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USAAVLABS TECHNICAL REPORT 70-37

XV-11A FLIGHT TEST PROGRAM

By

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February 1971

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-266(T)
THE DEPARTMENT OF AEROPHYSICS AND AEROSPACE ENGINEERING
MISSISSIPPI STATE UNIVERSITY
STATE COLLEGE, MISSISSIPPI

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118

Best Available Copy

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) Mississippi State University Department of Aerophysics and Aerospace Engr. State College, Mississippi	2a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED 2b. GROUP

3. REPORT TITLE

XV-11A FLIGHT TEST PROGRAM

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

5. AUTHOR(S) (First name, middle initial, last name)

L.J. Mertaugh, S.C. Roberts, N.S. Kiran

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
February 1971	116	None

8a. CONTRACT OR GRANT NO. DA44-177-AMC-266(T) b. PROJECT NO. Task 1F162203A14203 c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) USAAVLABS Technical Report 70-37 9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report) AASE Report No. 69-7

Approved for public release; distribution unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Eustis Directorate U.S. Army Air Mobility R&D Laboratory Fort Eustis, Virginia

13. ABSTRACT

This report presents the results of a test program that was conducted to evaluate the performance and stability and control characteristics of the XV-11A aircraft. This aircraft is a research vehicle designed to perform basic aerodynamic flight research in the areas of high-lift boundary layer control, propeller thrust augmentation, low drag geometry, and STOL aircraft handling qualities. The aircraft incorporates a number of unique design features including glass fiber reinforced plastic construction; a distributed-suction, high-lift boundary layer control system; a variable-camber wing; and a shrouded propeller. The test data show that the aircraft has sufficient performance and stability and control for conducting low-speed aerodynamic research. Handling qualities research would be limited by the high longitudinal and directional control force gradients. Although low stall speeds are demonstrated, the increment in lift due to the boundary layer control system is less than anticipated. Aircraft performance is somewhat limited by propeller deficiencies due to high blade loading.

DD FORM 1 NOV 65 1473 REPLACES DD FORM 1473, 1 JAN 64, WHICH IS OBSOLETE FOR ARMY USE.

UNCLASSIFIED
Security Classification

UNCLASSIFIED
Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
XV-11A Aircraft Aerodynamic Boundary Layer Control Propeller Thrust Augmentation STOL Variable-Camber Wing						

UNCLASSIFIED
Security Classification

DEPARTMENT OF THE ARMY
U. S. ARMY AIR MOBILITY RESEARCH & DEVELOPMENT LABORATORY
EUSTIS DIRECTORATE
FORT EUSTIS, VIRGINIA 23604

This report has been reviewed by the U. S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound.

The work was performed under Contract DA 44-177-AMC-266(T). In aircraft design, there is always an attempt to optimize certain parameters of the design. If the requirements for short takeoff and landing (STOL) predominate, then almost all other design parameters are compromised.

The XV-11A research aircraft was designed in an effort to obtain STOL performance with a minimum of compromise in the other design parameters. In order to meet these objectives, the aircraft design includes several unique features never before utilized on the same aircraft. The XV-11A employs suction boundary layer control (BLC) and unique variable camber wing for increased lift, a shrouded propeller for increased thrust at low airspeeds, all-fiberglass-reinforced polyester construction for smooth aerodynamic surfaces and reduced overall weight, and large transparent areas for good visibility.

Task 1F162203A14203
Contract DA 44-177-AMC-266(T)
USAAVLABS Technical Report 70-37
February 1971

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ABSTRACT

This report presents the results of a test program that was conducted to evaluate the performance and stability and control characteristics of the XV-11A aircraft. This aircraft is a research vehicle designed to perform basic aerodynamic flight research in the areas of high-lift boundary layer control, propeller thrust augmentation, low drag geometry, and STOL aircraft handling qualities. The aircraft incorporates a number of unique design features including glass fiber reinforced plastic construction; a distributed-suction, high-lift boundary layer control system; a variable-camber wing; and a shrouded propeller. The test data show that the aircraft has sufficient performance and stability and control for conducting low-speed aerodynamic research. Handling qualities research would be limited by the high longitudinal and directional control force gradients. Although low stall speeds are demonstrated, the increment in lift due to the boundary layer control system is less than anticipated. Aircraft performance is somewhat limited by propeller deficiencies due to high blade loading.

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LIST OF SYMBOLS

A_p	propeller disc area	feet squared
a	speed of sound	feet per second
b	wing span	feet
BLC	boundary layer control	———
C_{ℓ_r}	rolling moment coefficient due to yaw rate	———
C_{ℓ_β}	rolling moment coefficient due to sideslip	———
$C_{\ell_{\delta_r}}$	rolling moment coefficient due to rudder deflection	———
CG	center of gravity	———
C_T	theoretical wing tip chord	feet
C_R	theoretical wing root chord	feet
F_s	stick (wheel) force; push positive	pounds
F_{Dir}	rudder force; left rudder positive	pounds
F_{Lat}	aileron (wheel) force; right aileron positive	pounds
g	gravitational acceleration	32.2 feet per second squared
h	pressure altitude	feet
MGC	mean geometric chord	feet
MPH	velocity	miles per hour
mm	millimeters	———
N_2	power turbine speed	percent of design rotational speed

n	acceleration (nondimensionalized by the acceleration of gravity)	————
PIW	"equivalent" power	horsepower
PLF	power for level flight	horsepower
p	pressure	pounds per square foot
p	roll rate; right roll positive	radians per second or degrees per second
r	yaw rate; nose to right positive	radians per second or degrees per second
R/C	rate of climb	feet per minute
RPM	angular velocity	revolutions per minute
SHP	shaft horsepower	horsepower
T	temperature	degrees centigrade
T	thrust	pounds
TE	trailing edge	————
V	velocity	knots or feet per second
ΔV_{Pos}	airspeed position error	knots
VIW	"equivalent" airspeed (referred to a standard weight)	knots
W	aircraft gross weight	pounds
W_f	fuel flow	pounds per hour
α	angle of attack	degrees

β	angle of sideslip; slip to right positive	degrees
δ	pressure ratio ($\delta = p/p_{SSL}$)	————
δ_{Long}	longitudinal control position (measured from face of instrument panel)	inches
δ_{Lat}	lateral control position; clockwise rotation positive	degrees
δ_{Dir}	directional control position; left rudder positive (deflection measured at point where pilot's foot meets pedal with heel on floor)	inches
δ_e	elevator deflection; TE down positive	degrees
δ_a	aileron deflection; right aileron TE up positive	degrees
δ_r	rudder deflection; TE left positive	degrees
ζ_d	dutch roll mode damping ratio	————
η_p	propulsive efficiency	————
θ	angle of pitch; nose up positive	degrees
θ	absolute temperature ratio ($\theta = T/T_{SSL}$)	————
ρ	density	slugs per cubic foot
σ	density ratio ($\sigma = \rho/\rho_{SSL}$)	————
ϕ	angle of roll; roll to right positive	degrees
ω_d	dutch roll mode damped frequency	radians per second

Subscripts

a	ambient conditions
AVAIL	available
e	equivalent
i	indicated reading corrected for instrument error
o	observed instrument reading
REQ	required
S	standard conditions at a specified altitude
SSL	standard sea level conditions
T	test conditions
t	true
TL	tape line
x	longitudinal axis; forward positive
y	lateral axis; right positive
z	directional axis; down positive

Sub-Subscripts

S	standard conditions at a specified altitude
T	test conditions

INTRODUCTION

This report presents the results of a flight test evaluation of the XV-11A research aircraft. This aircraft was developed by Mississippi State University under Contract DA 44-177-AMC-266(T) with the U. S. Army Aviation Materiel Laboratories,* Fort Eustis, Virginia, to perform basic aerodynamic flight research in the areas of high-lift boundary layer control, propeller thrust augmentation, low drag geometry, and STOL aircraft performance and handling qualities. The vehicle incorporates a number of unique features that are a result of experimental and theoretical research conducted by this University over the last decade. These features include glass fiber reinforced plastic construction; a distributed-suction, high-lift boundary layer control system; a variable-camber wing; and a shrouded propeller. The purpose of this test program was to document the flight characteristics of the existing aircraft and to determine areas in which further investigation and improvement are desirable. There is no attempt in this report to evaluate the aircraft characteristics in terms of possible operational utilization, but comments relating to current military specifications are made where applicable.

The test program consisted of 30 flight hours, representing 21 flights over a 5-month period. The test program was completed on 30 April 1969.

The following tests were included in this program:

- Airspeed Calibration
- Stalls
- Speed Power Polars
- Climb Performance
- Longitudinal Static and Dynamic Stability and Control
- Lateral/Directional Static and Dynamic Stability
- Lateral Control
- Transient Trim Conditions

Full instrumentation was used during this program. Standard calibration and test techniques were followed. In an effort to provide test coverage of all the areas of interest within a relatively small test program, the test conditions were for the most part limited to two trim speeds (70 and 120 knots), one pressure altitude (3000 feet), three wing camber positions (0, 15, and 30 degrees), and one propeller speed. The single propeller speed also results in a constant boundary layer control system blower speed, which in turn gives essentially one internal wing pressure.

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*Redesignated Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory.

DESCRIPTION OF AIRCRAFT

The XV-11A is a two-place, high-wing, fixed-gear, pusher-propeller aircraft constructed entirely of glass fiber reinforced polyester plastic materials. The aircraft incorporates a distributed-suction, high-lift boundary layer control system on a wing, which also features a unique wing section camber changing mechanism on its inboard panels. A shrouded, pusher propeller is mounted in the aft fuselage and is powered by a T63-A5A(FE) gas turbine engine. The empennage is an integral part of the propeller shroud structure and utilizes short chord, fixed stabilizer surfaces and conventional rudder and elevator surfaces, forming a cruciform at the rear of the shroud. External views of the XV-11A are presented in Figure 1. A three-view drawing of the XV-11A is found in Figure 2. Pertinent aircraft geometry is given in the Appendix.

The fuselage is a semimonocoque structure constructed entirely of glass fiber reinforced plastic material. The forward portion of the fuselage consists of the crew compartment and is provided with extensive transparent areas, giving practically an unlimited field of view except directly aft. Seating is side by side; however, the right-hand seat was removed during this test program for the installation of test instrumentation. The middle of the fuselage houses the gas turbine engine, the fuel tank, the wing carry-through structure, the landing gear carry-through structure, and the boundary layer control system blower and related ducting. The aft fuselage houses the propeller drive shaft and gearbox and provides attachment for the empennage.

The wing is a high, tapered and unswept design utilizing a single, steel-reinforced spar with a forward "D" section and hinged subspars in the area of the variable-camber section. Except for the spar reinforcing, the wing material is glass reinforced plastic. Each half of the wing consists of a single panel which attaches to the aluminum carry-through structure at the wing fuselage intersection. The portion of each wing panel aft of the spar and inboard of the ailerons is provided with a camber changing mechanism. The camber changing mechanism bends the upper skin of the wing to provide the desired variable camber. The lower wing skins overlap to take up the resulting change in surface length. Figure 3 shows the wing in positions corresponding to 0, 15, and 30 degrees of camber. The wing camber may be changed to provide any deflection between 0 and 30 degrees. The upper surface of the wing is drilled with many small holes to give essentially distributed suction over 81 square feet of wing planform area. The suction holes extend from the wing root to the wing tip fairing and from 7 percent of the wing chord to the trailing edge.

The empennage consists of a propeller shroud to which are attached the directional and longitudinal control surfaces. These control surfaces form a cruciform within the shroud just aft of the propeller. The elevator extends out beyond the shroud and has a small stabilizing surface attached to the outer surface of the shroud. The empennage is constructed of glass fiber reinforced plastic. Four shroud struts are provided between the shroud and the propeller gearbox support structure to stabilize the upper portion of the shroud. The lower portion of the shroud attaches directly to the aft section of the fuselage. Longitudinal trim control is provided by a trim tab located on the inboard end of the elevator surface. The original tab configuration did not allow sufficient trim control for this test program. Two tab extensions were used during this program; Figure 4 shows these modifications. One modification is a straight 2-inch extension attached to the existing tab surface. The second modification also attached to the tab but provided a 3-inch extension which is bent 25 degrees trailing edge down with respect to the tab chord line. The 3-inch extension was used with the forward center-of-gravity position; the 2-inch extension was used on all other flights.

The landing gear consists of four wheels attached to the fuselage structure by fixed cantilever legs. Secondary skins enclose these legs to form a single strut on each side of the aircraft. The forward wheels are steerable, and brakes are provided on all four wheels. The gear structure is a combination of wood and glass fiber reinforced plastic. A tail skid is provided under the aft fuselage.

The propulsion system consists of a T63 gas turbine driving the aft-mounted propeller through a 7-foot drive shaft and a propeller gearbox mounted just forward of the propeller. The propeller blade angle is controlled by electric motors mounted on the propeller hub. Electric motor failure was experienced early in the test program. Improvements in the armature windings corrected this problem. Although an automatic β control system is provided in the XV-11A, the system did not operate satisfactorily, and the manual "beeper" system was used for this test program. The T63 engine also provides the drive power for the boundary layer control system blower. Inlet air for the T63 engine is supplied by a flush inlet located in the upper surface of the fuselage just aft of the crew station.

The boundary layer control system consists of a centrifugal blower mounted forward of and powered by the T63 engine.

Ducts connect the blower to the wing roots and to the exhaust outlets located in the side of the fuselage below the T63 engine exhaust outlets. The forward part of the blower ducting is shown in the upper picture in Figure 5. The interior of the wing acts as a plenum for the suction system. The present configuration supplies an internal wing pressure of 15 inches of water below ambient pressure for 0-degree wing camber and 95.5-percent N_2. This value is reduced by about 1 inch of water with 30 degrees of wing camber. Although suction flow rate has not been measured, blower characteristics indicate a flow rate in excess of 6000 cubic feet of air per minute.

The aircraft center of gravity was controlled by 90 pounds of lead ballast, which was located within the aircraft to give a constant average gross weight of 2630 pounds and a center-of-gravity travel from 21.5-percent MGC to 34.6-percent MGC at the assumed average mid-mission fuel load.

TEST EQUIPMENT

The test instrumentation used in the flight test program was supplied by the U. S. Army Aviation Materiel Laboratories and Mississippi State University. All equipment installation and calibration were performed by Mississippi State University. Calibration of the instrumentation calibration equipment used is traceable to the National Bureau of Standards.

The recorded test data were obtained from pilot notes, a photo-panel mounted aft of the pilot and incorporating a 16mm camera set to take 1 frame per second throughout the test flight, and a 50-channel recording oscillograph mounted in the area normally occupied by the observer's seat. The functions that were recorded by the photo-panel and the oscillograph are as follows:

Photo-Panel
- Event Light
- Counter
- Time
- Airspeed
- Altitude
- Power Turbine Speed (T63)
- Gas Producer Speed (T63)
- Engine Torque
- Instrumentation Excitation Voltage

Recording Oscillograph
- Event Marker
- Counter
- Airspeed
- Altitude
- Stick Force
- Right Rudder Force
- Left Rudder Force
- Normal Acceleration
- Lateral Acceleration
- Longitudinal Acceleration
- Pitch Rate
- Roll Rate
- Yaw Rate
- Pitch Angle
- Roll Angle
- Elevator Deflection Angle
- Right Aileron Deflection Angle
- Rudder Deflection Angle
- Wheel (Longitudinal) Position
- Wheel (Lateral) Angle
- Rudder Pedal Position

Angle of Attack
Angle of Sideslip

Two flight test booms were mounted on the wing tips of the test aircraft. The design of the boom pivoting head is such that the position error is minimized. In addition to the total and static pressure sources located on the boom head angle of attack and sideslip are also provided as data output. The left wing tip boom was used as the airspeed and altitude source for the pilot's system and the photo-panel. The angle-of-attack and sideslip output from this boom were recorded by the oscillograph. The right wing tip boom pressure sources were fed to pressure transducers mounted in the wing tip. These data were recorded by the oscillograph. The flight test boom is shown in Figure 6.

A trailing static bomb was used for the calibration of the static pressure source (left flight test boom). The static bomb was trailed on a 45-foot line. The bomb is a fin-stabilized, cylindrical body with a hemispherical nose. The static source is an annular slot located three body diameters aft of the leading edge of the bomb and seven tube diameters forward of the supporting tube. This static source location corresponds to the classic "Prandtl" pitot probe. The bomb in its "retracted" position is shown in Figure 7. An acceptable calibration of this bomb is not available, but a preliminary comparison with a trailing cone system has shown good agreement. A brief test in the Mississippi State University low-speed wind tunnel has indicated a constant plus 1-knot error over most of the tested speed range. However, the magnitude of tunnel turbulence makes this result subject to question. In light of these uncertainties, no correction was applied to the bomb data.

The longitudinal and lateral control forces were obtained with a modified stick force transducer. The unit is shown mounted on the control wheel in the bottom picture of Figure 5. This strain-gage, beam transducer senses forces only in one plane and must be rotated 90 degrees to provide data on either longitudinal or lateral control forces. The effective moment arm about the lateral control axis is 3.7 inches.

TEST PROGRAM

The purpose of this program was to document the existing flight characteristics of the XV-11A aircraft. To allow evaluation of all significant flight characteristics, it was necessary to restrict the number of flight conditions to two trim speeds, one altitude, three wing camber deflections and one propeller speed. The two trim speeds (70 and 120 knots) represent an approach condition and a cruise condition respectively. Both speeds allow a sufficient range of speeds about the trim speed to define speed stability characteristics and a range of load factors to define the maneuvering characteristics. Seventy knots is also a speed that can be used with all wing camber deflections. Although 120 knots does not represent a maximum cruise speed, it does demonstrate the cruise speed characteristics. In a couple of configurations the 70-knot trim speed was not obtained because of limitations of the longitudinal trim system. This restriction was corrected on later flights, but some tests were not rerun since the speed differences were not considered to be important for that phase of the program. A pressure altitude of 3000 feet was used for most of the test flights. The stall tests were conducted at 5000 feet to ensure sufficient recovery altitude. The three wing camber deflections used were 0, 15, and 30 degrees. Zero and 30 degrees represent the minimum and maximum camber deflections available. The single propeller speed (95.5 percent) represents the maximum engine governor setting that will prevent engine overspeed under all transient conditions. The single propeller speed also ensures a constant boundary layer control system blower speed which in turn gives essentially a constant wing suction pressure.

Two center-of-gravity positions were investigated. The forward position (21.5-percent MGC) was given only token evaluation. The aft position (34.6-percent MGC) represents a limiting position for the gross weight tested because of landing gear loads. It should be noted that this aft position still results in a significant level of longitudinal static stability; for this reason, it was not possible to provide a sufficient range of positions approaching the neutral point to allow identification of the neutral point, maneuver point, etc.

Tests were conducted with the boundary layer control system operating, the blower disconnected with the wing surface suction holes sealed, and the blower disconnected with the wing surface suction holes open. The suction holes were sealed by the use of a thin plastic tape applied to the upper surface of the wing. The tape was applied in a chordwise direction to minimize airflow disturbances. With the holes open and the blower disconnected, the flow of air through the wing surface is dependent upon the relative pressure across the skin. The inner wing is essentially vented to cabin ambient pressure in this configuration. This configuration was tested only with 0 degrees of wing camber as it represents an emergency situation with limited application.

A summary of the flight test program is given in Table I.

