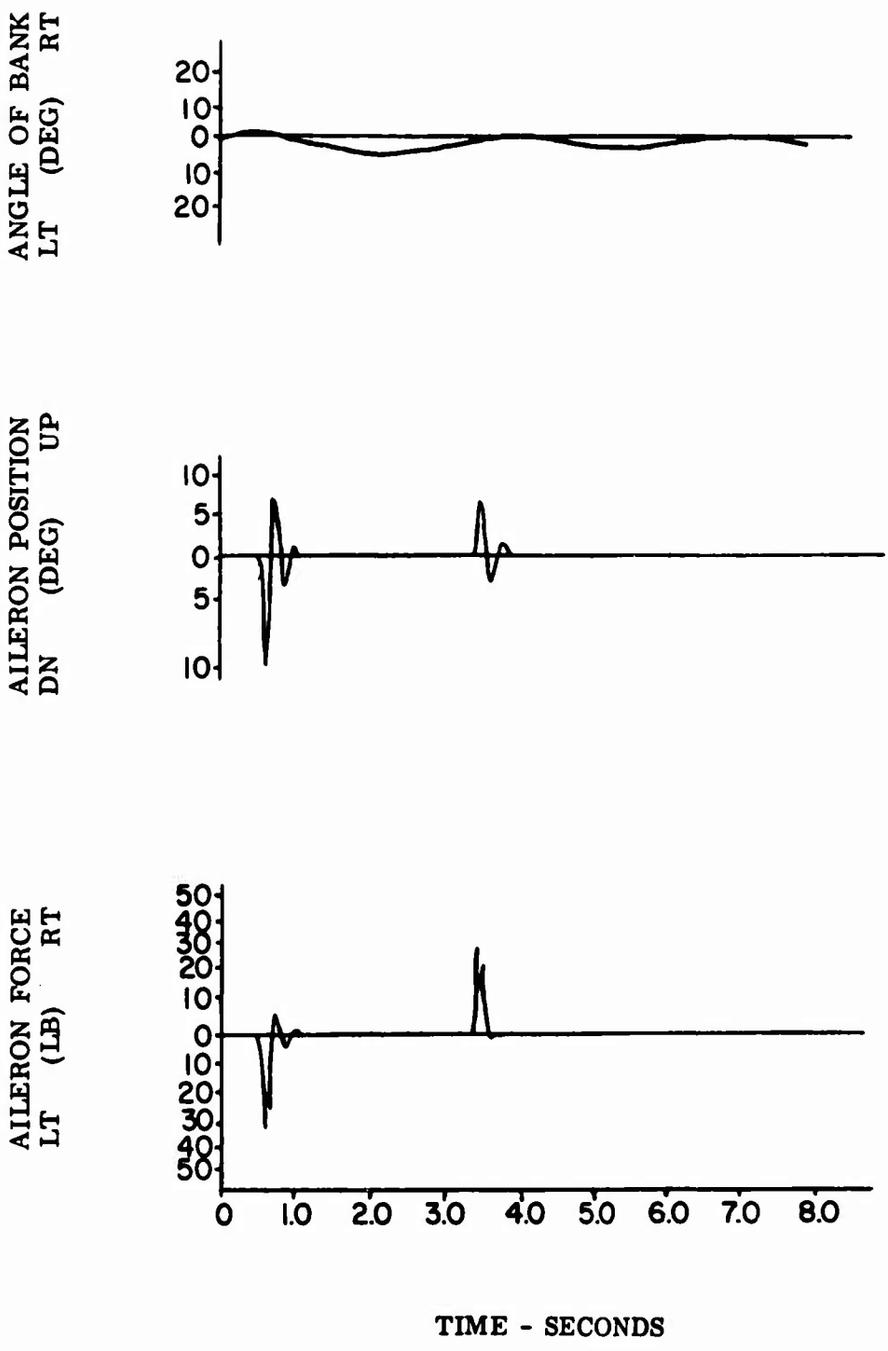


Figure 70. Dynamic Lateral-Directional Stability - Landing Configuration - Aileron Bump.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT.
195.8	6950	321	8921	30.3	59°F

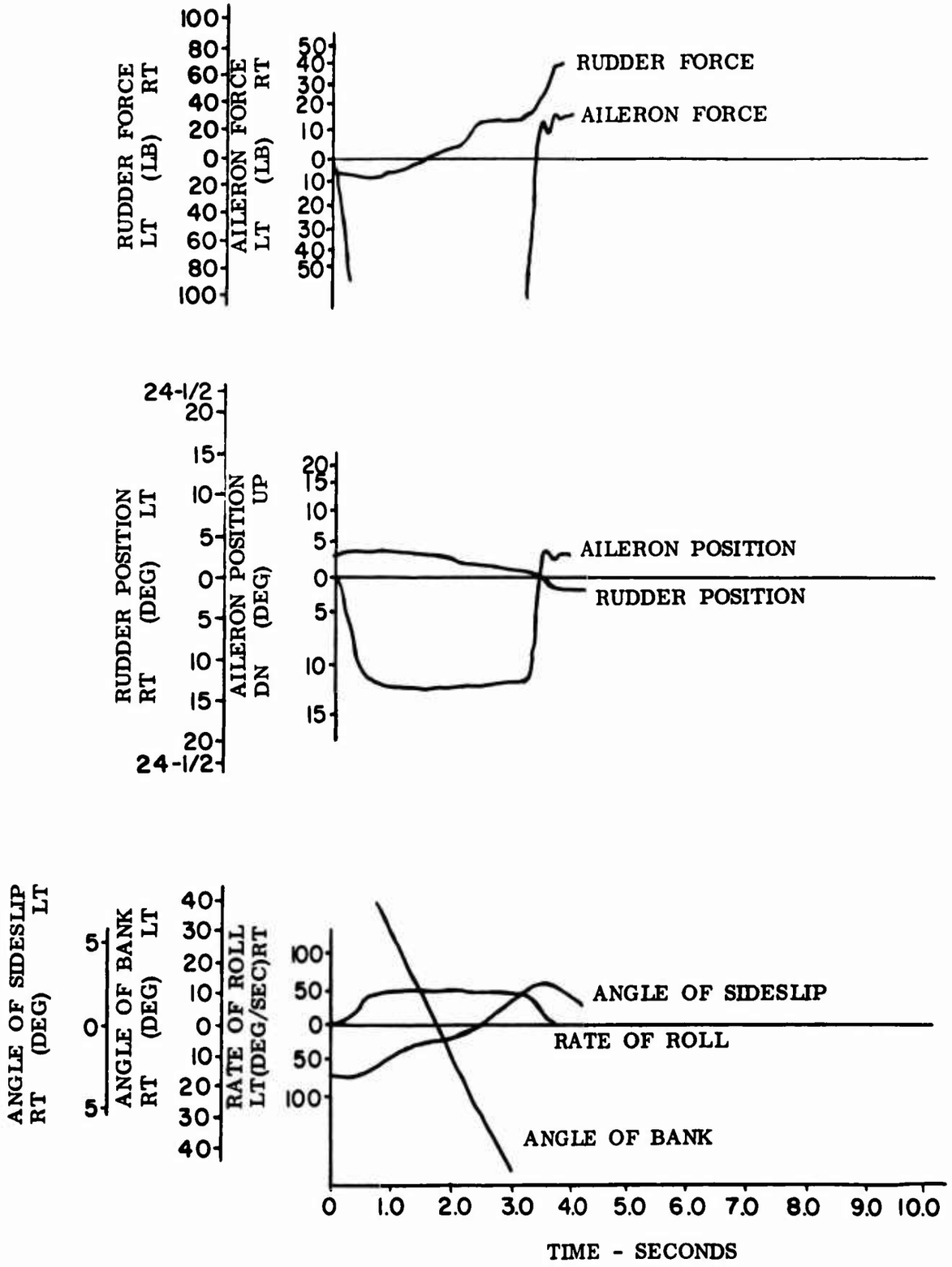


Figure 71. Aileron Rolls - Cruise Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT.
187.9	6940	328	8926	30.3	55.5°F

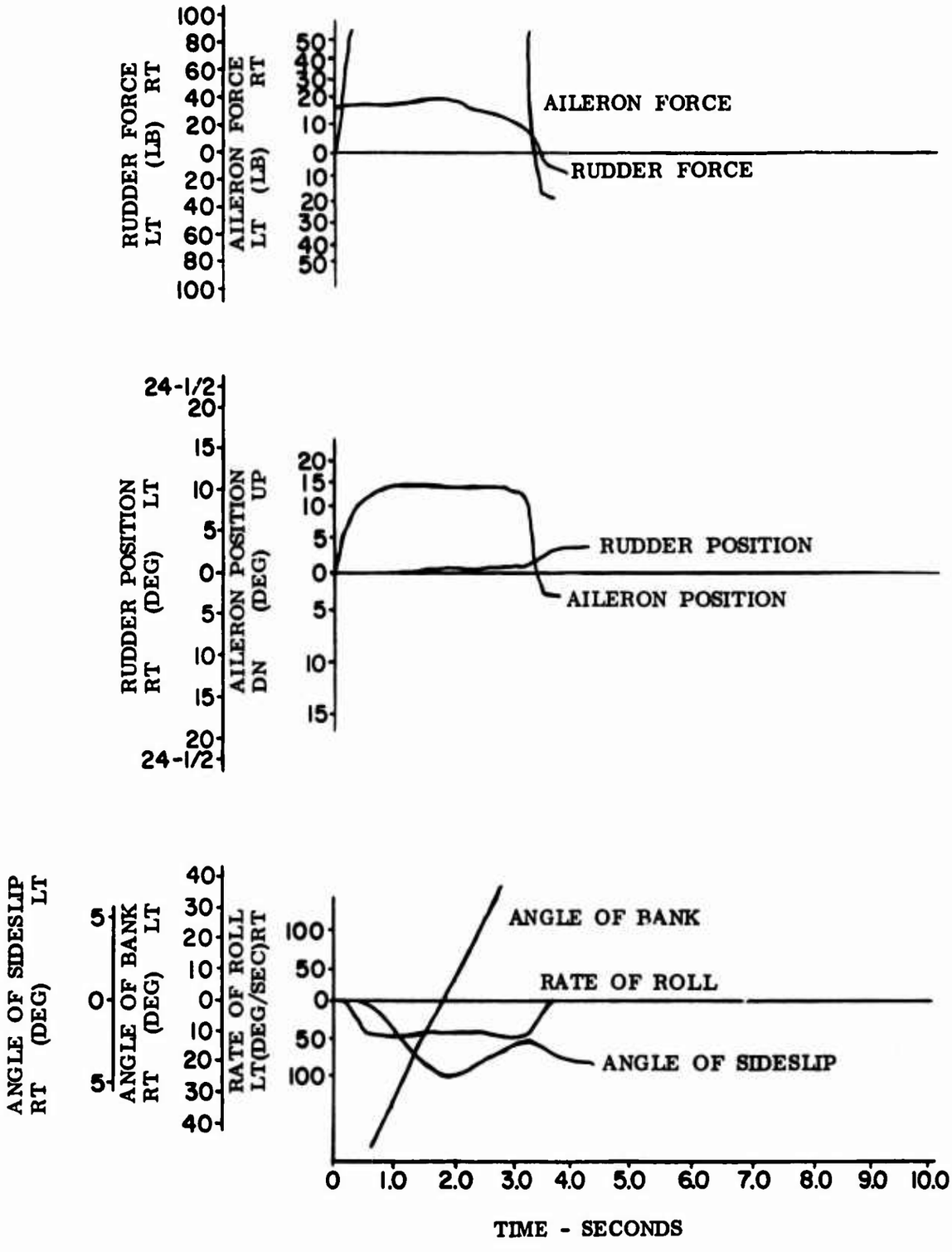


Figure 72. Aileron Rolls - Cruise Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT.
167.6	6789	---	8756	30.1	55.5°F

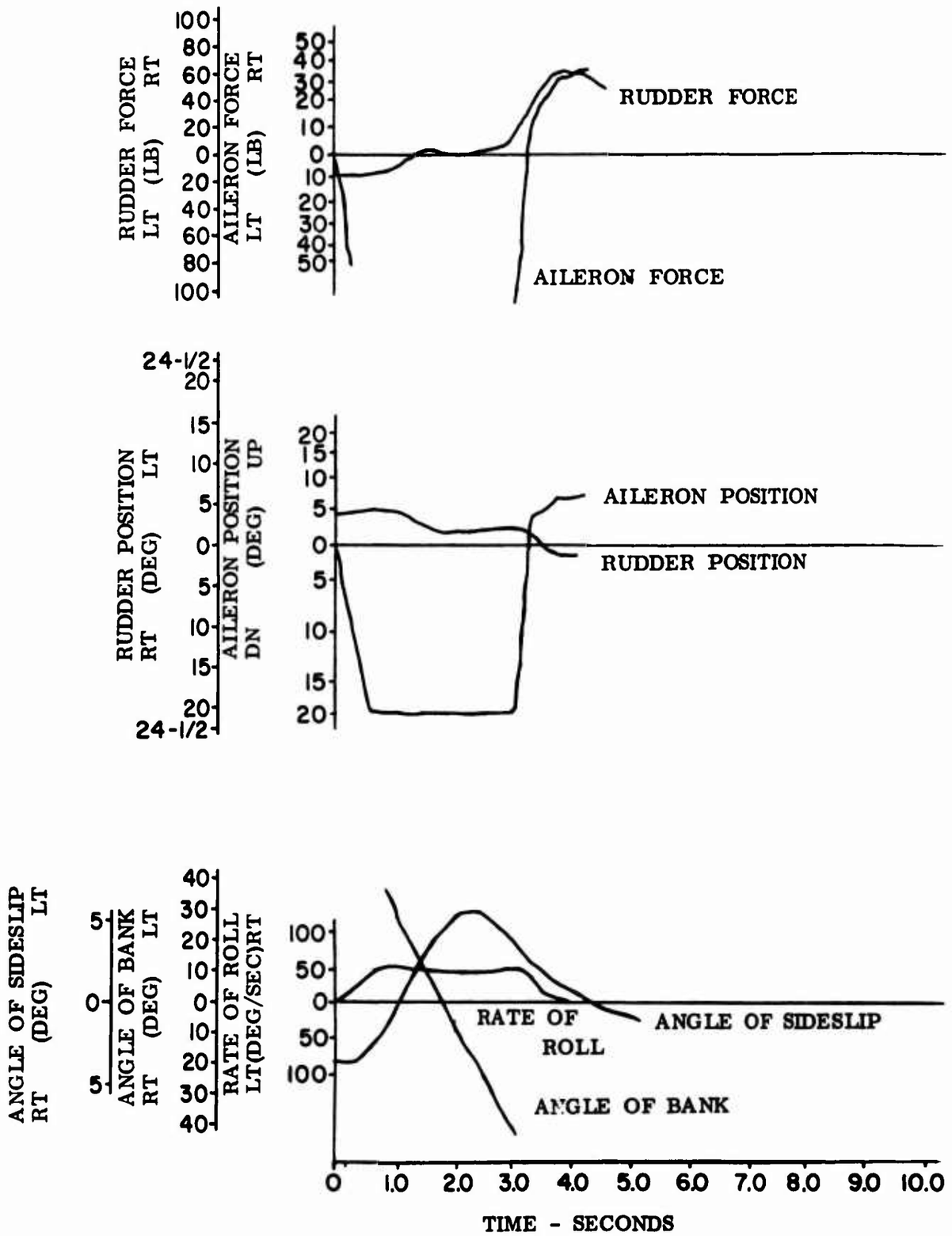


Figure 73. Aileron Rolls - Glide Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT.
169.7	5425	---	8759	30.1	59°F