TABLE I. TEST FLIGHT PROGRAM

CG	Camber (deg)	BLC	Test*
AFT	0	ON	1,2,2a,3,4,5,6,7,8,8a 9,10,11
AFT	15	ON	1,2,3,4,5,7,8a 1,2,2a,3,4,5,6,7,8,8a 9,10
AFT	30	ON	1,2,2a,3,4,5,6,7,8,8a 9,10
AFT	0	OFF/SEALED	2,3,4,5,7
AFT	15	OFF/SEALED	2
AFT	30	OFF/SEALED	2,5,7,10
AFT	0	OFF/OPEN	2
AFT	15	OFF/OPEN	2
AFT	30	OFF/OPEN	2
FWD	0	ON	2,5,6
FWD	15	ON	2,5
FWD	30	ON	2,5

*1. Airspeed Calibration
2. Stalls
2a. Accelerated Stalls
3. Speed Power Polars
4. Sawtooth Climbs
5. Longitudinal Static Stability
6. Longitudinal Maneuvering Stability
7. Lateral/Directional Static Stability
8. Lateral/Directional Dynamic Stability (Short Period)
8a. Lateral/Directional Dynamic Stability (Long Period)
9. Longitudinal Dynamic Stability, Dihedral Effect, Spiral Stability, Adverse Yaw
10. Lateral Control
11. Out-of-Trim Conditions

In general the test techniques used in the program were consistent with those discussed in the various service flight test manuals.[1,2,3] Performance and longitudinal static stability data were obtained using the stabilized point method. Longitudinal maneuvering characteristics were found with the steady turn method. Elevator doublets were used in the longitudinal dynamic stability tests. Lateral control was determined from 45-degree banked turns. Rudder doublets and sideslip maneuvers were used for lateral/directional stability. Control friction characteristics were measured by slowly varying the control deflection throughout the range of control travel.

The flight restrictions approved by the Department of the Army for this test program were that the flight velocity not exceed 200 knots true airspeed and that the load factor not exceed the range from 0 to +2.5g. In addition it was arbitrarily decided that flight speeds greater than 100 knots would not be flown with wing camber deflections other than 0 degrees.

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[1] PERFORMANCE TESTING MANUAL, U. S. Naval Test Pilot School, Naval Air Test Center, Patuxent River, Maryland, August 1966.

[2] Herrington, Russel M., Shoemacher, Paul E., Bartlett, Eugene R., and Dunlap, Everett W., FLIGHT TEST ENGINEERING HANDBOOK, Air Force Technical Report No. 6273, U. S. Air Force, Air Force Systems Command, Air Force Flight Test Center, Edwards Air Force Base, California, Revised January 1966.

[3] STABILITY AND CONTROL HANDBOOK, FTC-TIH-64-2004, U. S. Air Force, Air Force Systems Command, Air Force Flight Test Center, Edwards Air Force Base, California, Date Unknown.

DATA REDUCTION

The methods used in correcting the flight test data to standard conditions are given in the following paragraphs. The "United States Standard Atmosphere" was used. This is a geopotential atmosphere that agrees with the ICAO and NASA atmosphere in the range of altitudes used in this program. The standard sea level conditions used in this report are given below.

$T_O = 518.69\ ^{o}R$

$p_O = 2116.2$ psf or 29.92 in. H_g

$\rho_O = 0.0023769$ lb sec^2/ft^4

$g_O = 32.174$ ft/sec^2

$a_O = 661.48$ kn

The standard aircraft weight used with these test data is 2630 pounds. This weight represents the known takeoff weight minus an allowance for an average, midmission fuel consumption. The takeoff gross weight was 2690 pounds. This definition is necessary because of the position error in the fuel quantity measuring system. For this reason no correction for aircraft gross weight is applied to the test data. The possible error in gross weight is in general less than 2 percent. The center-of-gravity positions used in this report are based upon measured weight and balance data for each test configuration with a computed allowance for the average, midmission fuel weight.

Essentially all of the test instruments, including the pilot's flight instruments, were calibrated for instrument error and these corrections were applied to the test data. The only exceptions to this statement are the engine tachometers and the engine torque transducer built into the T63 engine. Equipment was not available to calibrate these items, and no corrections were applied. The T63 engine is not a calibrated engine.

No position error was necessary for the airspeed or altitude systems. The static pressure source position error was negligible (see Airspeed Calibration under Discussion and Results), and the self-aligning design of the flight test boom pitot head makes the assumption of no total pressure error reasonable. No compressibility corrections were used because of the relatively low flight speeds.

Instrument corrections were applied to the ambient temperature measurements, but no recovery factors were applied. The transducers were aligned to the fuselage axis system, and no alignment corrections were used. The rudder pedal force transducers were calibrated in terms of a force applied at the lower edge of the rudder/brake pedal where the ball of the pilot's foot would normally be with the heel resting on the floor.

The rate-of-climb test data were reduced to standard atmospheric conditions based on the maximum available military power at 95.5-percent power turbine speed and 3000 feet pressure altitude. This turbine speed represents the maximum overspeed governor setting found to be usable on this aircraft. The rate-of-climb correction was developed from the expression

$$R/C = \eta_P (SHP_{AVAIL} - SHP_{REQ}) \frac{33000}{W}.$$

The actual correction as used in this report is given by

$$R/C|_{TL_S} = \frac{R/C|_T \sqrt{\frac{T_T}{T_S}} \left(SHP_{AVAIL_S} - \frac{PIW}{\sqrt{\sigma_S}}\right)}{\left(SHP_{AVAIL_T} \sqrt{\frac{T_S}{T_T}} - \frac{PIW}{\sqrt{\sigma_S}}\right)}$$

and results in the tape-line rate of climb for standard conditions at 3000 feet pressure altitude.

The level flight data were reduced in terms of "equivalent" power and speed.

$$PIW = SHP_T \sqrt{\sigma_T} \left(\frac{W_S}{W_T}\right)^{\frac{3}{2}} = SHP_T \sqrt{\sigma_T}$$

$$VIW = \sqrt{\frac{V_e^2 \, W_S}{W_T}} = V_e$$

Fuel flow data were reduced to standard conditions by plotting $\frac{W_f}{\delta_a \sqrt{\theta_a}}$ versus PIW. Since all of the fuel flow data were obtained at a pressure altitude of 3000 feet, it is not known if a single curve of the fuel parameter versus PIW would be obtained for other test altitudes.

DISCUSSION AND RESULTS

The test results obtained during this test program and a discussion of these results are presented on the following pages. Comments regarding the suitability of the XV-11A for its intended use as a research vehicle are provided where pertinent. Due to the nonoperational role of this aircraft and the changing status of MIL-F-8785A, comments regarding the ability of this aircraft to satisfy the requirements of the flying qualities specification are not provided. However, the analysis of the test data does consider the various parameters used in both the earlier and current revisions of MIL-F-8785A, and where possible, values are given which will allow direct comparison with the requirements of this specification.

Some disagreement will be found between the trim airspeeds given in figure titles and those shown on the various airspeed time history curves of this report. The trim airspeeds given in the titles represent the pilot's reading corrected for instrument error and are considered to be the most accurate measure of the aircraft airspeed. The airspeed time histories come from the oscillograph records and are subject to zero shift and low sensitivity. The low sensitivity is a result of the limited trace travel for each channel and the type of pressure transducer that was used. Because of these problems, the airspeed time histories are considered to be reasonable records of the changes in airspeed but a poor indication of the absolute airspeed at any instant in time.

AIRSPEED CALIBRATION

The calibration of the static pressure source used for the pilot's airspeed and altitude instruments, as well as the corresponding photo-panel instruments, was accomplished with a trailing static bomb. A sensitive differential pressure gage was used to measure the pressure difference between the two static sources. The total pressure error is assumed to be negligible because of the self-aligning feature of the flight test boom head. The calibration data converted to airspeed error are shown in Figure 8. Since the error is essentially within the reading error of the instruments, the position error is assumed to be zero throughout this report.

CONTROL FRICTION CHARACTERISTICS

The longitudinal, lateral, and directional control force characteristics obtained on the ground in the absence of any aerodynamic loads are given in Figures 9, 10, and 11. The arrows shown along the curves represent the direction of surface travel during the test. The partial mass balance of the elevator produces the offset of the wheel forces in the "pull" direction seen in Figure 9. The large rudder pedal forces shown in Figure 11 result for the most part from the direct connection of the front wheel steering to the rudder cables. The influence of the steering system was minimized by lifting the forward wheels off of the ground. Except for the rudder forces, the high breakout forces are not reflected in the various force measurements made in flight and shown in the following sections of this report. The significant vibration level produced in flight by the propulsion system appears to reduce the longitudinal and lateral control friction to an acceptable value.

PERFORMANCE

The evaluation of the performance characteristics of the XV-11A aircraft, based on the test data obtained during this test program, consists of the presentation of the following performance parameters: power required for level flight, fuel flow, rate of climb, and aircraft stall characteristics. Investigations into the operating efficiency of the various, unique XV-11A aircraft components were not conducted. These features, such as low drag geometry, the shrouded propeller, the variable camber wing and the high-lift boundary layer control system can, therefore, only be evaluated collectively in terms of the above mentioned performance parameters. The few comments and conclusions pertaining to these components that may be derived from the results of this program are presented in the appropriate, following paragraphs. It should be noted that neither wind tunnel tests of this configuration nor isolated tests of the shrouded propeller have been conducted. Limited tests were performed on the boundary layer control system blower impeller, but facilities were not available to test the complete blower configuration outside of the aircraft. Static in-place testing was restricted by engine limitations.

Although static thrust measurements were not made as a part of this test program, static thrust data were obtained in an earlier series of tests. Only the maximum value of static thrust is discussed here to provide a measure of the low speed thrust capabilities of this aircraft.[4] These measure-

[4] Roberts, S., Stewart, D., Boaz, V., Bryant, G., Mertaugh, L., Wells, G., Gaddis, M., XV-11A DESCRIPTION AND PRELIMINARY FLIGHT TEST, Mississippi State University; USAAVLABS Technical Report 67-21, U. S. Army Aviation Materiel Laboratories, Fort Eustis, Virginia, April 1967, AD 654,469.

ments were obtained with a shrouded propeller configuration which was essentially the same as used during the present test program. The only difference between the configuration used in the earlier test and the present configuration was two additional shroud-to-fuselage struts now being used on the XV-11A. These struts are constructed of 1.375 inches by 0.625-inch streamlined tubing nominally located 6.5 inches forward of the plane of the propeller. These struts should not have any significant effect on the validity of these test data. The propeller used in both tests is one originally provided with the YT-63 turboprop engine and modified by removing eleven inches from the tip of each blade. All of the measured power was available to the propeller drive system since the boundary layer control system blower was not installed during the earlier tests. Static thrust was measured by means of a 2500-pound-capacity dynamometer attached to the tail skid of the XV-11A. A maximum thrust of 1200 pounds was obtained with 248 shaft horsepower. These values give a figure of merit of 0.907 where

$$FM = \frac{0.707\ T^{\frac{3}{2}}}{550\ SHP\sqrt{\rho A_p}}$$

Test stand data obtained by the manufacturer on the original, 88-inch-diameter propeller gave 1100 pounds of static thrust with 250 shaft horsepower. This gives a figure of merit of 0.589. Based on these results, the use of the shroud provides a nine-percent increase in static thrust with a propeller having 44-percent less disc area and gives a 54-percent increase in the figure of merit.

CLIMB PERFORMANCE

The maximum rate of climb for a standard day at a pressure altitude of 3000 feet and the standard gross weight of 2630 pounds with military power (95.5-percent N_2) is given as a function of equivalent airspeed in Figure 12. Wing camber positions of 0, 15, and 30 degrees with boundary layer control on as well as 0-degree camber with boundary layer control off and the wing sealed are presented. For this altitude, power turbine speed, and range of flight velocities, the military power available is 262.7 shaft horsepower.

The higher rates of climb shown with the boundary layer control system off and the suction holes sealed result from the larger value of power available to the propeller when the boundary layer control blower is not being used. The small increase in speed for maximum rate of climb with boundary layer control off again is a result of the larger power available, which tends to rotate the climb curve as well as increase the ordinate values. The fuel flow associated with military power and 95.5-percent N_2 is 179 pounds per hour.

This value corresponds to $W_f/\sigma_a\sqrt{\theta_a}$ = 202 pounds per hour

and was obtained from Figure 14 using 262.7 shaft horsepower or an equivalent power (PIW) of 251.3 shaft horsepower.

LEVEL FLIGHT PERFORMANCE

The variation of equivalent power required for straight and level flight at the standard gross weight of 2630 pounds as a function of equivalent airspeed is shown in Figure 13. These speed-power polars are given for 0, 15, and 30 degrees of camber with boundary layer control on and zero camber with boundary layer control off and the wing suction holes sealed.

A comparison of the power required for the 0-degree camber configuration with and without the boundary layer control system operating shows a difference of about 35 equivalent shaft horsepower over a large part of the speed range. There is some reduction in this difference at the higher speeds, but this may be more a matter of the curve fairing used. A preliminary measurement of the boundary layer control system exhaust velocity has shown a nominal value of about 90 knots. This value would indicate that at a flight velocity of 90 knots, the power difference is essentially that required to drive the blower.

The maximum speed reached in straight and level flight was 159 knots. The pilot noted that at this flight condition there was a significant vibration present. Previous experience indicates that this is due to propeller stall and a result of the high blade loading existing with the present propeller on the XV-11A. With a propeller that is better matched to the available power of the T63 engine, maximum speeds of at least 175 knots should be realized.

The fuel flow characteristics determined during this test program as a function of the equivalent shaft horsepower are given in Figure 14. All of these data points were obtained at an average pressure altitude of 3000 feet.

STALL CHARACTERISTICS

In general, the stall characteristics of the XV-11A aircraft, when a well-defined wing stall is achievable, result in a sharp break with little or no warning when entered at idle power. Slight buffet occurs just prior to stall when power for level flight at 70 knots is carried into the stall. Some wing drop is experienced, but there is no tendency for a spin to develop, and aileron control is available through wing stall. Stall recovery is normal; however, the lack of a significant stall warning makes the development of a secondary stall a problem when minimum altitude loss is desired. The inability to reach a well-defined wing stall in some configurations is a result of the large longitudinal control force gradient with speed (stick-free speed stability) that this aircraft displays (i.e., the minimum flying speed is limited by control force and trim authority). The stall recovery technique used during this test program did not use additional engine power during recovery. The altitude lost during recovery would be reduced with the use of full power.

The minimum flying speeds as a function of wing camber position are shown in Figure 15. These speeds are given for an aft center-of-gravity position with boundary layer control on and both idle power and power for level flight, boundary layer control off with the suction holes sealed and idle power, and boundary layer control off with the suction holes open and idle power. The minimum flying speeds are also given for a forward CG position with boundary layer control on and idle power. The minimum speeds obtained with the suction holes open at all wing camber positions and with the suction holes sealed at 30 degrees of wing camber were the result of high control forces and not a well-defined wing stall.

Although the minimum flying speeds with the boundary layer control system operating are of the predicted magnitude, two basic discrepancies are apparent from these data. The decrease in stall speed due to increases in wing camber is small, and the increment in stall speed due to the use of the boundary layer control system is less than anticipated. The reasons for this are not known at this time.

Representative stall maneuver time histories are given in Figures 16, 17, and 18. Figure 16 presents the stall maneuver time history for 30 degrees of wing camber, boundary layer control on, aft CG, and power for level flight at a trim speed of 70 knots. A well-defined break is obtained with this configuration; the left wing drops about 40 degrees, and 200 feet of altitude is lost during recovery. The time history for 0 degrees of camber with the suction holes sealed, idle power, and an aft CG position is shown in Figure 17. A well-defined stall is obtained with little wing drop, and less than 100 feet of altitude is lost in recovery. Figure 18 gives the time history for 0 degrees of wing camber, boundary layer control off with the suction holes open, idle power, and an aft CG position. No wing stall is obtained with this configuration because of the high longitudinal control forces required. A sink rate of 400 feet per minute developed, with some longitudinal oscillations being apparent to the pilot.

Accelerated stalls were performed at 0 and 30 degrees of wing camber, boundary layer control on, and an aft CG position. The stall characteristics were the same as the 1 g stalls.

STATIC LONGITUDINAL STABILITY

The variation of wheel position and force and elevator position with equivalent airspeed for 0, 15, and 30 degrees of camber with an aft and forward CG position and the boundary layer control system operating is given in Figures 19 through 25. The increase in apparent static stability with increases in wing camber, as shown by the control position and force gradients, is illustrated. The change in stability with CG position is less well defined and does not lend itself to any estimate of the neutral point location.

Figures 26, 27, and 28 give the variation of wheel position and force and elevator position with airspeed for an aft CG and boundary layer control system off with suction holes sealed and with them open. A comparison with the corresponding data for boundary layer control on shows some reduction in the displacement and force gradients due to removing the boundary layer control at the lower trim speeds. No change in static longitudinal stability due to the boundary layer control system configuration is seen at the higher cruise speeds.

In general, the force gradients associated with speed changes are large and result in a high dependency upon the longitudinal trim system.

MANEUVERING STABILITY

The maneuvering stability characteristics given in terms of longitudinal control position and force and elevator position as a function of normal load factor are presented in Figures 29, 30, and 31. These data are shown for 30 degrees of wing camber at a trim speed of 71 knots with boundary layer control on and an aft CG position and for 0 degrees of wing camber at a 122-knot trim speed, boundary layer control on, and both an aft and a forward CG position. A measurable difference in the position and force gradients for right and left turns was found with the 30-degree camber configuration at 71 knots. This difference is not apparent with the 0-degree camber configuration at 122 knots.

The nominal force gradient is in excess of 100 pounds per g for the 30-degree camber configuration and is considered to be excessive. The 36 pounds per g gradient shown with the 0-degree camber configuration at 122 knots is high but not excessive for the intended use of the aircraft.

LONGITUDINAL DYNAMIC CHARACTERISTICS

The classic longitudinal short-period characteristics have sufficiently high frequency and damping ratio to render the aircraft motions unobservable on the recorder or by the pilot. One instance of a relatively short period (i.e., less than the phugoid period) oscillation was observed with the 30-degree wing camber configuration at 71 knots. This oscillation is seen in Figure 32. The motion is most noticeable on the normal acceleration and angle-of-attack traces. The period is of the order of 3.5 seconds, and there would appear to be little damping.

The phugoid mode for boundary layer control on, aft CG position, and 30 and 0 degrees of camber is shown in Figures 32 and 33. The 30-degree camber configuration at 71 knots displays a period of oscillation of 16.5 seconds with about 3.2 cycles to damp to half amplitude. The 0-degree camber configuration at 122 knots shows a 33-second period with about 0.82 cycle required for the oscillation to damp to half amplitude.

LATERAL/DIRECTIONAL STATIC STABILITY

The lateral/directional control characteristics in steady-state sideslips are shown in Figures 34, 35, 36, and 37 for boundary layer control on, 71 knots with camber deflections of 0, 15, and 30 degrees as well as 0 degrees of camber at 122 knots. Figures 38 and 39 show the 0-degree camber configuration with boundary layer control off and the suction holes sealed at 83 knots and boundary layer control off with the suction holes open at 122 knots.

For all the test configurations, the static directional stability as shown by the variation in rudder pedal deflection and force with sideslip angle is of the proper sign and essentially linear over most of the range of rudder travel available. There is little change in gradient with wing camber or boundary layer control system configuration. There is an increase in rudder force gradient for the 122-knot trim speed as compared to the 71-knot data. The maximum available sideslip angle is determined by the available rudder deflection, which is limited by rudder force. The available sideslip angles are 13 and 10 degrees for 0 degrees of wing camber at 71 and 122 knots, respectively, and 10 degrees for the 30-degree wing camber configuration. These angles are adequate for the intended use of this aircraft.

The "stick free" and "stick fixed" dihedral effect, as given by the variation in aileron control displacement and force with sideslip angle β, is reasonably linear and is in the direction of a stabilizing dihedral effect. There is a tendency for the aileron force to drop off at the higher deflection angles, but it does not reverse sign. There is little change in characteristics with changing wing camber or boundary layer control system configuration, but the aileron control force gradient does increase with increasing airspeed. Less than 75 percent of the available aileron control deflection is required for the maximum available sideslip angles.

LATERAL CONTROL POWER

The lateral control characteristics are given in terms of the maximum roll rate, lateral control force, and maximum helix angle as a function of lateral control deflection in Figures 40 through 43. Data are given for boundary layer control on and 0 and 30 degrees of wing camber at 71 knots, 0 degrees of wing camber at 122 knots, and 0 degrees of camber with boundary layer control off and the suction holes open at 122 knots. The linear variation of the rolling characteristics with control deflection is illustrated. These characteristics are considered to be more than adequate for this aircraft. No degradation in roll control due to loss of the boundary layer control system is indicated at 122 knots.

Time histories of the rolling maneuvers for boundary layer control on and 30 degrees of camber at 71 knots and 0 degrees of camber at 122 knots are presented in Figures 44 and 45. Figure 46 shows the time history for 0 degrees of wing camber with boundary layer control off and the suction holes open at 122 knots. The 30-degree wing camber configuration gives 30 degrees change in bank angle 1.3 seconds after control input and 55 degrees change after 1.7 seconds. A fair amount of adverse yaw is developed during this maneuver and the resulting small reduction in roll rate is noted. Changes in roll angle of 72 and 85 degrees after 1.3 seconds and 100 and 115 degrees after 1.7 seconds are shown for 0 degrees of wing camber with boundary layer control on and boundary layer control off, respectively. A small buildup in adverse yaw is shown. Adequate aircraft response to removing the aileron control input is shown by the corresponding reduction in roll rate for all configurations.

The ability to roll an aircraft with rudder input alone is of some importance, especially in the low-speed regime near stall. Since the XV-11A maintains aileron control in stall, this characteristic is less significant but is included for the sake of completeness. Figures 47 and 48 show the time history of rudder induced rolling maneuvers for boundary layer control on with 30 degrees of camber at 71 knots and 0 degrees of camber at 122 knots. Although it is not possible to separate the roll due to rudder ($C_{\ell_{\delta_r}}$), the roll due to yaw rate (C_{ℓ_r}), and the roll due to sideslip (C_{ℓ_β}), the correlation between roll rate and sideslip angle implies that the roll is predominantly a result of sideslip. In any case, the test data show the roll control possible with rudder input for both configurations tested.

LATERAL/DIRECTIONAL DYNAMIC STABILITY

Spiral stability characteristics are shown in Figures 49 and 50 for the case of boundary layer control on with 30 degrees of wing camber at 71 knots. Right and left turns are illustrated. Spiral stability is demonstrated with this configuration. Neutral spiral stability is shown for boundary layer control on and 0 degrees of camber at 122 knots in Figures 51 and 52.

The normally heavily damped roll mode is not discernible in the flight test records or to the pilot. No overshoot in roll rate or roll angle due to rapid aileron input is found.

The time histories of the dutch roll mode for 0, 15, and 30 degrees of wing camber and boundary layer control on at 71 knots are presented in Figures 53, 54, and 55. The characteristics for 0 degrees of wing camber with boundary layer control on at 122 knots are shown in Figure 56. The dutch roll mode is considered to be satisfactory, with relatively small changes in oscillating characteristics due to changes in airspeed or wing camber. The damping ratio varies between 0.13 and 0.19 with damped frequencies between 2.0 and 2.6 radians per second. The roll to yaw ratio (ϕ/β) varies between 1.0 and 1.6. The nominal value of the parameter $\zeta_d\omega_d$ is 0.35 radian per second with $\omega_d^2|\phi/\beta|$ about 4 radians per second squared.

TRIM CHARACTERISTICS

The changes in longitudinal trim resulting from various configuration changes are shown in Figures 57 through 64. Time histories of elevator deflection, longitudinal control deflection and force, flight altitude, and airspeed are presented. The longitudinal trim tab setting was not changed in these maneuvers after the initial straight and level trim condition was established prior to the introduction of each configuration change. The trim tab may normally be used 5 seconds after the configuration change has been initiated; the use of the trim tab will significantly reduce the forces required.

The ability of the pilot to maintain aircraft altitude during the application of the high power setting while initially trimmed at 71 knots with 0 and 30 degrees of wing camber is shown in Figures 57 and 58. The high forces associated with the resulting speed change are illustrated. The power was reduced after 20 seconds with the 30-degree camber configuration; however, sufficient time is shown to demonstrate the significant characteristics.

The effect of wing camber changes is shown in Figures 59 and 60. A trim speed of 71 knots was used with these maneuvers; the resulting large trimming forces are shown. The pilot was attempting to hold a constant airspeed during the camber change. The camber change is more rapid when camber is being reduced, and even the use of the trim tab 5 seconds after the configuration change would not prevent trim forces in excess of 50 pounds from being reached.

The control deflections and forces required when accelerating from trimmed flight at 62.2 knots with 30 degrees of camber to a high flight speed with 0-degree camber are shown in Figure 61. The pilot was trying to maintain a constant altitude during this maneuver. The high control forces associated with the camber change are the dominating feature of this maneuver. The maximum equivalent airspeed for flight with wing camber deflections other than 0 degrees was arbitrarily set at 100 knots for this test program. For this reason the pilot had to use a camber change rate that would allow this restriction to be satisfied. This consideration plus the tasks associated with operating the flight test instrumentation resulted in a high pilot work load and made the maneuver difficult to perform. A more moderate rate of change of wing camber would reduce the rate at which the control forces are changing and would reduce the maximum force.

Figure 62 shows the deceleration maneuver from 0 degrees of wing camber at 122 knots to a low speed at 30 degrees of camber while attempting to maintain altitude. The pilot was unable to use the control force transducer on this maneuver because of the high work load. The camber change was started in excess of 100 knots, and once started, a lower rate of change was possible when decelerating as compared to the acceleration maneuver. A more gradual change in elevator position is seen, although a maximum control force of about 50 pounds was reported by the pilot.

The effect of a runaway trim tab motor at 122 knots trim speed with 0 degrees of camber is shown in Figures 63 and 64. Figure 63 illustrates the tab moving into the nosedown position. Control forces in excess of 100 pounds are required to hold airspeed under this condition. Figure 64 shows the tab moving into the nose-up position with control forces in excess of 50 pounds being required.

PILOT COMMENTS

PREFLIGHT INSPECTION

Inspection is readily accomplished from ground level except for checking the engine oil level, which requires a step up to the top of the engine compartment.

COCKPIT LAYOUT

1. The seating is adequate, although the angle between the foot and the rudder pedals makes it difficult to steer using wheel steering alone without depressing the brake.

2. The side-by-side seating is snug with little excess room.

3. The power quadrant position is reasonable, however, small pilots require the seat to be pulled forward such that the quadrant is against the seats.

4. The instrumentation layout is adequate for research purposes.

5. The visibility in all directions except straight aft and down is excellent, although there is slight image distortion in the highly curved front windshield.

PILOT ENTRANCE AND EXIT

1. The entrance doors need to be propped open or held open by the ground crew, which is bothersome.

2. Pilot entrance is smooth and easy when the correct method is used. The pilot, facing forward with his head lowered, places the left foot on the front of the left undercarriage fairing and steps into the cockpit with the right foot.

3. The seat belt arrangement is awkward, as the shoulder straps and inertial reel are in the aft part of the cabin on the sidewall, out of the reach of the pilot.

4. The door-locking arrangement leaves much to be desired since simultaneous, precise alignment

of two pins with their respective holes is required. An additional securing pin must then be inserted in the front door frame. In flight, the door is jettisoned either by releasing the front pin and unlatching the door or by pulling the hinge pin from the hinge at the top of the door. Neither arrangement is desirable.

5. Aircraft egress on the ground is accomplished by a reversal of the entrance technique described in Item 2.

6. Emergency exit is accomplished by jettisoning the door and rolling over the door sill to the undercarriage and then clear of the aircraft.

ENGINE START

The engine is started with the propeller in full coarse pitch and the N_2 control set at 75 percent. Starting procedure is reasonable.

ENGINE CONTROLS

The N_1 and N_2 controls work in the accepted sense, i.e., forward to increase power. The propeller control motion is similar to the propeller control in a reciprocating engine, i.e., forward for takeoff and back to increase propeller pitch angle. This is opposite from normal turboprop practice, and a change in this linkage would be desirable.

The standard propeller control system was plagued with stability problems during this test program. Propeller surging occurred quite often, and the system reliability was such that the pilot reverted to the beep switch backup system most of the time. The beep switch, located forward of the propeller control levers, is spring loaded in the off position and supplies power directly to the propeller pitch change motors. This system works reasonably well when increases in propeller pitch (increased power) are desired, since the blade loading is in the opposite direction and a moderate pitch change rate is obtained. However, when reduced pitch is required the rate of change is too rapid. Precise control is difficult in this direction; the smallest increment in RPM that could be achieved was about 4 percent.

This situation made it difficult to obtain specified trim points.

ENGINE GROUND RUNNING

The engine can be operated at ground idle for approximately 15 minutes before the oil temperature approaches the prescribed redline. Operation at higher power settings appreciably reduces the possible running time. Full throttle operation is not possible without a tie down because of brake slip and the front of the undercarriage touching the ground.

TAXI

The ground handling is quite good with a 21-foot turning radius with steering alone and 13 feet with differential braking. Taxiing over rough ground excites a slight longitudinal porpoising motion. High-speed ground operation above 70 knots excites a light, lateral vibration of the front wheels.

TAKEOFF

Directional control on the takeoff roll is good even in crosswinds up to 15 MPH. The aircraft was operated as if the boundary layer system was off and the holes were open. This operation technique ensured operation safety similar to a conventional single-engine aircraft. Due to the large changes in trim required with camber setting, care has to be taken to ensure the proper trim setting for the camber position selected prior to takeoff. Aircraft rotation at 80 knots and climb-out at 90 knots using zero wing camber gives satisfactory takeoff performance with ground rolls of approximately 2000 feet and climb rates of approximately 1000 feet per minute. During the takeoff roll, it is necessary to monitor engine torque and to adjust with the propeller beep switch to stay within engine limits. The torque situation is adequate for a research aircraft but would not be acceptable for an operational STOL aircraft.

CLIMB-OUT

Climb rates exceeding 1000 feet per minute at an indicated airspeed of 100 knots are usual. Power settings close to the engine limits are not normally used in the climb because of the coarseness of the propeller pitch change system. Visibility in the climb is good at 100 knots with zero camber, however, at speeds just above the stall, the forward visibility is slightly reduced.

CRUISE

In the high-speed cruise condition, the aircraft flies in a fuselage-level attitude which ensures good visibility. The maximum level speed of the aircraft is limited by propeller stall, which gives considerable vibration at airspeeds above 150 knots.

Level-flight accelerations from the 30-degree camber position to high-speed cruise conditions demand a high pilot work load for maximum acceleration due to the changing trim conditions with varying flap and power settings. This situation is aggravated by the fact that torque is a function of airspeed, and the engine conditions must be continually monitored. Increasing camber gives a nose-up pitching moment, and out of trim stick forces can be of the order of 50 pounds. The effect of a runaway trim tab would be serious because of the resulting high stick forces.

STALLS

The stalling characteristics of the aircraft in all configurations are quite mild, generally with a clean break and less than 20 degrees of wing drop. There is no stall warning in any configuration at low power settings and very slight buffet, 1 or 2 knots above stall, at large power settings. This is insufficient warning for an operational aircraft. Due to the high stick force gradient with speed, the aircraft must be trimmed close to the stall speed to ensure a clean break. Adequate aileron control is available through the stall, and no pitch-up or spin tendencies were noted during the stall program.

STATIC STABILITY

The stick-free static longitudinal stability of the aircraft is such that the aircraft is essentially flown using the trim tab. The stick force gradients with speed are excessive. The high control friction breakout forces measured on the ground are not apparent in flight.

The lateral/directional static stability indicates reasonably linear gradients. The maximum sideslip angle is limited by rudder control forces, which may be due to the front wheel steering interconnect.

MANEUVERING STABILITY

The stick force per g gradients are excessive in the 30-degree camber condition (i.e., greater than 100 pounds per g) and high in the 0-degree camber condition (i.e., greater than 30 pounds per g).

LATERAL CONTROL POWER

The aircraft has very acceptable lateral control power in all configurations and can demonstrate roll rates greater than 90 degrees per second, which are more than adequate for this type of aircraft. However, the usefulness of this lateral control power is reduced by the large values of wheel rotation required. The adverse yaw associated with these rolling maneuvers is reasonable and is less than 5 degrees.

DYNAMIC STABILITY

In the cruise conditions, the period of the phugoid is about 33 seconds and is well damped, with the aircraft returning to within 1 knot of the original trim condition. The phugoid period in the landing conditions is 16.5 seconds and is lightly damped; this does not present a problem during the approach.

No short-period oscillations can be detected in any configuration with the exception of one flight at 71 knots and 30 degrees camber, which is described in this report. This condition was not repeatable.

The aircraft has a positive dihedral effect, and the rudders are quite adequate to induce rolling. The aircraft is neutrally stable in the spiral mode in the cruise condition and spirally stable in the approach configuration.

The dutch roll mode has a roll to yaw ratio of approximately 1.0 and is well damped in all configurations.

DESCENT

Descent rates greater than 2000 feet per minute at 80 knots with 30 degrees camber and the propeller in reverse thrust can be readily achieved. The aircraft demonstrates adequate longitudinal and lateral/directional control in this condition, and the effect of the increase in drag at the rear of the aircraft is stabilizing in turbulent air.

APPROACH AND LANDING

The use of the reverse propeller thrust is successful for glide path control and for increasing the rate of sink of the aircraft. Normal approach speeds of 90 knots with 15 degrees camber give acceptable handling characteristics with regard to stick force gradients and trim characteristics. Full reverse pitch is not used in the approach until the aircraft is relatively close to the runway. The nose-up pitching moment associated with the reverse thrust assists in the flare maneuver. The aircraft touches down on the two rear wheels and with the use of reverse thrust and brakes can stop in short distances.

GENERAL COMMENTS

The XV-11A is an easy aircraft to fly, and it has adequate performance and stability for its research mission. The control problems associated with the high stick force gradients could be corrected by aerodynamic balance. A new propeller capable of absorbing the power available would appreciably increase the level-flight speed envelope.

The ventilation in the cabin is inadequate. The combination of large transparent areas and lack of thermal insulation of the engine compartment results in excessive cockpit temperatures during the summer months.

The proximity of the engine compartment and the boundary layer control system blower results in a very high noise level in the cabin. The sound level is such that tightly fitted helmets and shielded microphones (as provided in an oxygen mask) are necessary for any radio communications. In general, radio communications are unsatisfactory. It is possible that in addition to the ambient noise, problems associated with static charge buildup on the fiber glass structure or a malfunctioning radio contribute to this problem.

CONCLUSIONS

1. The XV-11A aircraft demonstrated sufficient performance, stability and control for conducting low and medium speed aerodynamic research; however, the following deficiencies were evident:

    a. marginal propeller pitch control system

    b. propeller deficiency due to high blade loading

    c. high longitudinal and directional control force gradients

    d. inadequate longitudinal trim capability

    e. unsatisfactory radio communications

    f. excessive cabin noise and heat

2. Although low stall speeds were demonstrated, the contribution of the boundary layer control system was less than anticipated.

3. The stalls in all configurations were characterized by a clean break and slight wing drop. Adequate aileron control was available through the stalls and no pitch-up or spin tendencies were noted. The stall warning was less than adequate.

4. The following conclusions are made based on the operational experience obtained over the course of four years of testing the XV-11A aircraft. These conclusions are qualitative in nature and are an attempt at providing a limited evaluation of the merits of the various unique features of the XV-11A which were not individually considered in the present test program.

    a. The use of fiber glass reinforced plastic construction can provide a vehicle with complex surface curvature and a minimum of protuberances within the restrictions of a moderate cost, limited production construction program.

    b. The difficulties experienced with the structure of the XV-11A were associated with the use of bonded joints between the stressed skin and the many ribs, frames and stiffeners. Predicting the structural integrity of these joints and controlling the quality of these joints during construction represents a major obstacle to the extensive use of this material in a limited production program.

c. The state of the art in the use of distributed suction for high-lift boundary layer control is not adequate for predicting the necessary suction distribution. A satisfactory system will require a significant amount of development testing.

d. The use of a boundary layer control system blower driven by the main propulsion system is satisfactory when used with a reliable constant speed propeller system. An operational system would probably require some form of emergency drive to provide limited duration power to the blower in the event of an aircraft power or blower drive failure during STOL operations.

e. Failure of the boundary layer control system in cruise flight does not present a flight hazard. Non-STOL operations are considered to be acceptable in this mode.

f. No problems were experienced with the variable wing camber system. Full realization of the potential advantages of this feature would require additional refinement of the boundary layer control system.

g. The shrouded propeller provides an effective means for increasing the low-speed thrust of a propeller. The use of the shroud presupposes that it is not possible to significantly increase the propeller diameter and that an aft mounted propeller is acceptable.

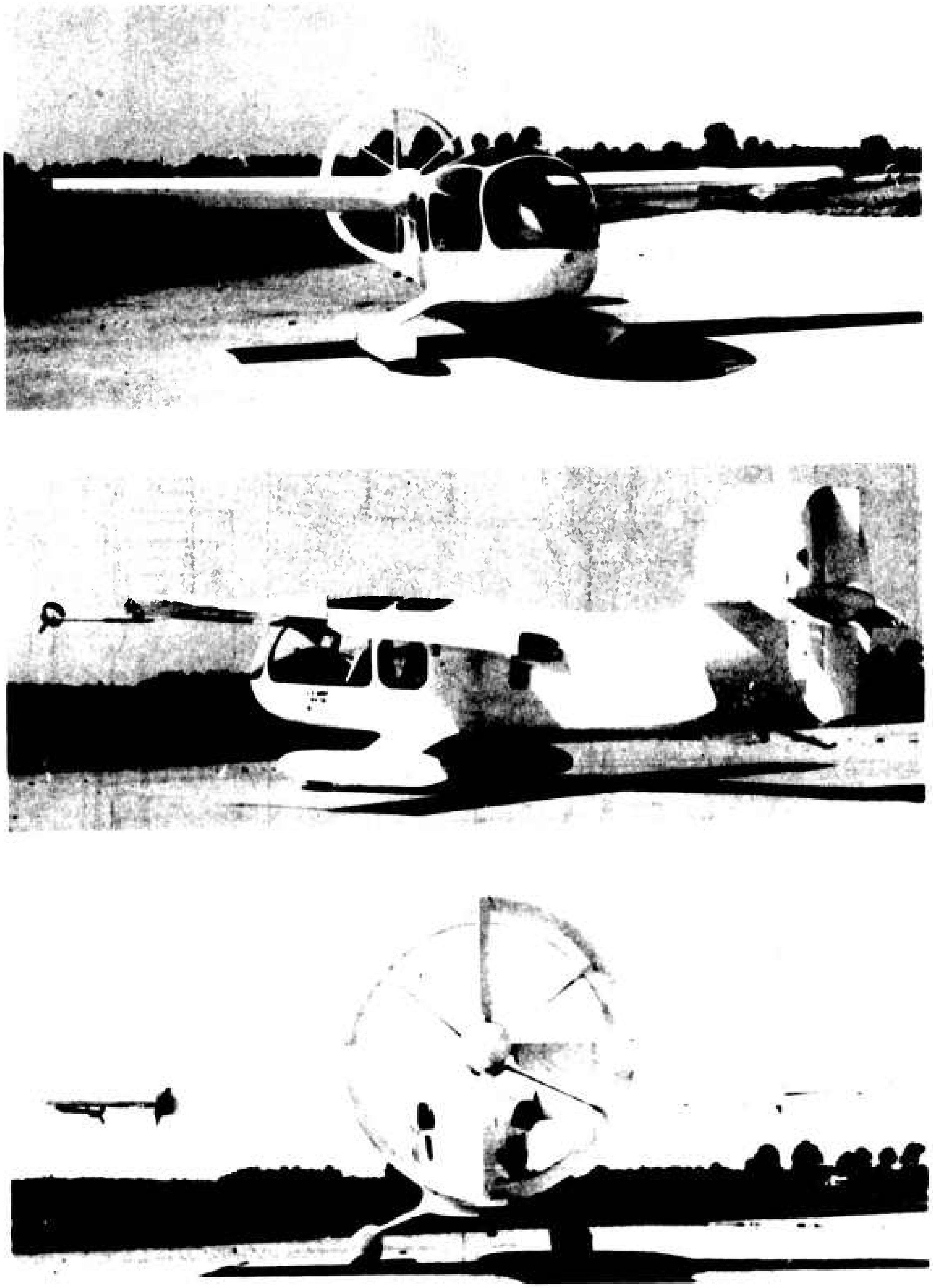

Figure 1. XV-11A External Views.