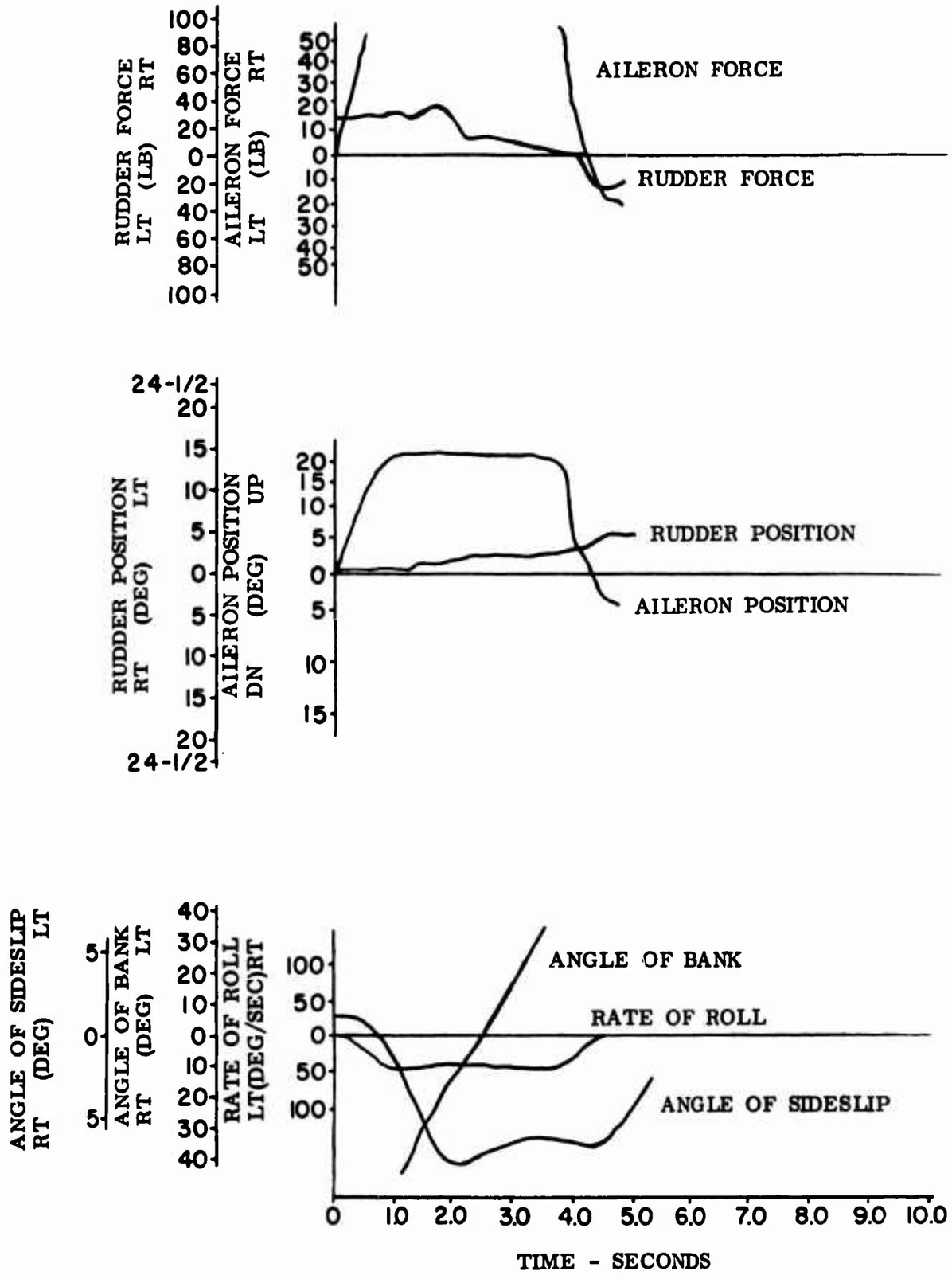


Figure 74. Aileron Rolls - Glide Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT,
132.4	7432	---	8744	30.1	52°F

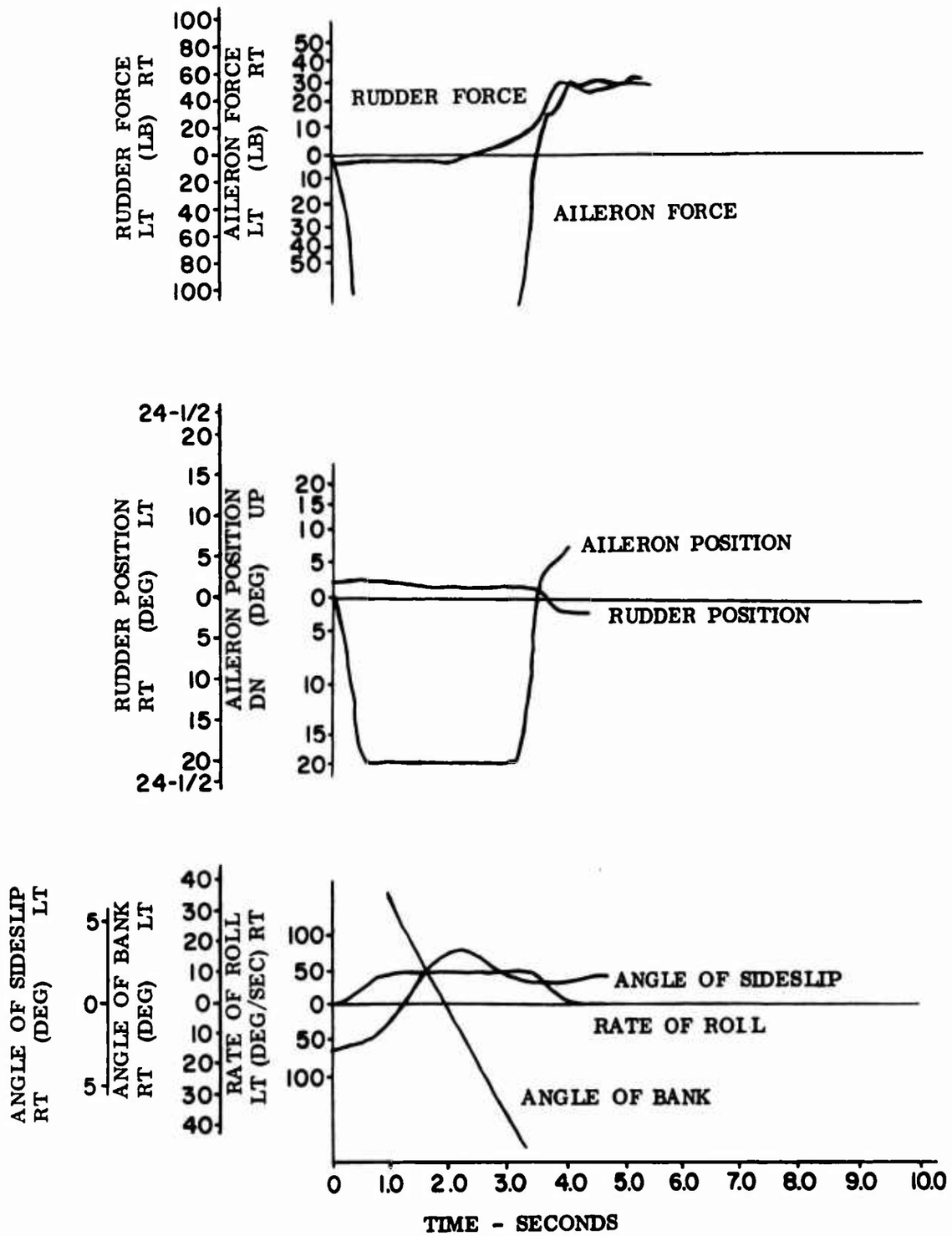


Figure 75. Aileron Rolls - Landing Configuration.

CAS	ALTITUDE	SHP	WEIGHT	CG	OAT.
131.4	5955	---	8748	30.1	51°F

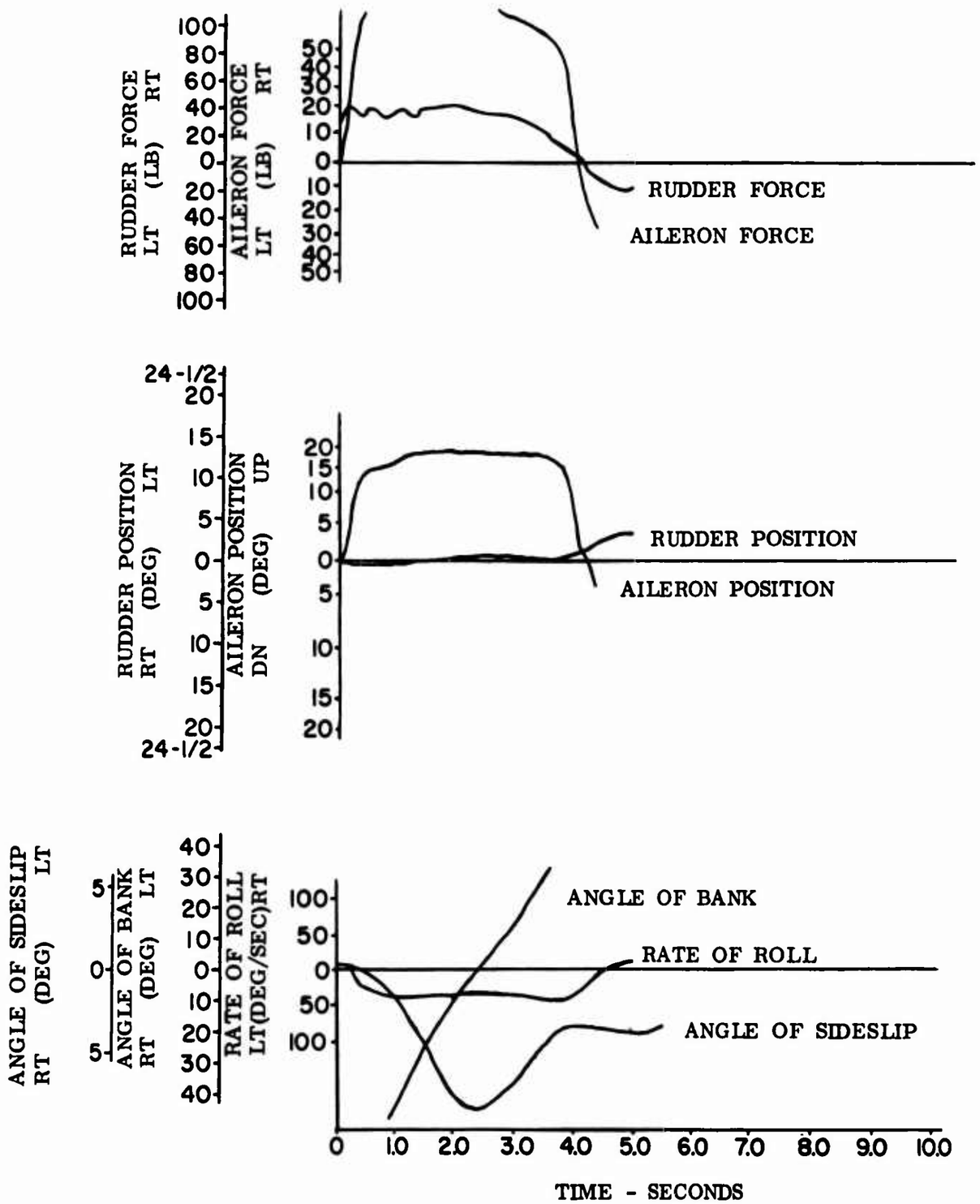


Figure 76. Aileron Rolls - Landing Configuration.