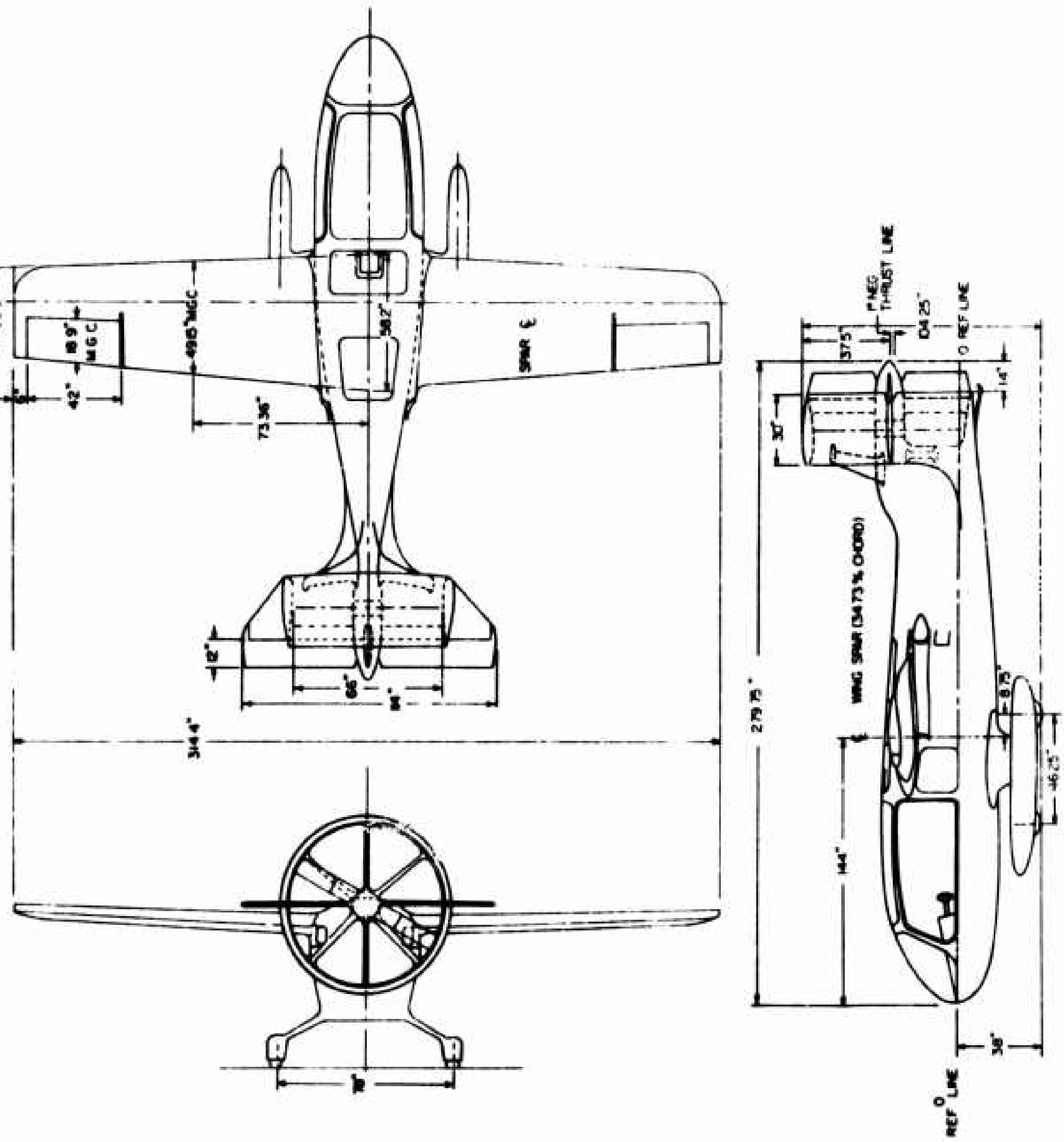

Figure 2. XV-11A, Three-View.

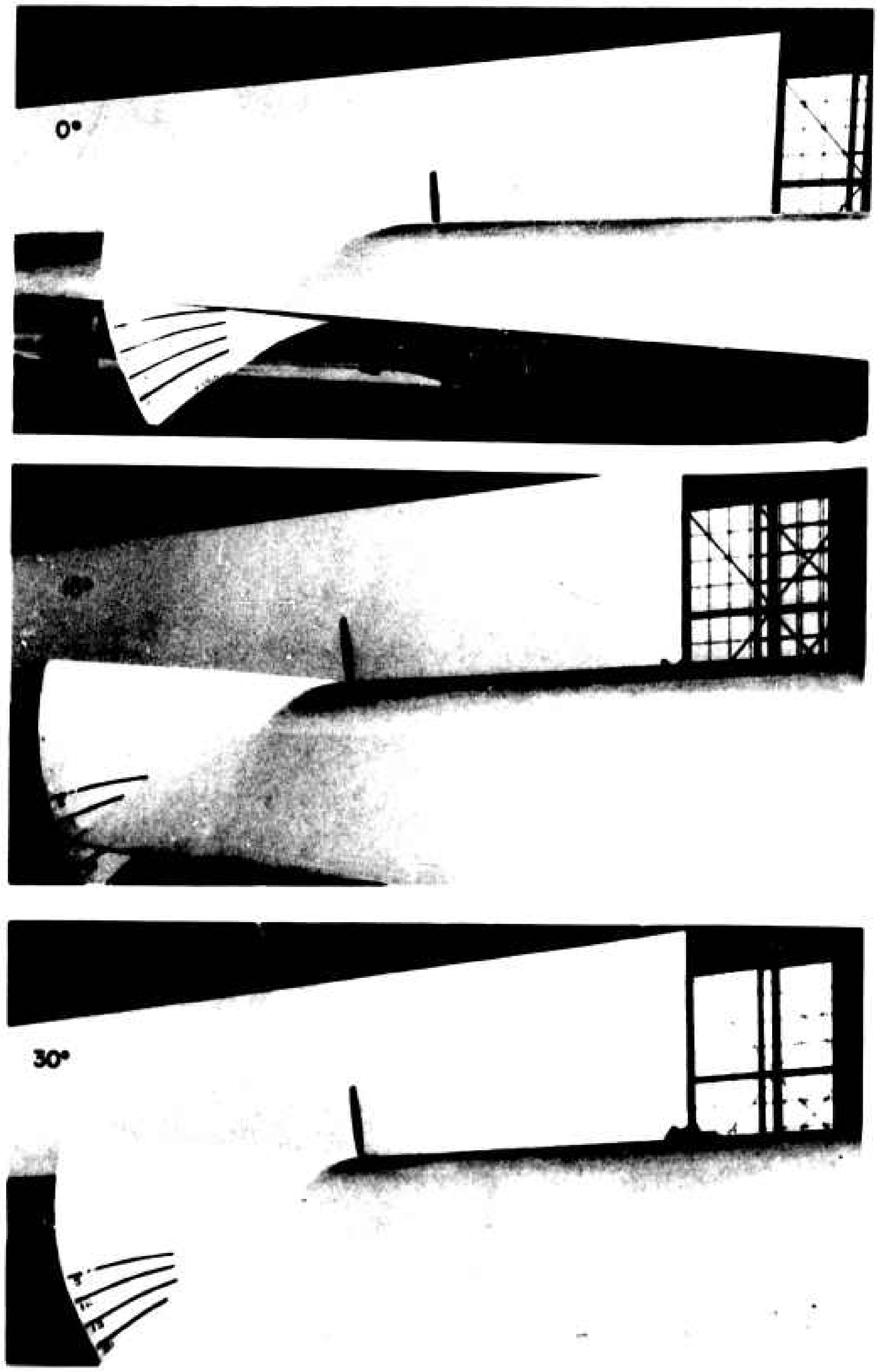

Figure 3. Variable-Camber Wing.

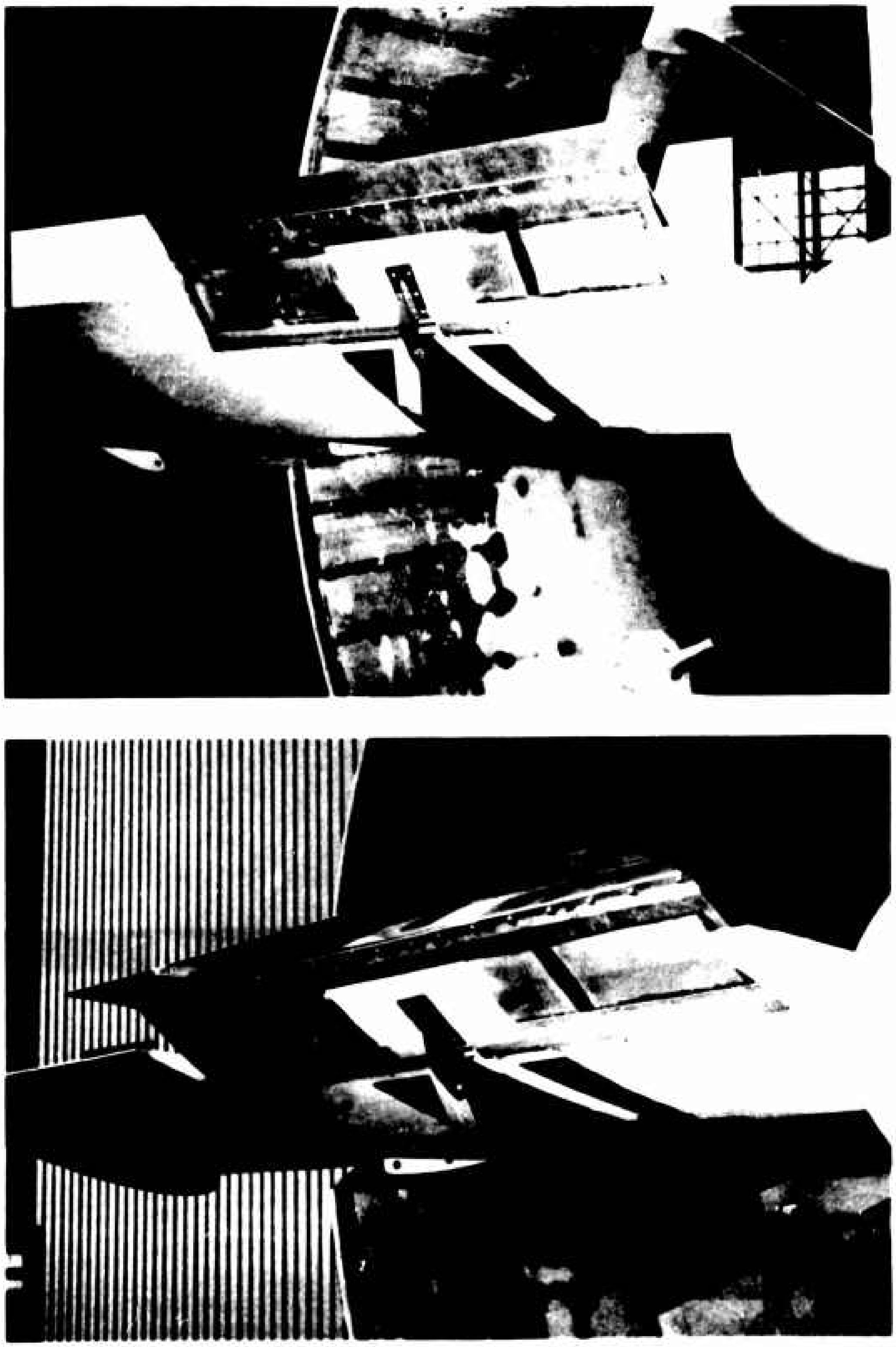

Figure 4. Elevator Trim Tab Modifications.

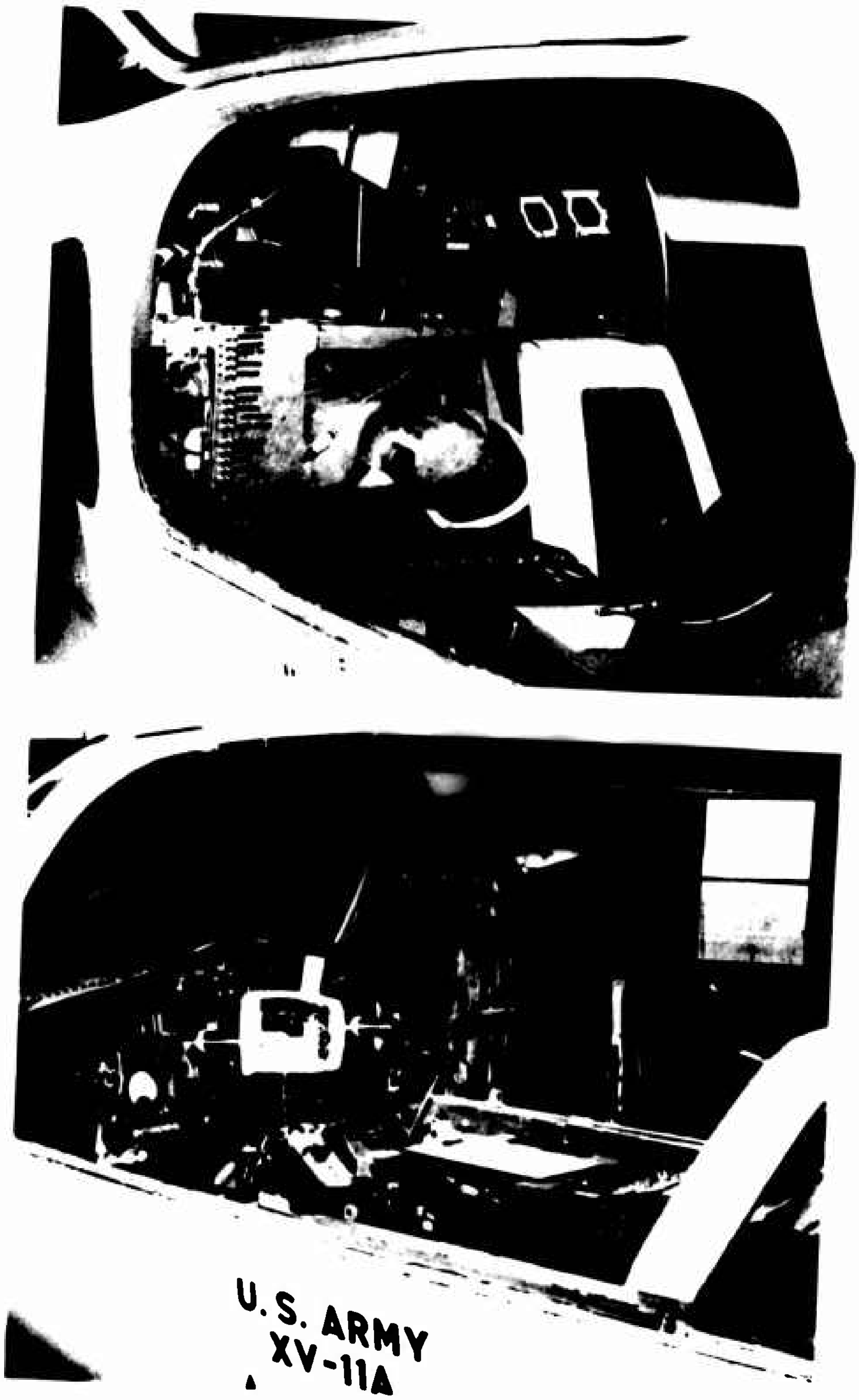

Figure 5. General Arrangement of Instrumentation in Cockpit.

Figure 6. Flight Test Boom.

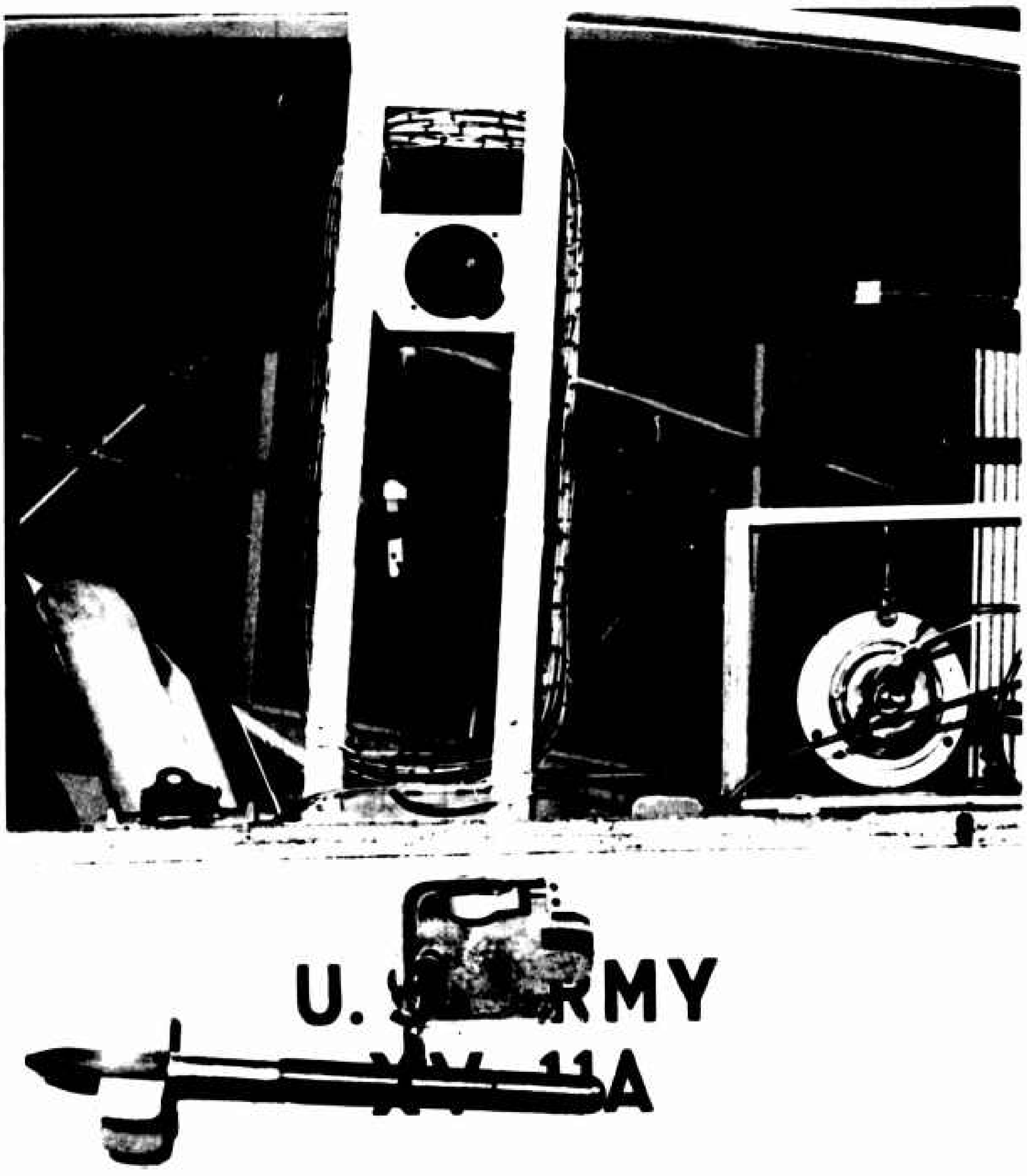

Figure 7. Trailing Static Bomb.