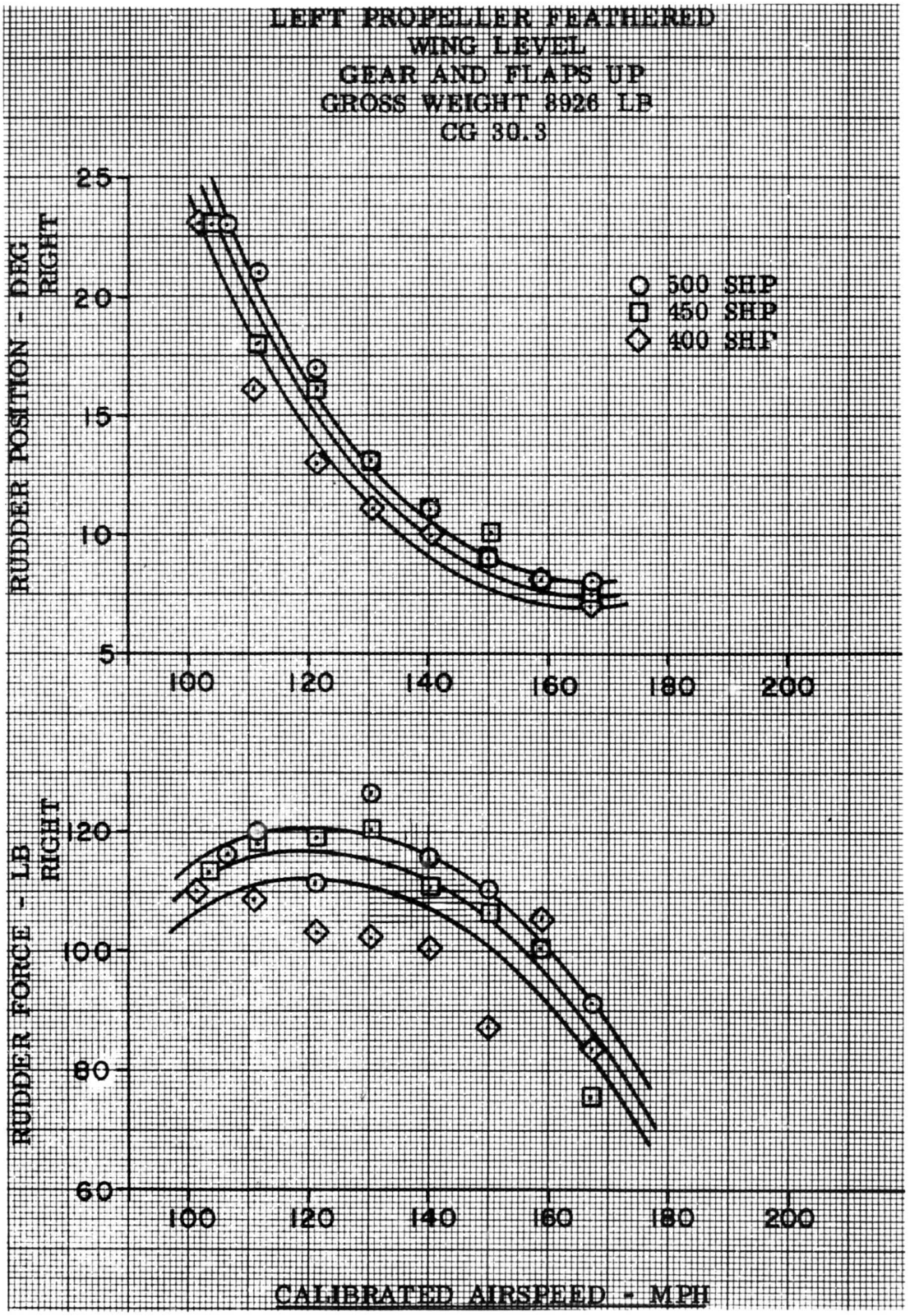


Figure 77. Rudder Power.

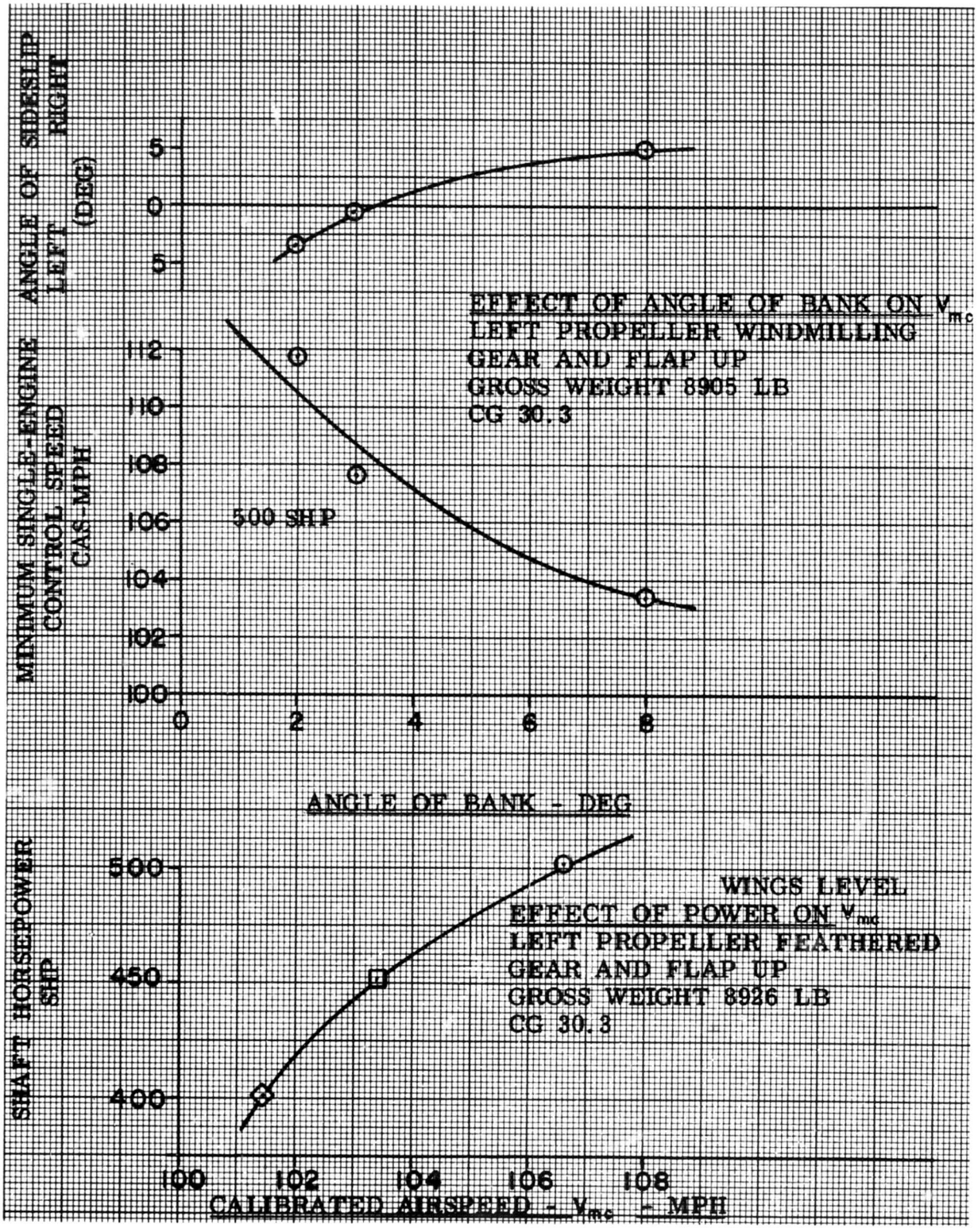


Figure 78. Minimum Control Speed.