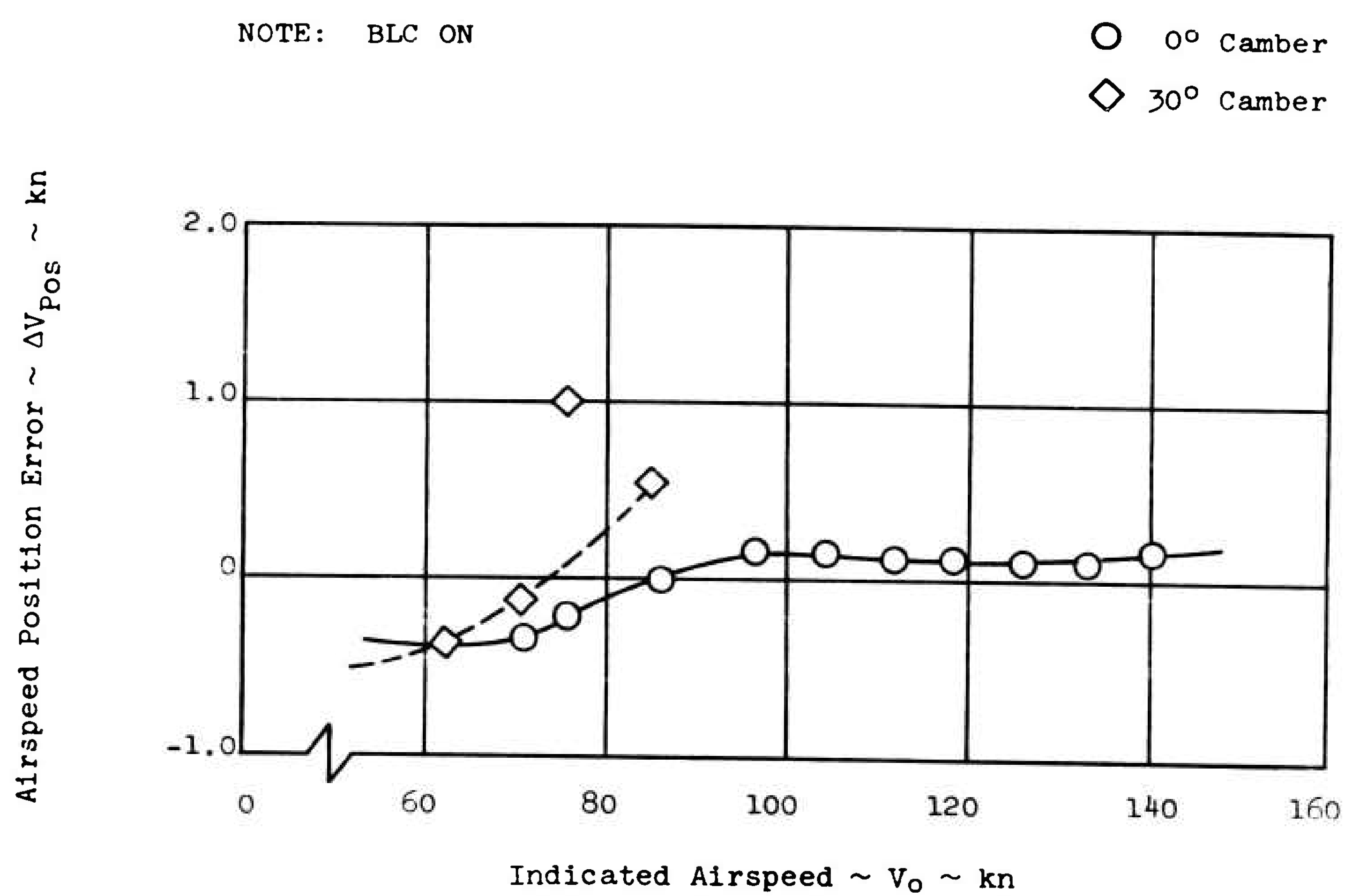

Figure 8. Airspeed Calibration.

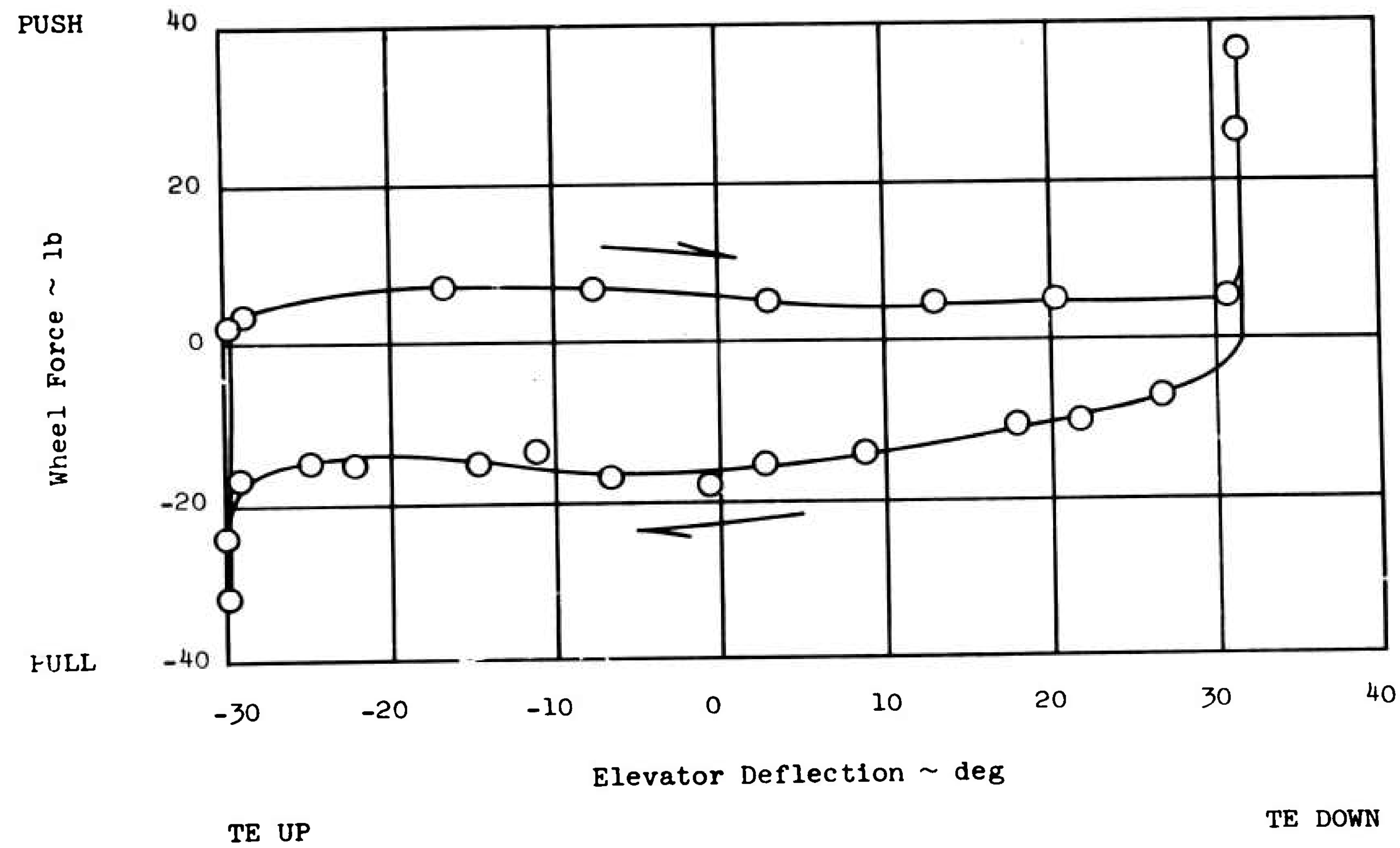

Figure 9. Longitudinal Control Friction Characteristics.

NOTE: 3000 Feet Pressure Altitude, Standard Conditions

○ 0° Camber, BLC ON

△ 15° Camber, BLC ON

◇ 30° Camber, BLC ON

□ 0° Camber, BLC OFF/SEALED

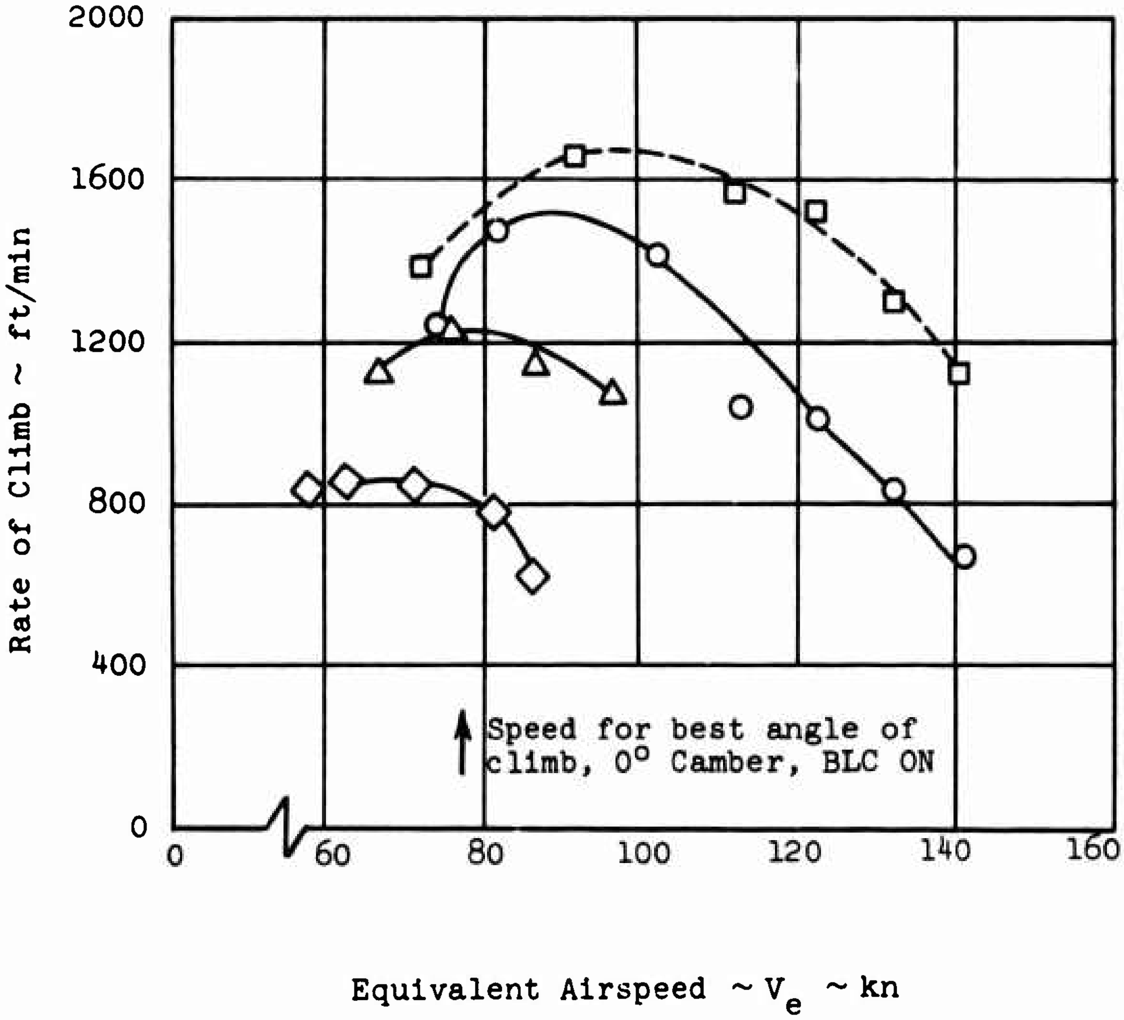

Figure 12. Climb Performance.

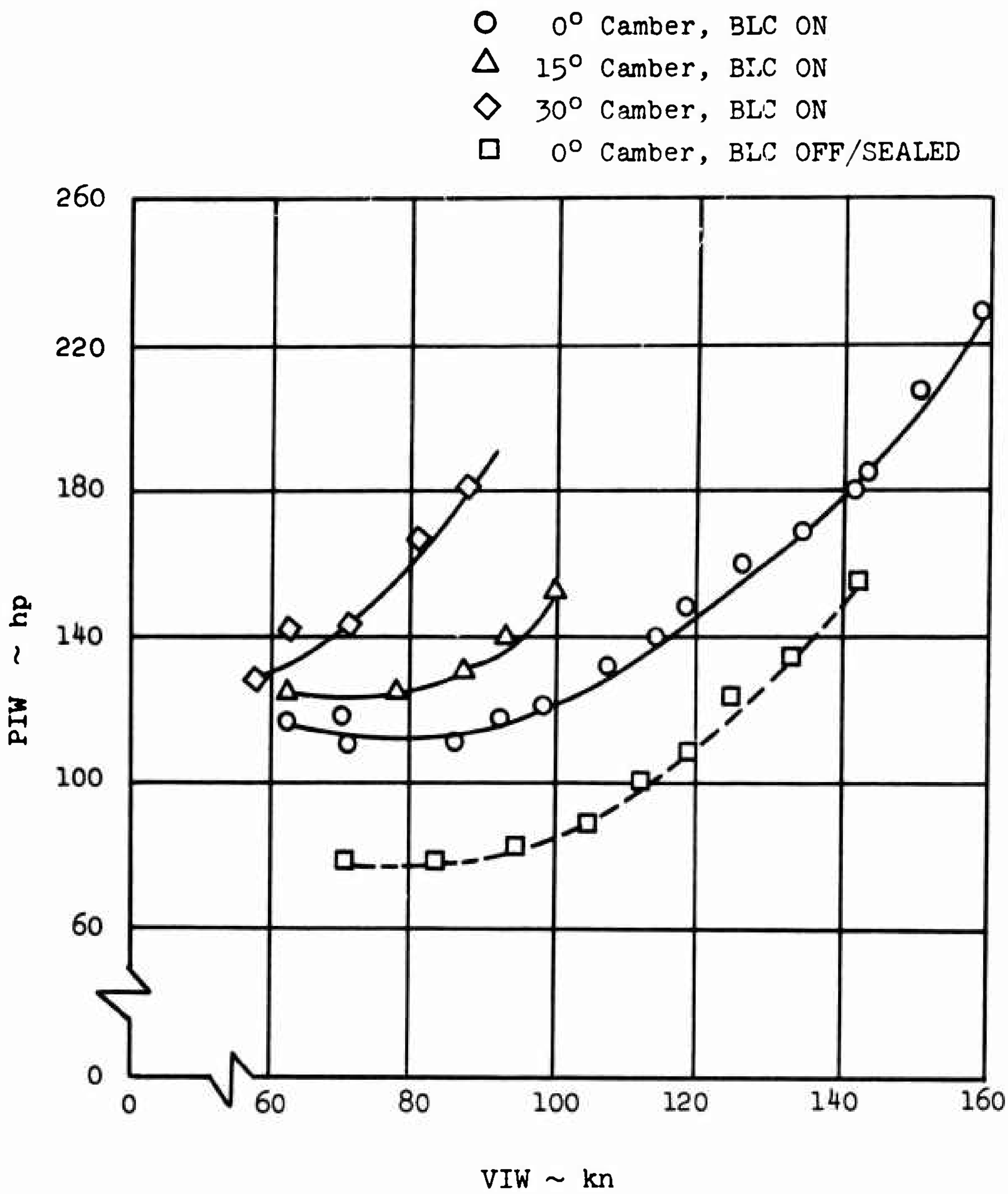

Figure 13. Speed-Power Polars.

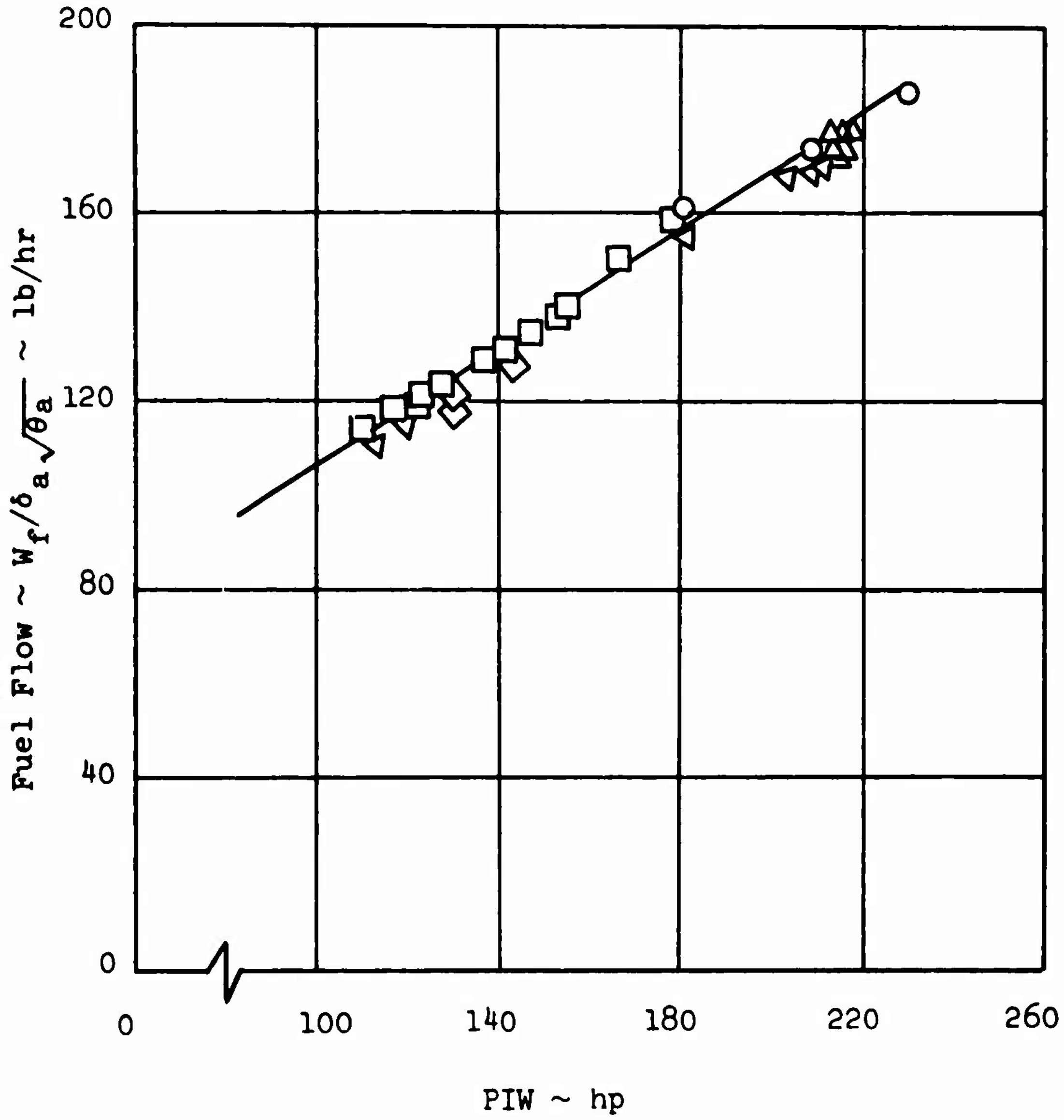

Figure 14. Fuel Flow Characteristics.