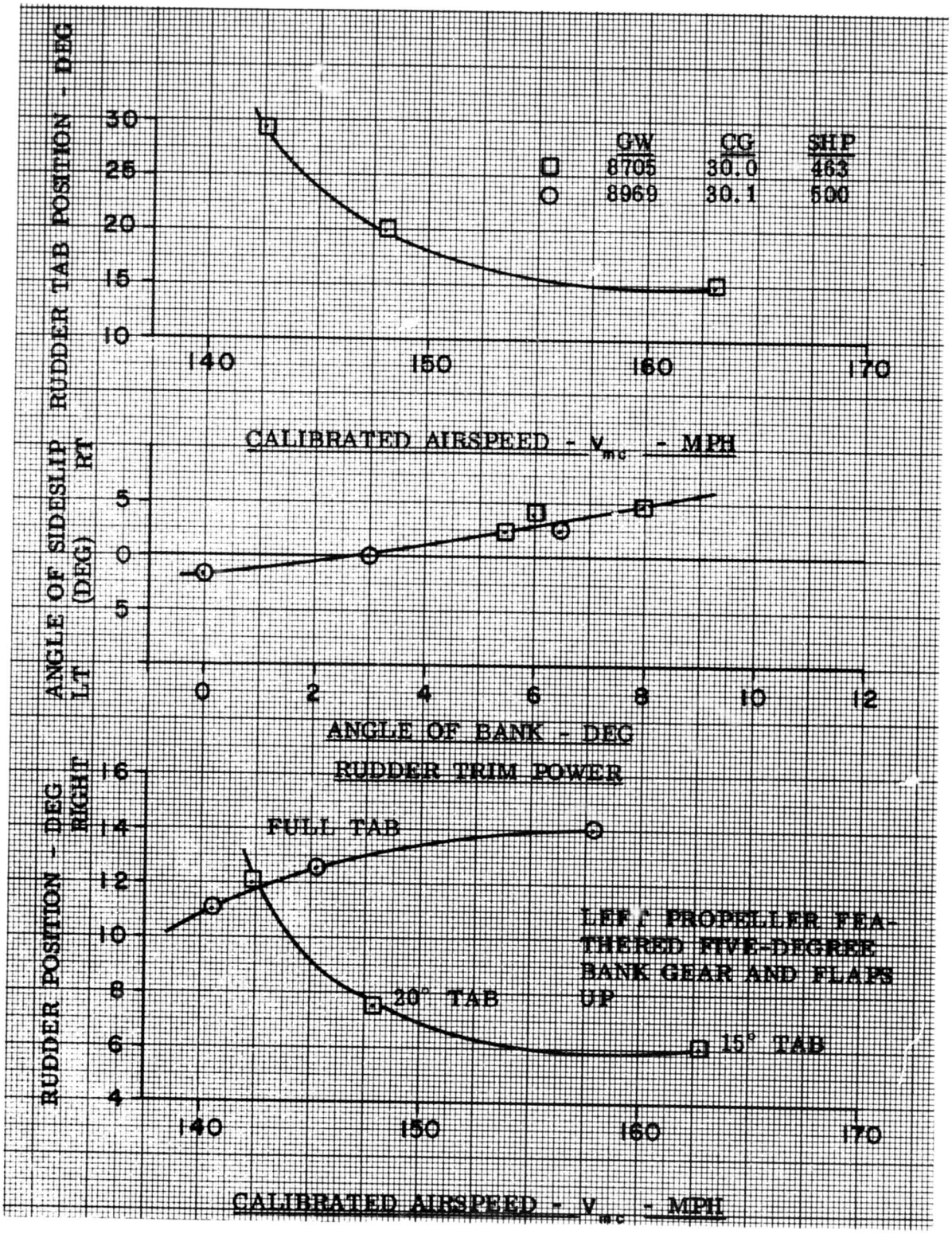


Figure 79. Rudder Trim Position.

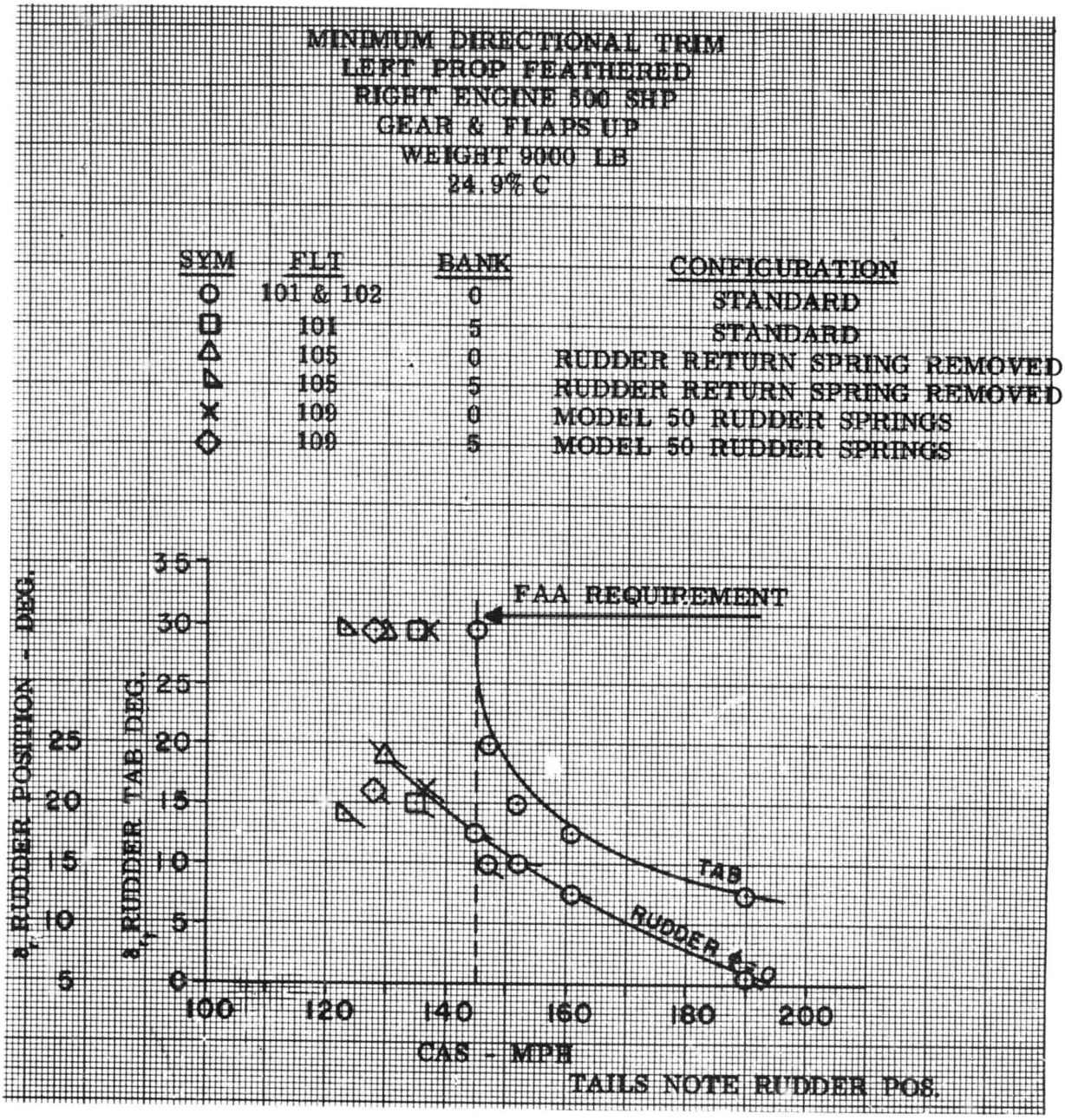


Figure 80. Single-Engine Minimum Rudder Trim.