NOTE: AIRCRAFT GROSS WEIGHT 2630 POUNDS; FOR BLC ON, N_2 WAS MAINTAINED AT 95.5 PERCENT.

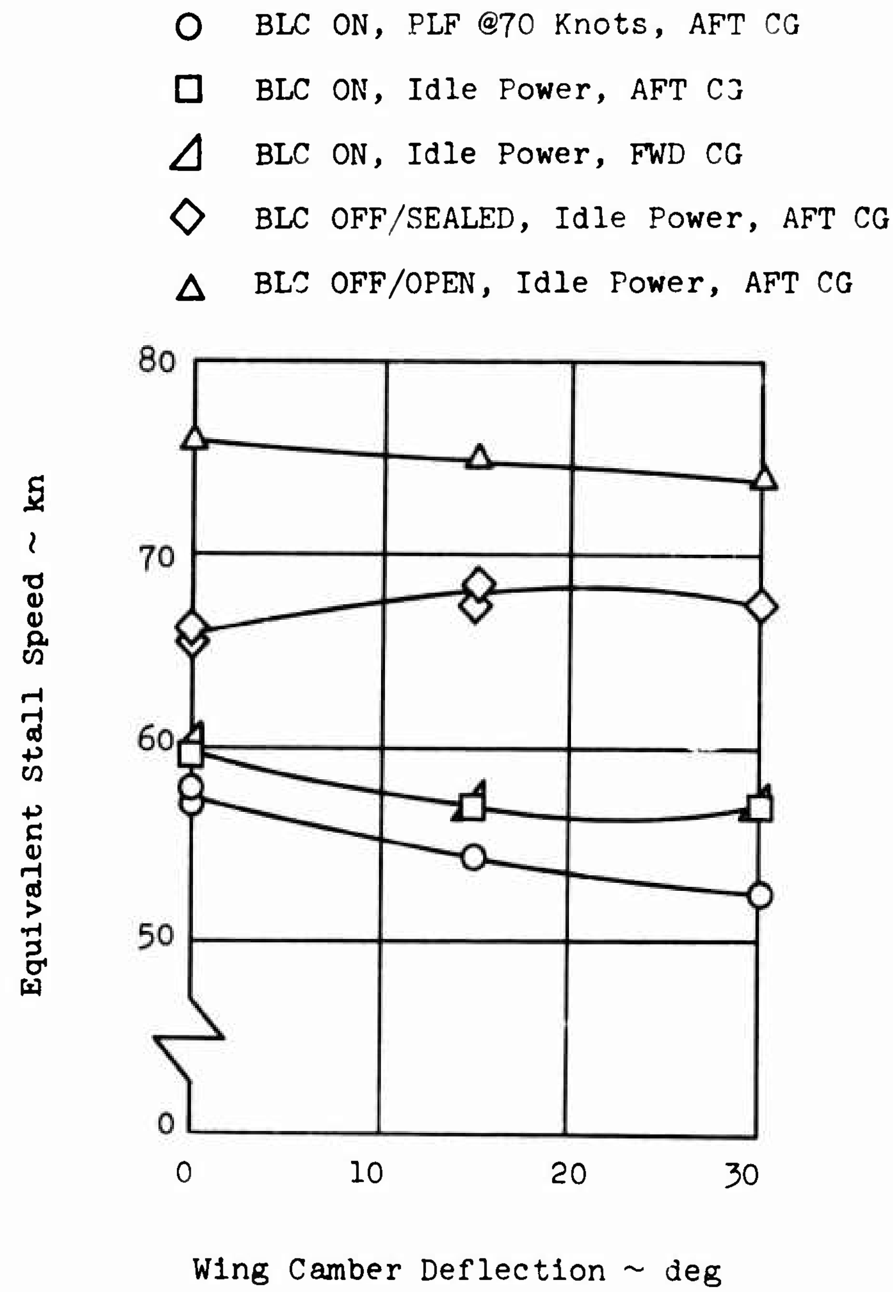

Figure 15. Minimum Flying Speeds.

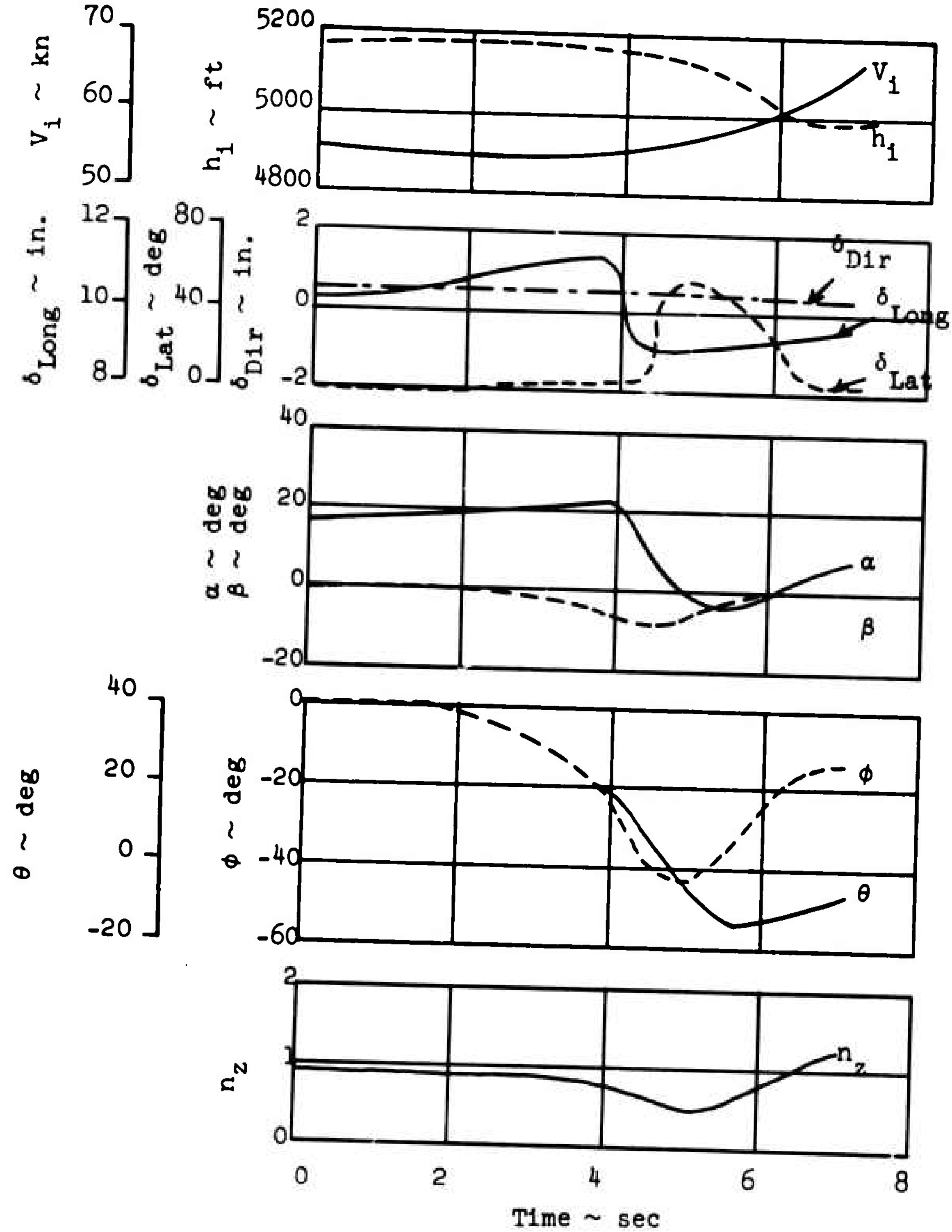

Figure 16. Stall Time History, 30° Camber, BLC ON, PLF at 70 Knots, AFT CG.

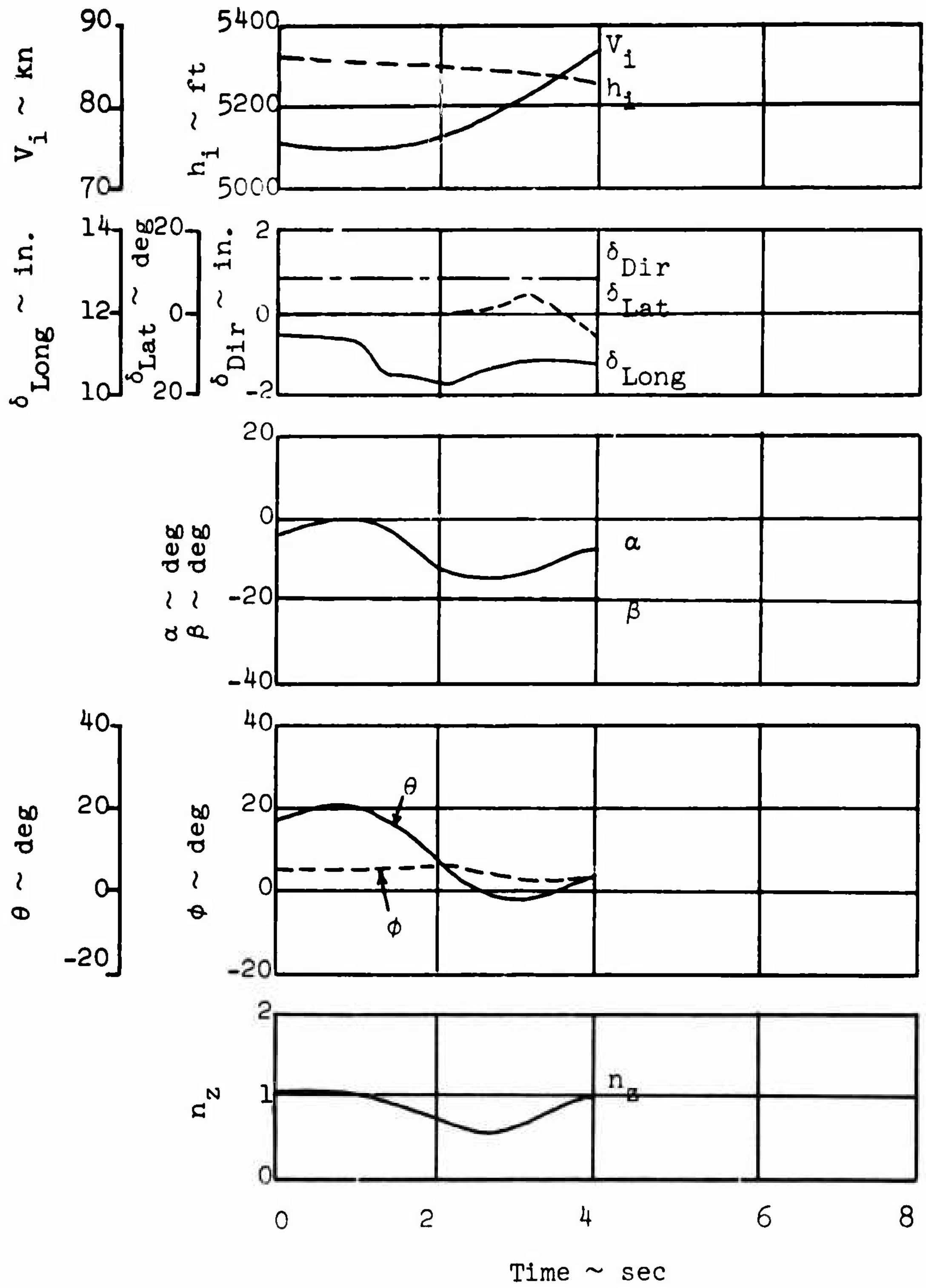

Figure 17. Stall Time History, 0° Camber, BLC OFF/SEALED, Idle Power, AFT CG.

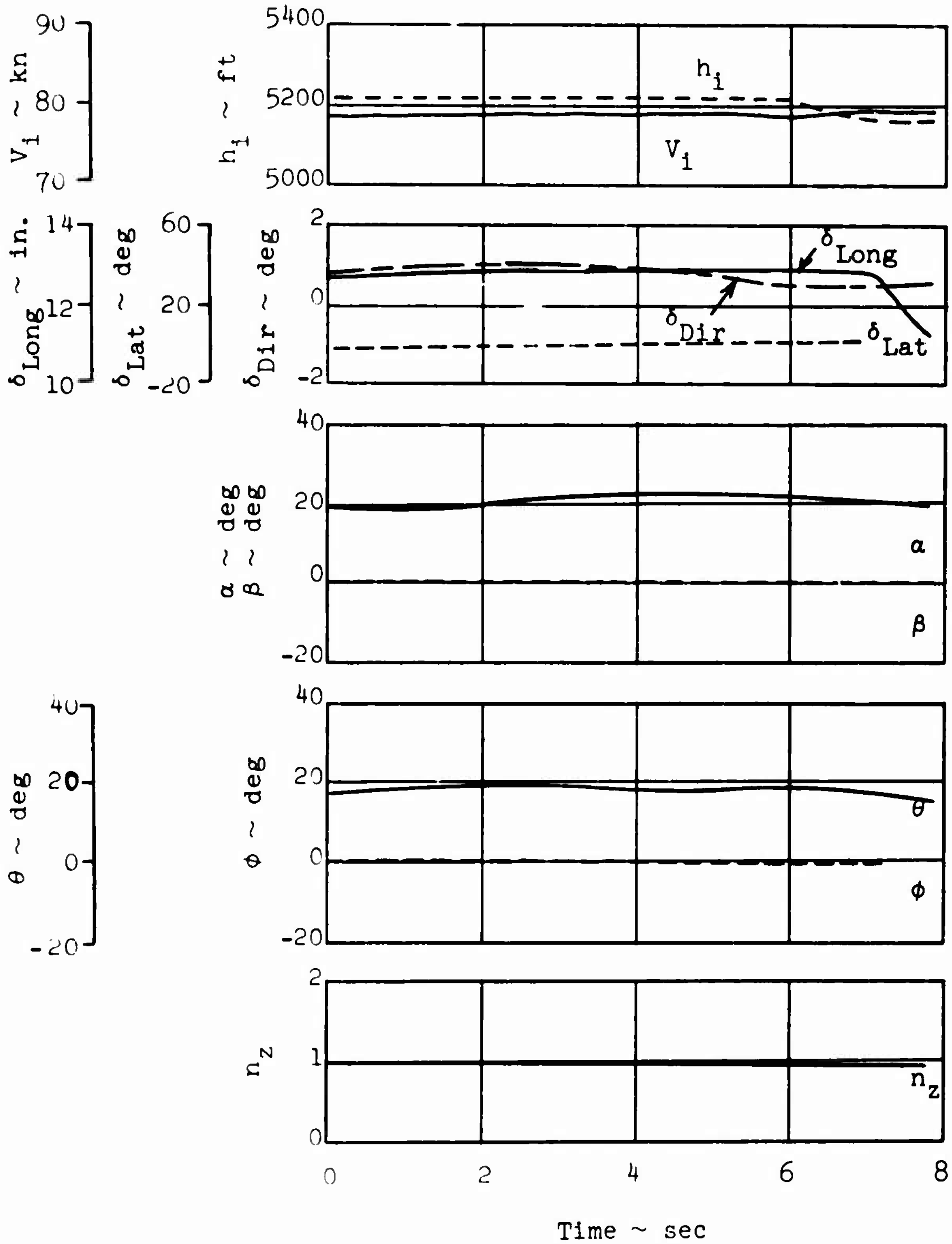

Figure 18. Stall Time History, 0° Camber, BLC OFF/OPEN, Idle Power, AFT CG.

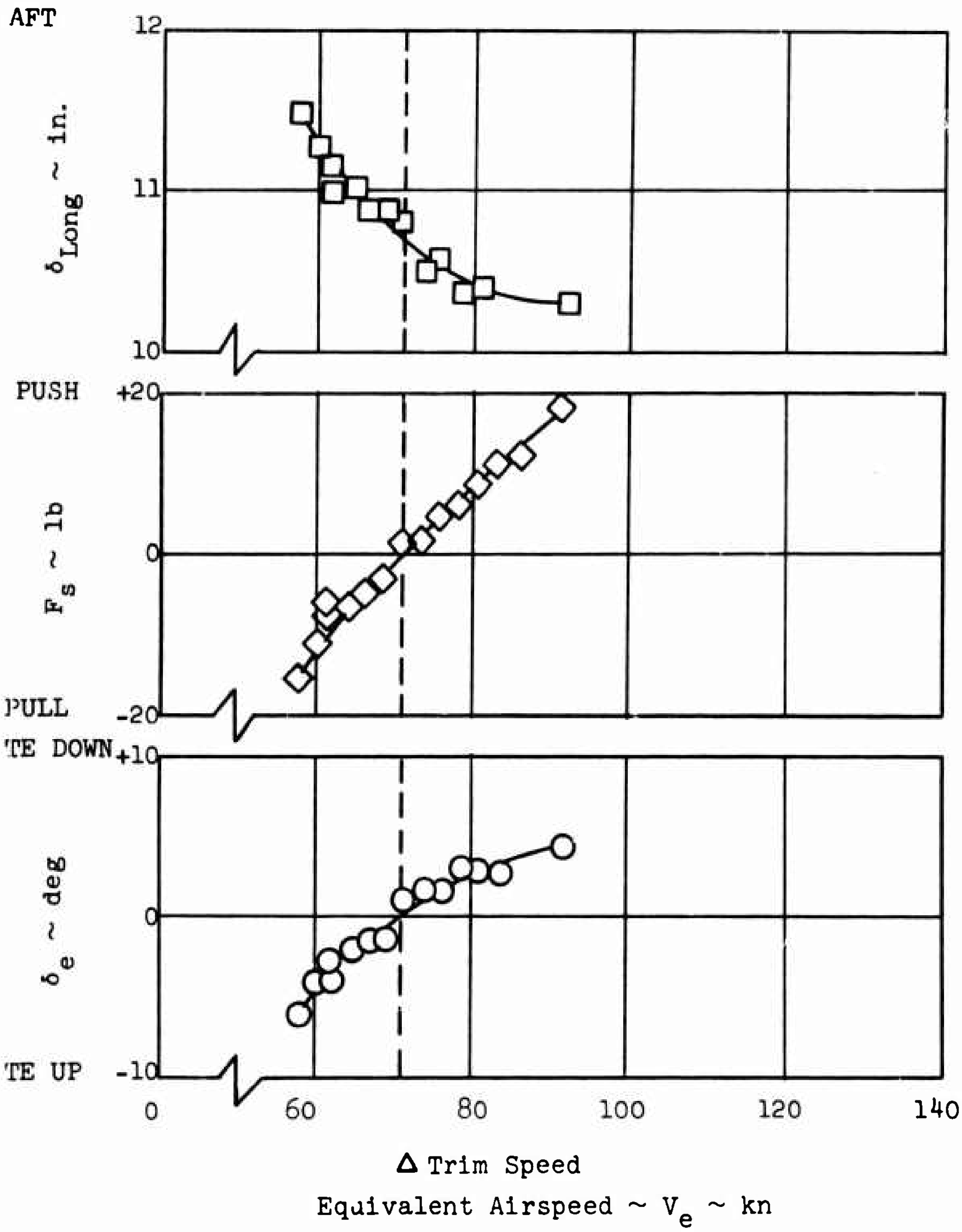

Figure 19. Static Longitudinal Stability, 0° Camber, 71 Knots Trim Speed, BLC ON, AFT CG.

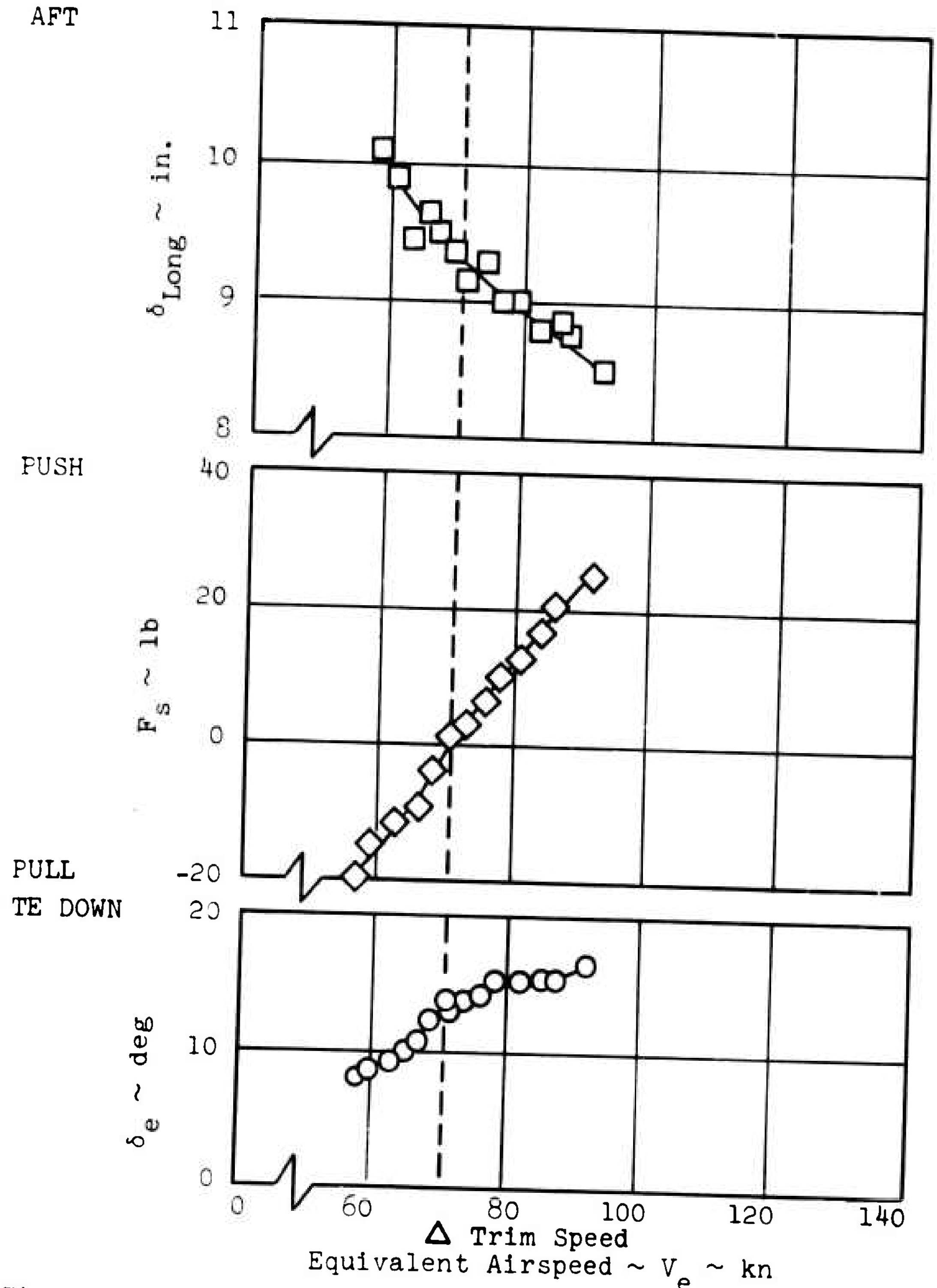

Figure 20. Static Longitudinal Stability, 15°, 71 Knots Trim Speed, BLC ON, AFT CG.

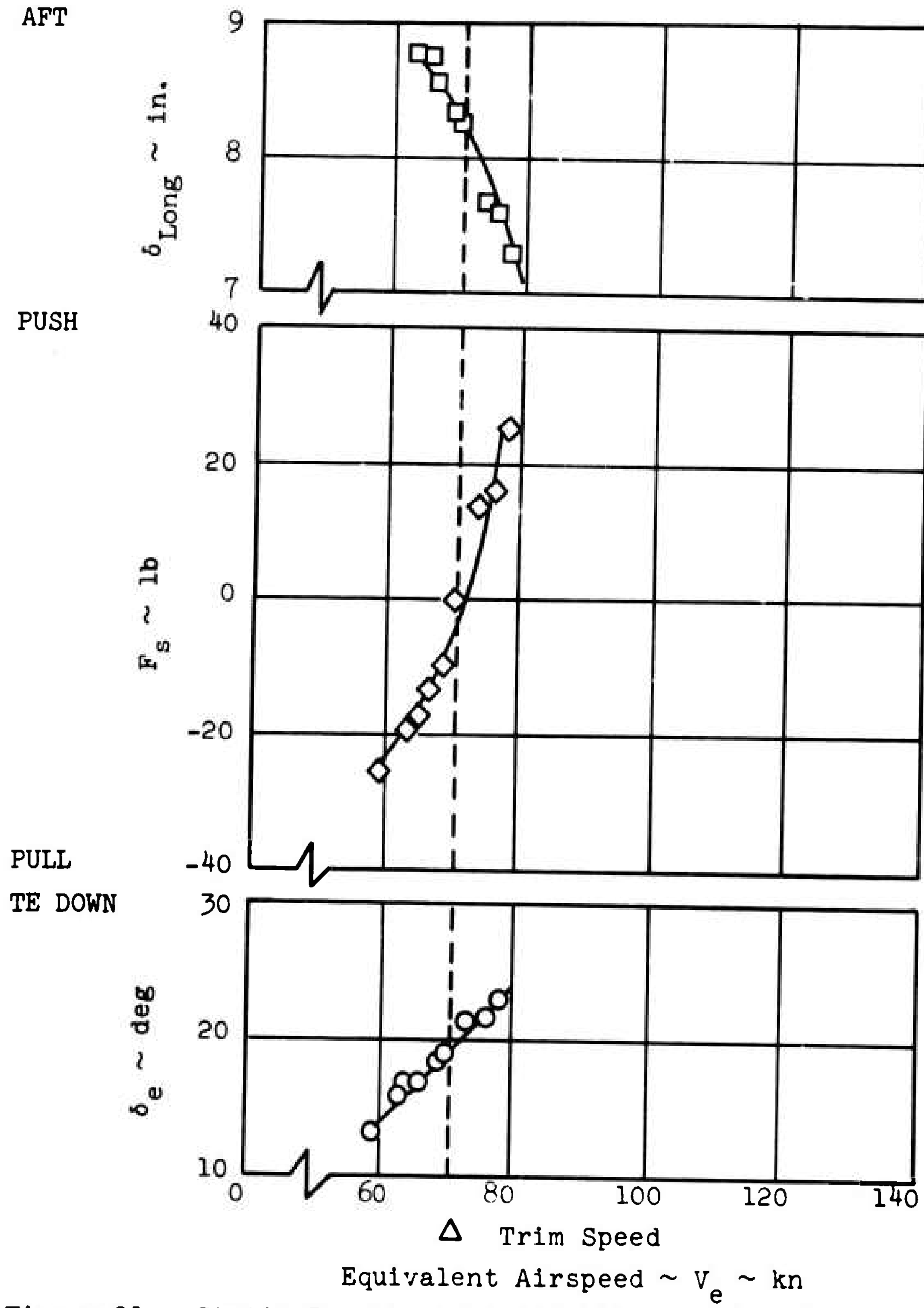

Figure 21. Static Longitudinal Stability, 30° Camber, 71 Knots Trim Speed, BLC ON, AFT CG.

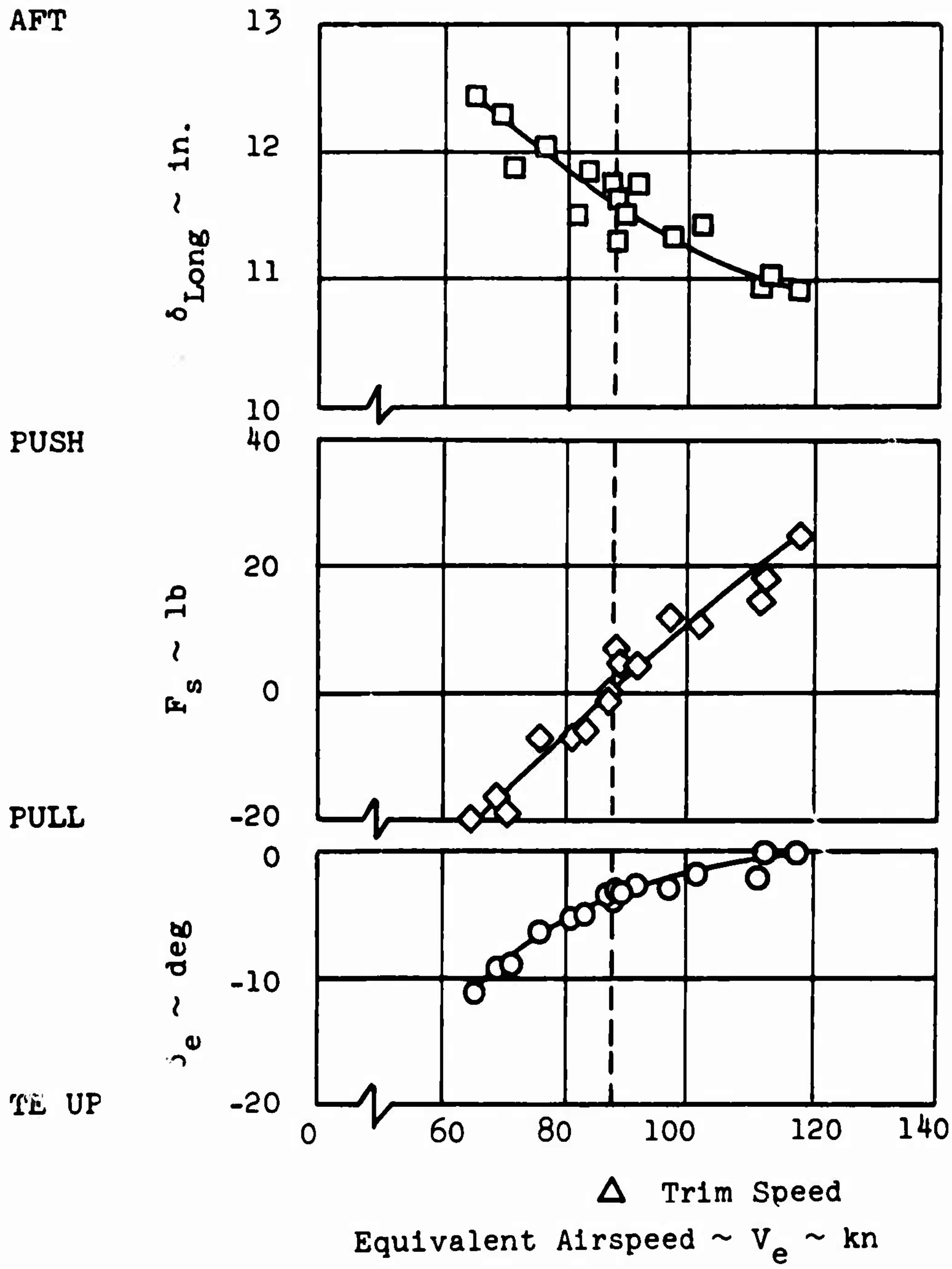

Figure 22. Static Longitudinal Stability, 0° Camber, 88 Knots Trim Speed, BLC ON, FWD CG.

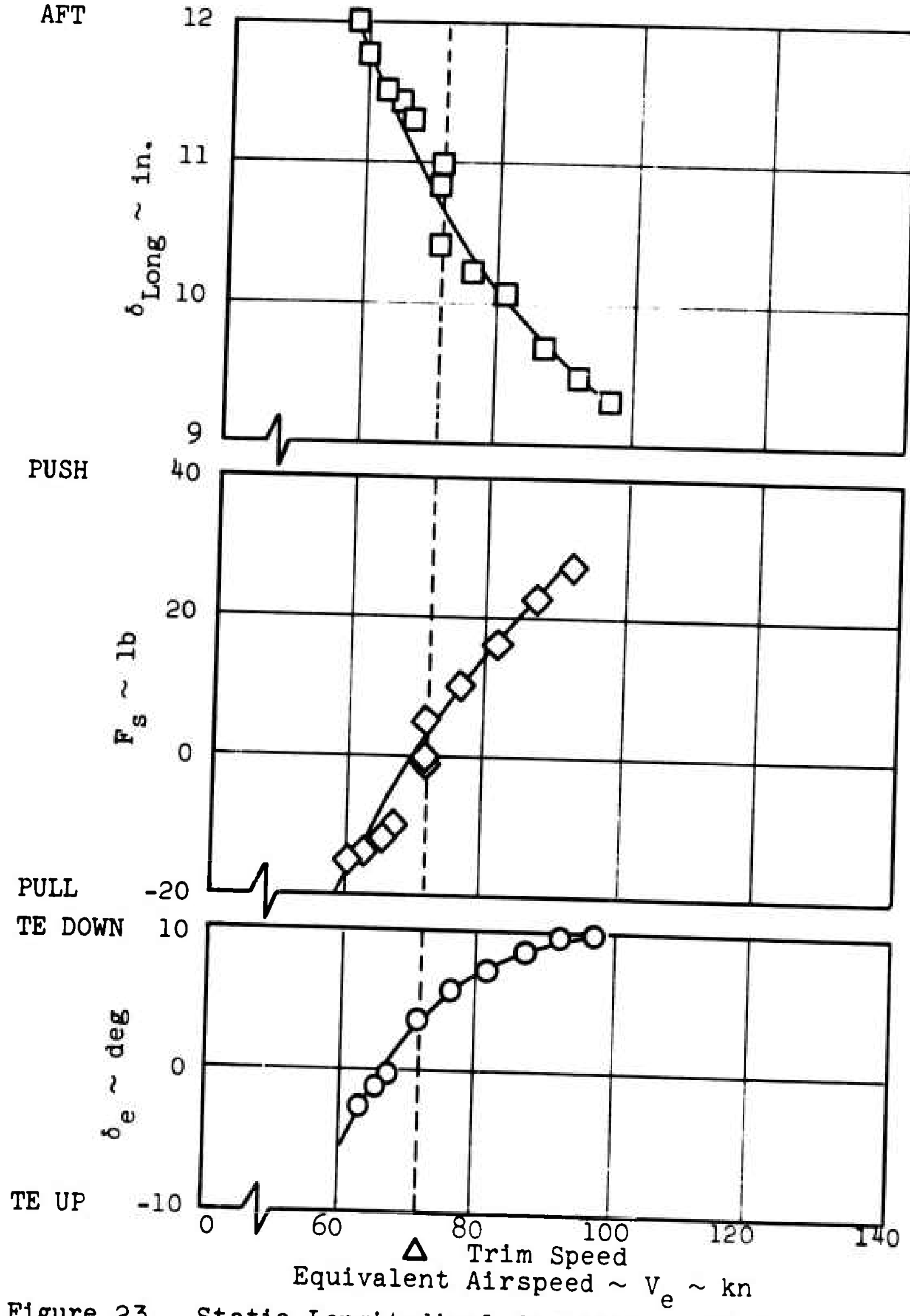

Figure 23. Static Longitudinal Stability, 15° Camber, 71 Knots Trim Speed, BLC ON, FWD CG.

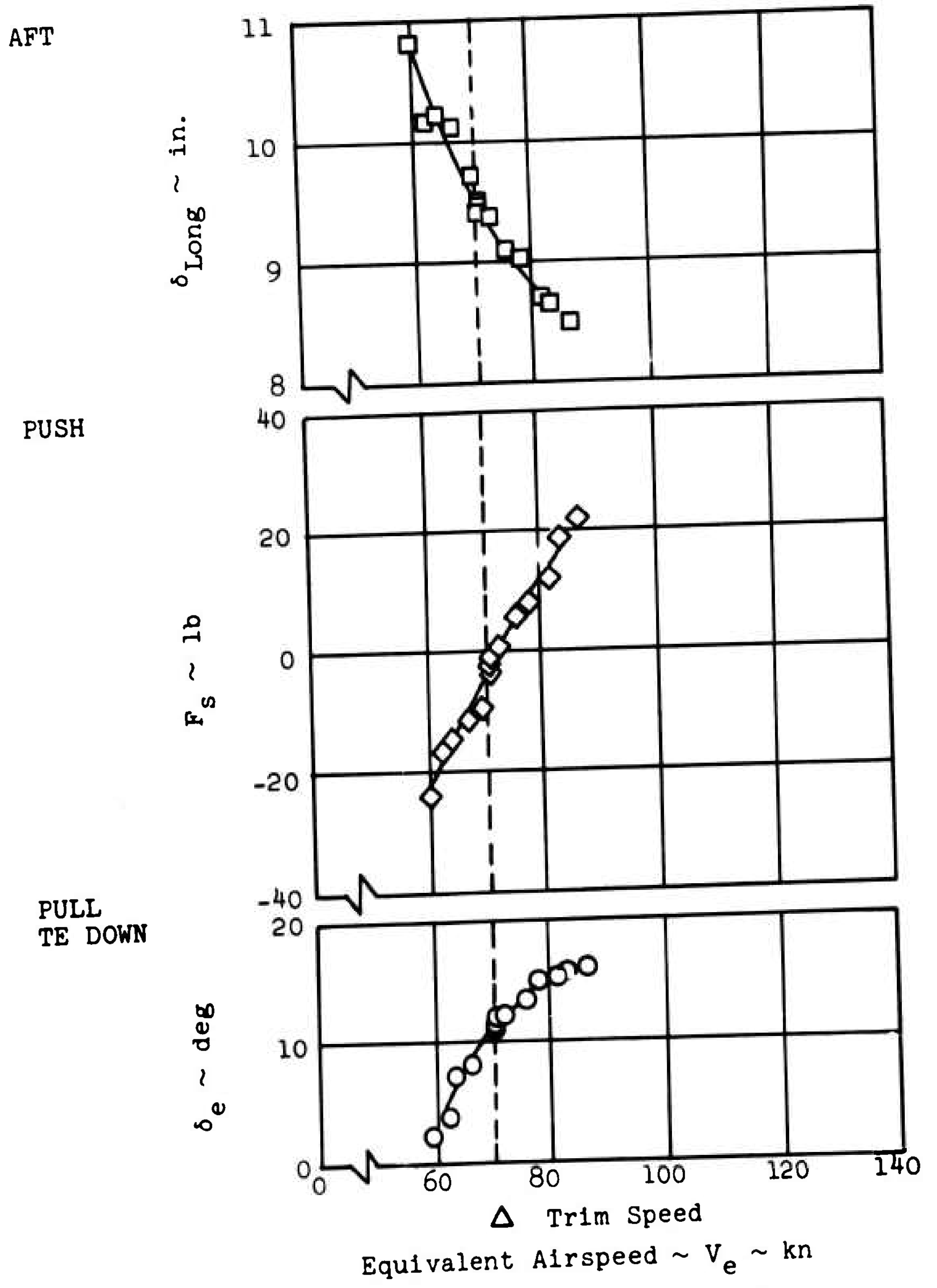

Figure 24. Static Longitudinal Stability, 30° Camber, 71 Knots Trim Speed, BLC ON, FWD CG.