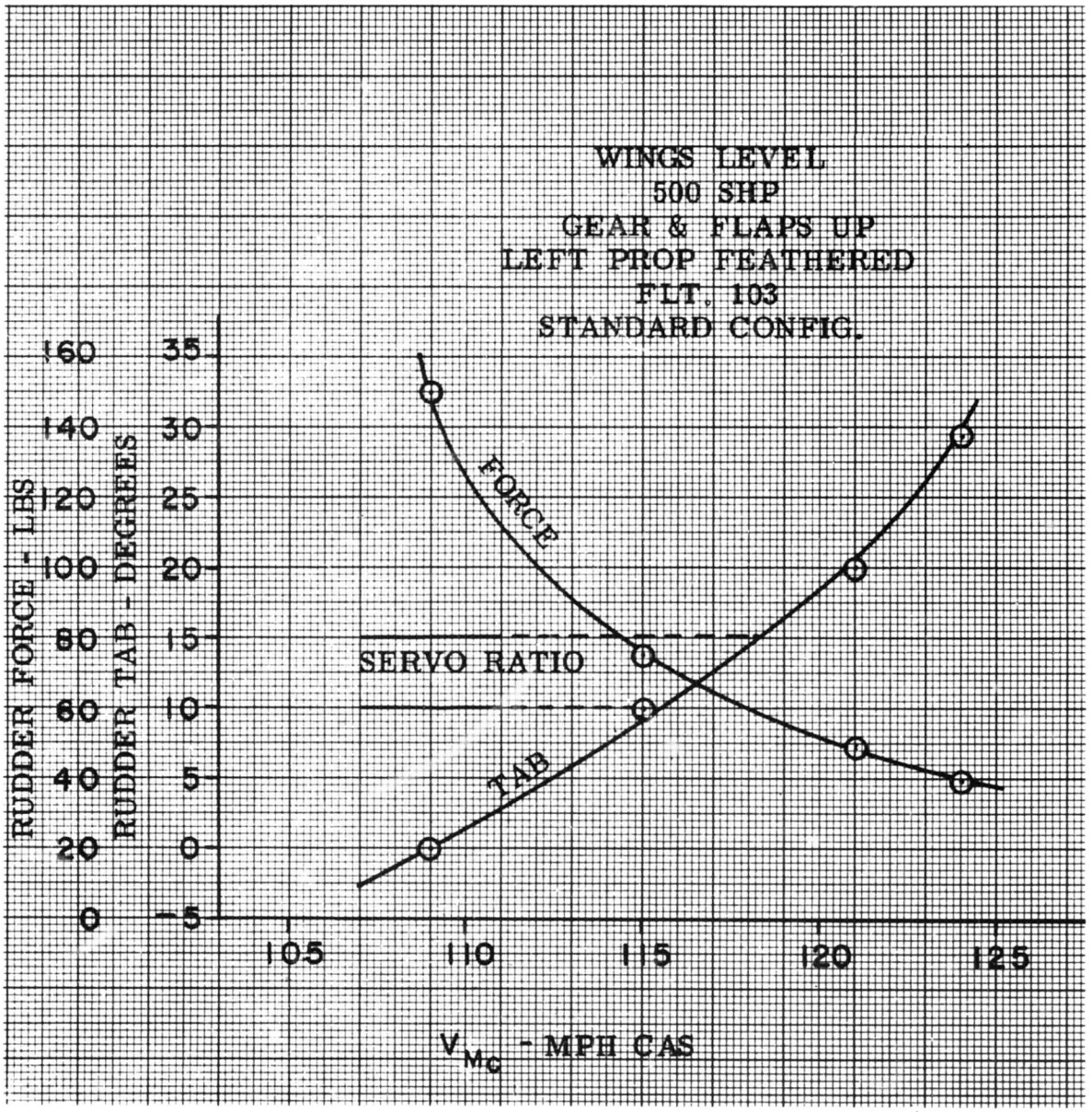


Figure 81. Effect of Rudder Tab on V_{mc}.

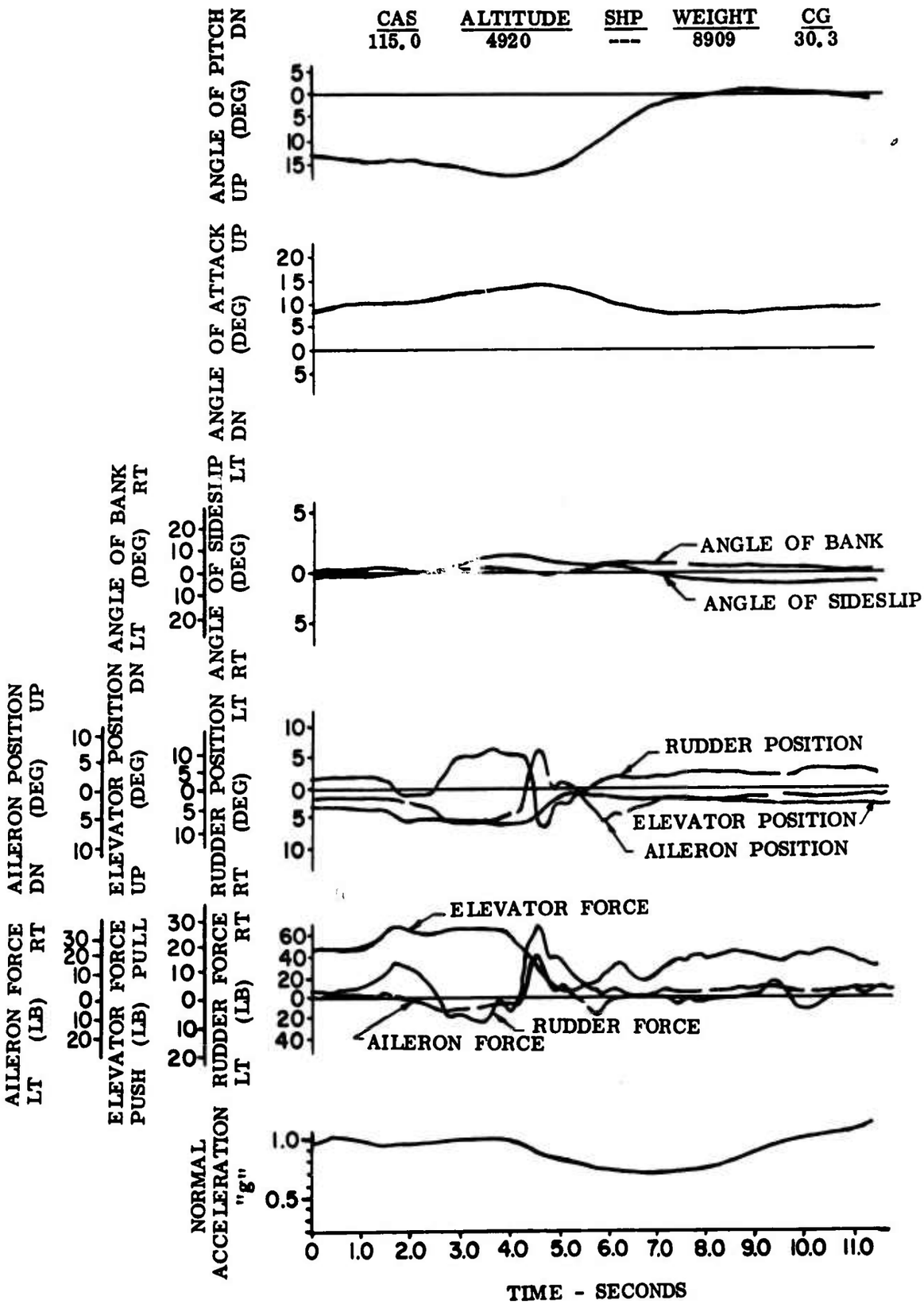


Figure 82. Stall Characteristics - Power-Off Stall - Gear and Flaps Up.

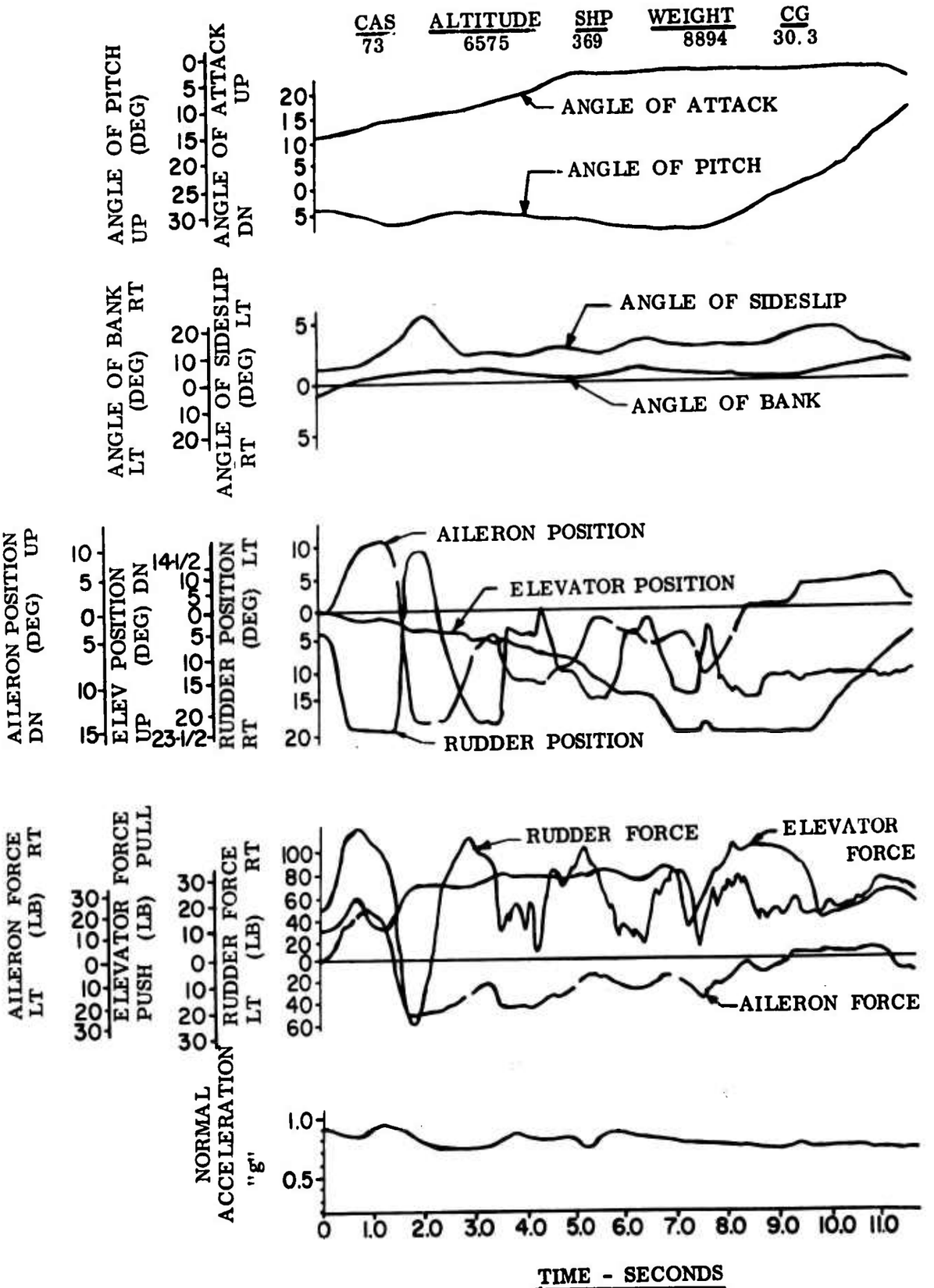


Figure 83. Stall Characteristics - Power-On Stall - Gear and Flaps Up.

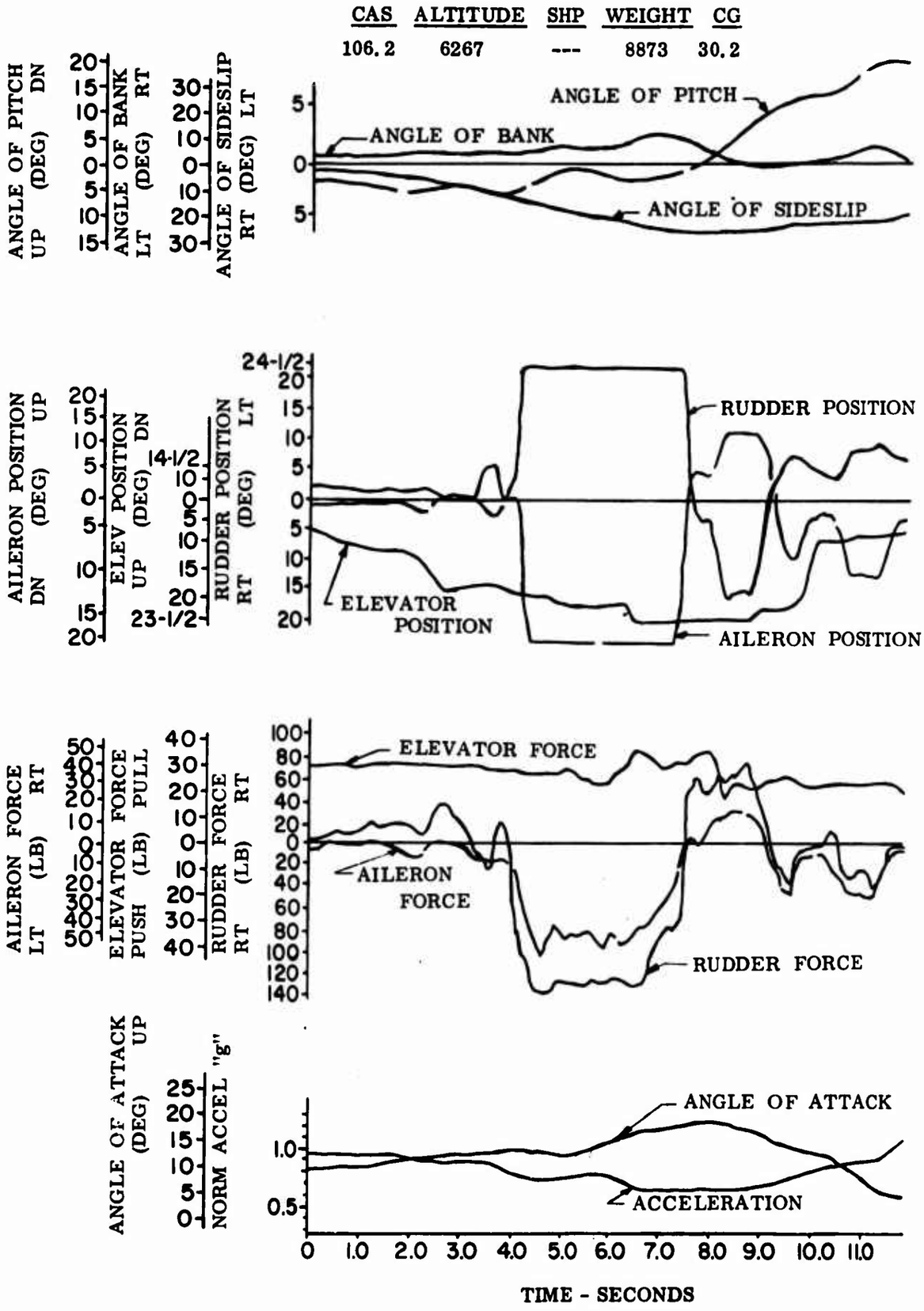


Figure 84. Stall Characteristics - Power-Off Stall - Gear and Flaps Down.

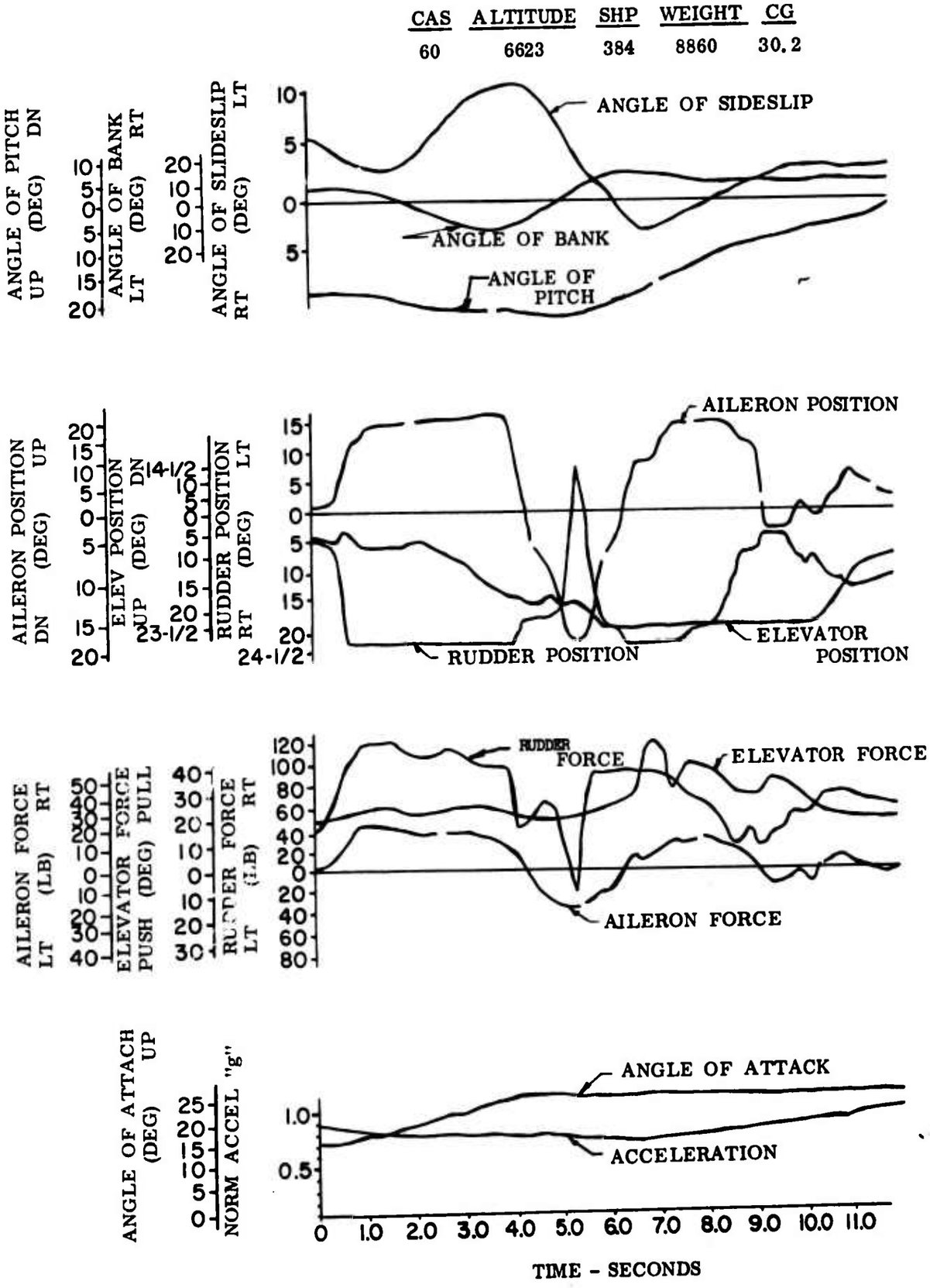


Figure 85. Stall Characteristics - Power-On Stall - Gear and Flaps Down.

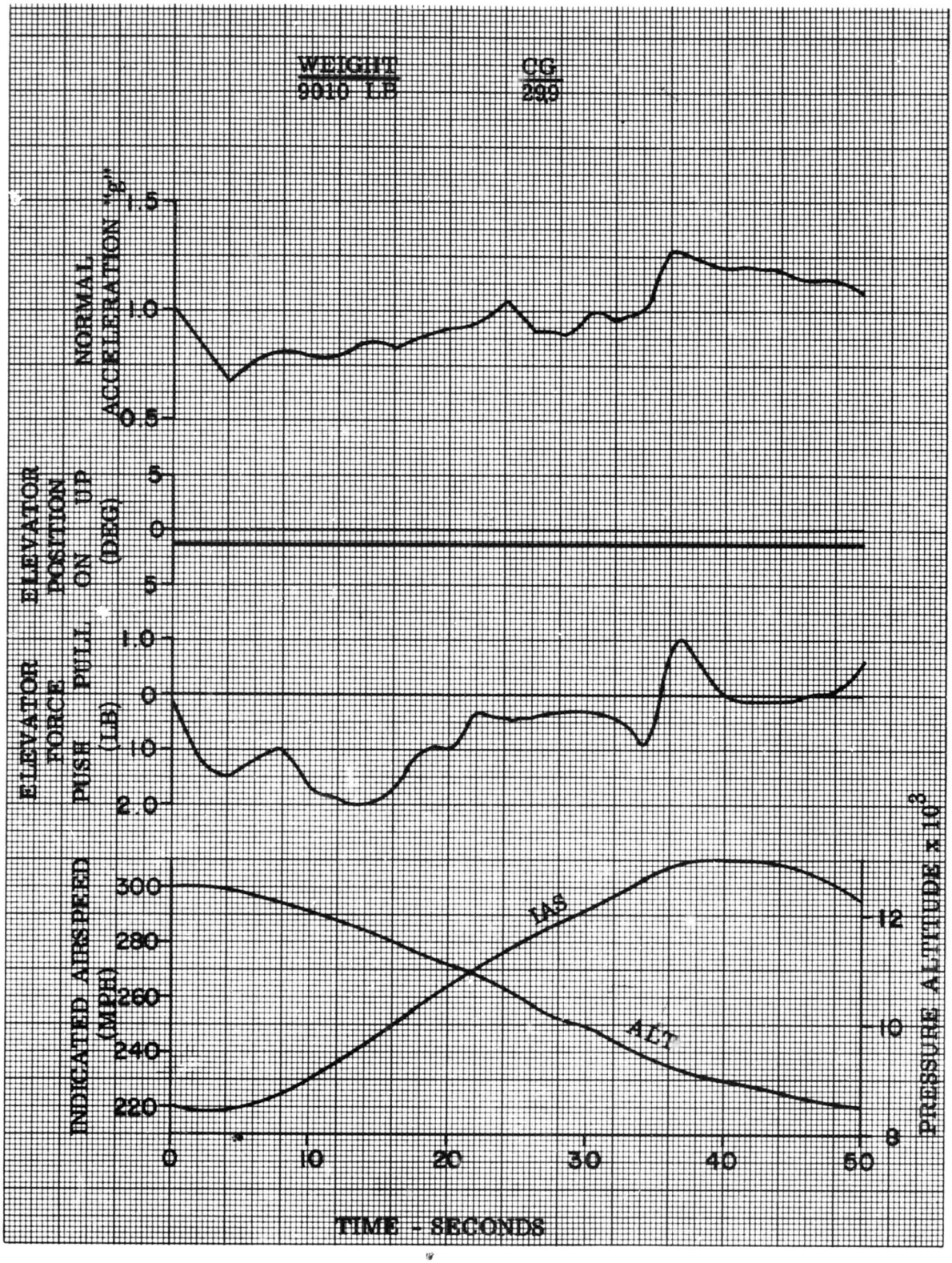


Figure 86. High-Speed Dive.

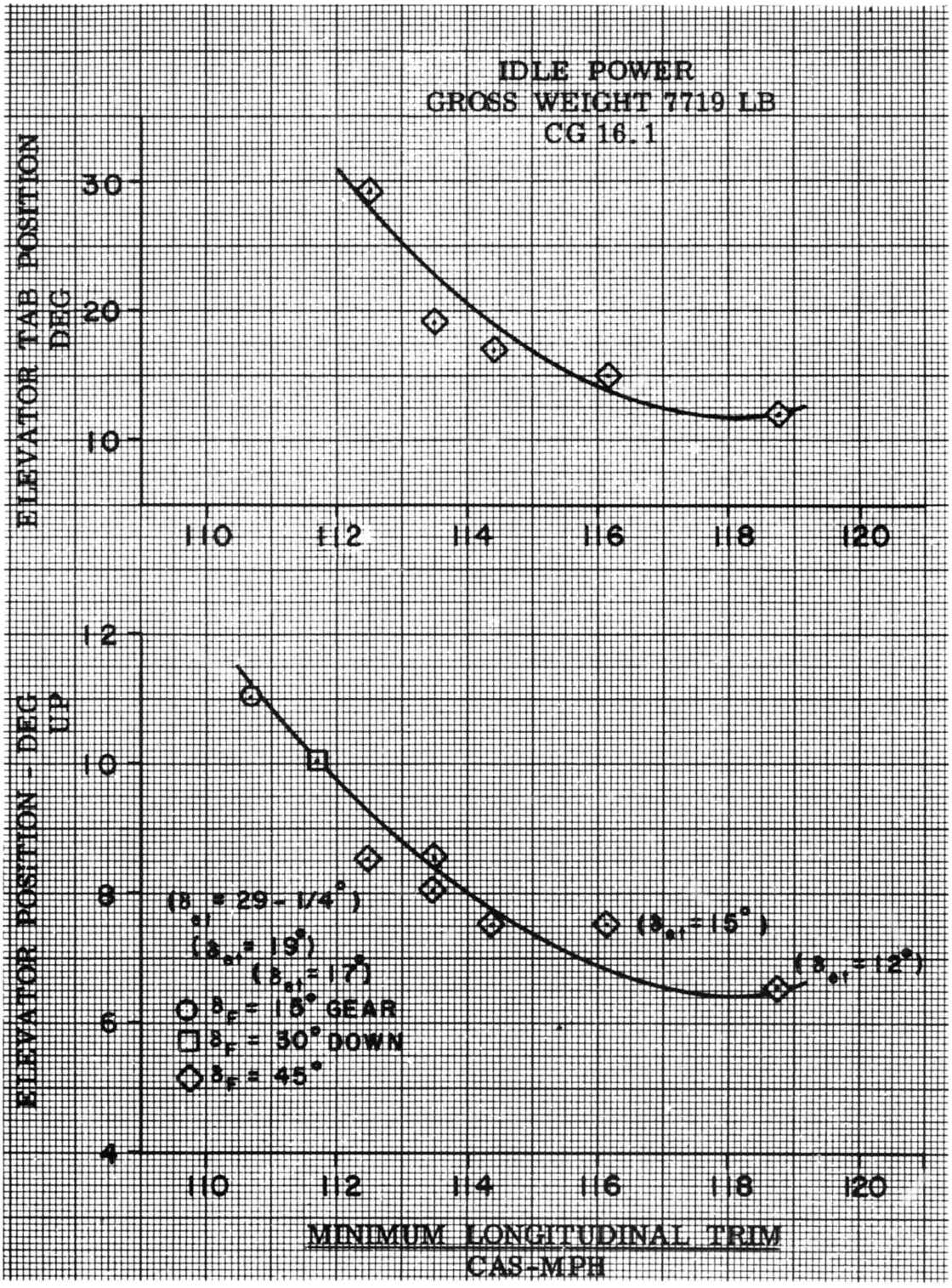


Figure 87. Elevator Tab Effectiveness - Gear and Flaps Down.

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PT6A Turboprop Engine Maintenance Manual - Part Number 3008102, United Aircraft of Canada, Limited, Longueuil, Quebec, Canada, 1 March 1965.

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13. ABSTRACT

The development and flight test program of the NU-8F was conducted by Beech Aircraft Corporation and United Aircraft of Canada, Ltd., for the U. S. Army Aviation Materiel Laboratories under Contract DA 44-177-AMC-27(T). A Beech Model U-8F was modified by the installation of two UACL PT6A-6 turbine engines in place of the reciprocating engines, and by replacing the stabilizer and rudder with a Beech Model A80 stabilizer and rudder. Ground and flight tests were conducted using FAA regulations as the standard of acceptability. The report contains a comparison of performance characteristics of the NU-8F and the U-8F. A discussion of each phase of the performance tests along with tables and graphs of the flight test data is presented in some detail. The NU-8F has more useful load, increased speed, and a more simple fuel system than the U-8F. It is suitable for use as a turbine-powered trainer or as a liaison and transportation aircraft.

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