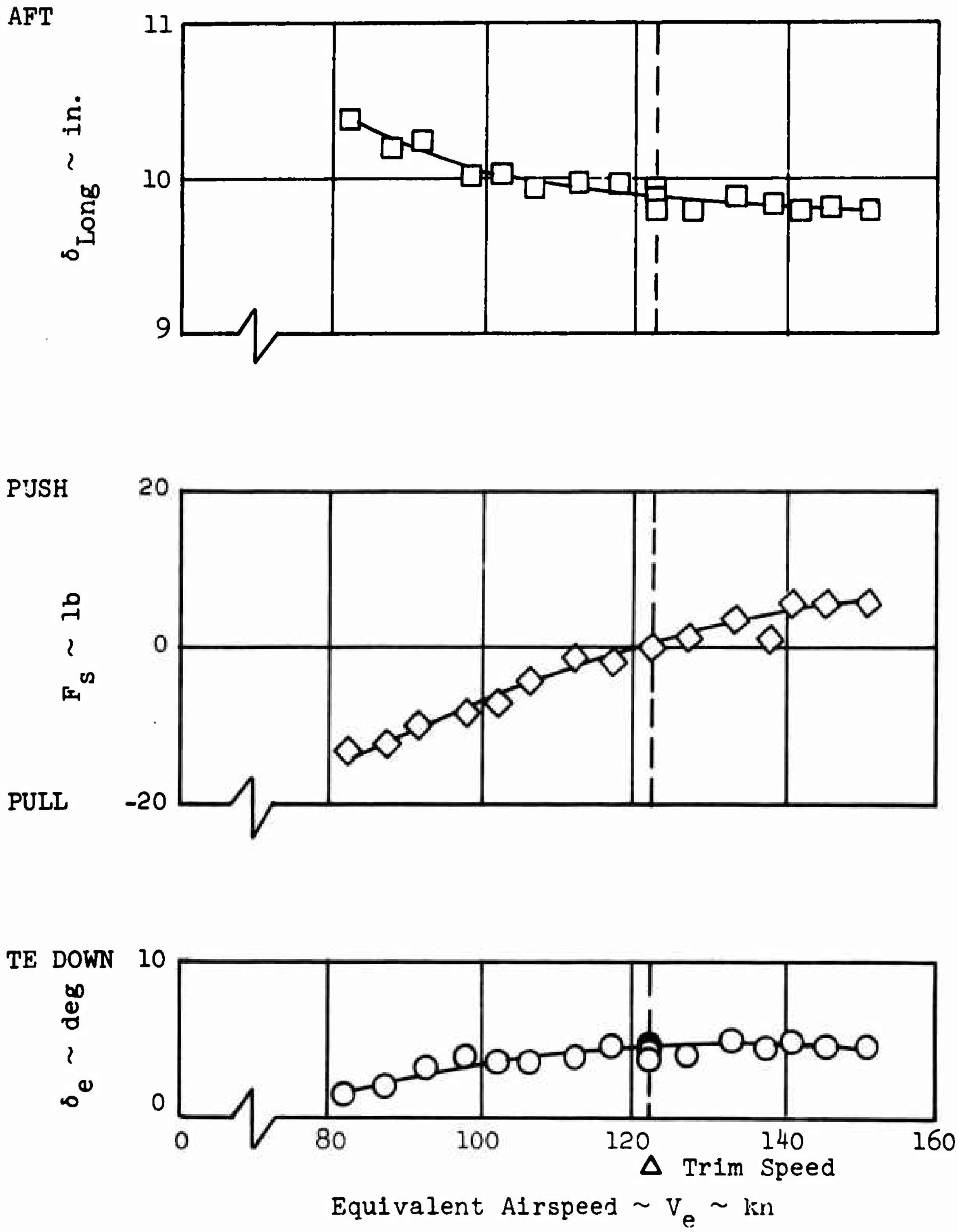

Figure 25. Static Longitudinal Stability, 0° Camber, 122 Knots Trim Speed, BLC ON, AFT CG.

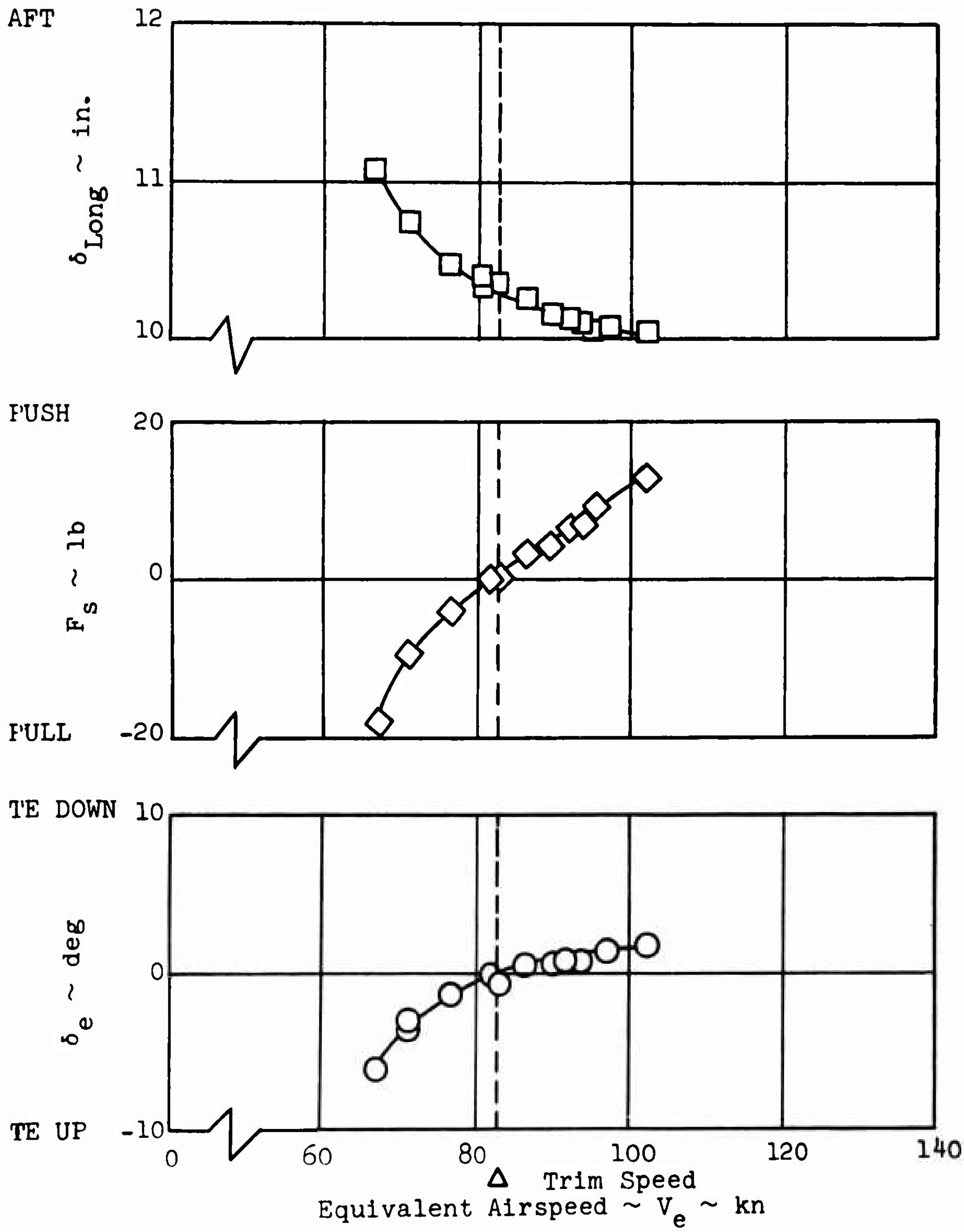

Figure 26. Static Longitudinal Stability, 0° Camber, 83 Knots Trim Speed, BLC OFF/SEALED, AFT CG.

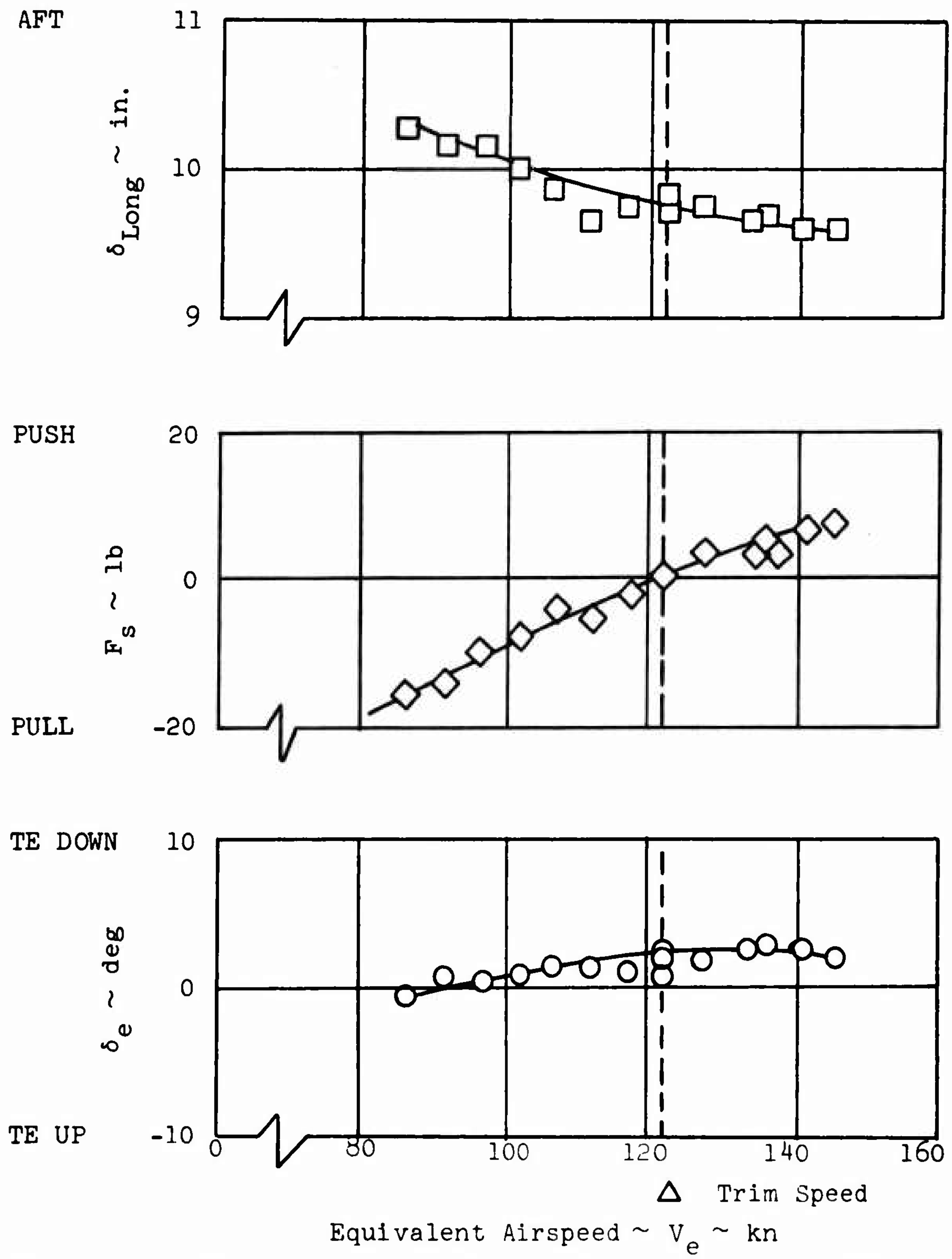

Figure 27. Static Longitudinal Stability, 0° Camber, 122 Knots Trim Speed, BLC OFF/SEALED, AFT CG.

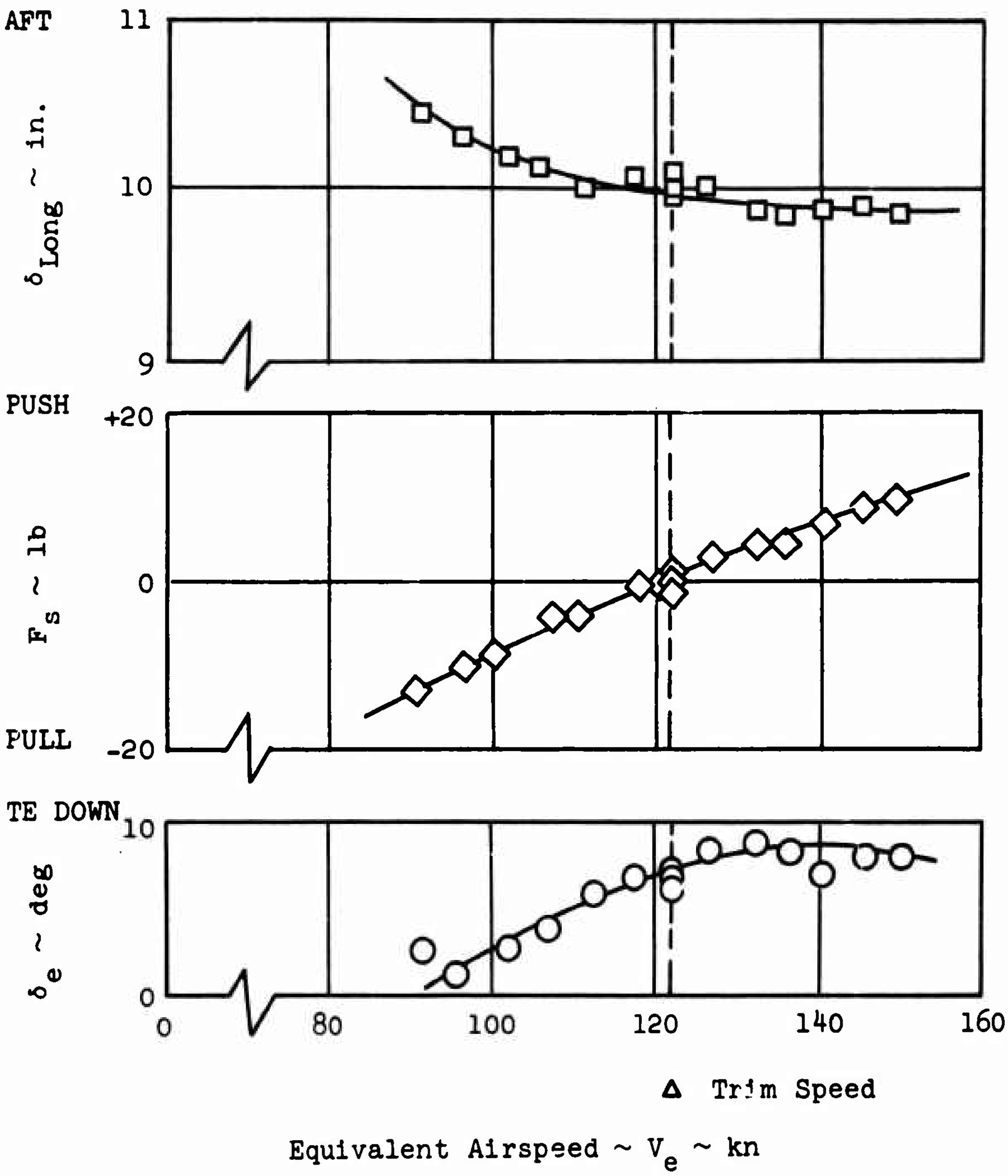

Figure 28. Static Longitudinal Stability, 0° Camber, 122 Knots Trim Speed, BLC OFF/OPEN, AFT CG.

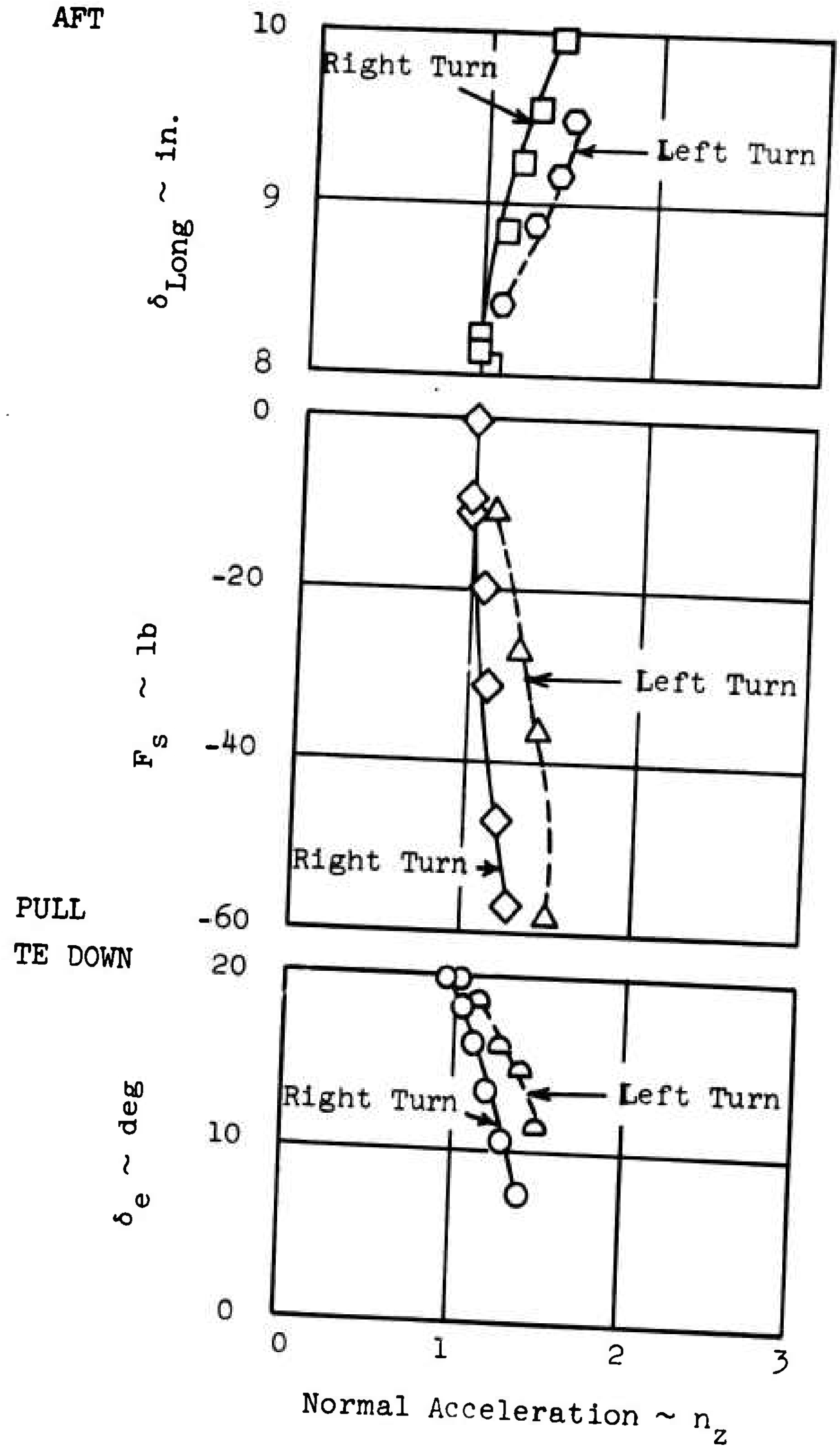

Figure 29. Maneuvering Flight Characteristics, 30° Camber, PLF at 71 Knots, BLC ON, AFT CG, 3000 Feet Altitude.

NOTE: RIGHT- AND LEFT-TURN TEST DATA

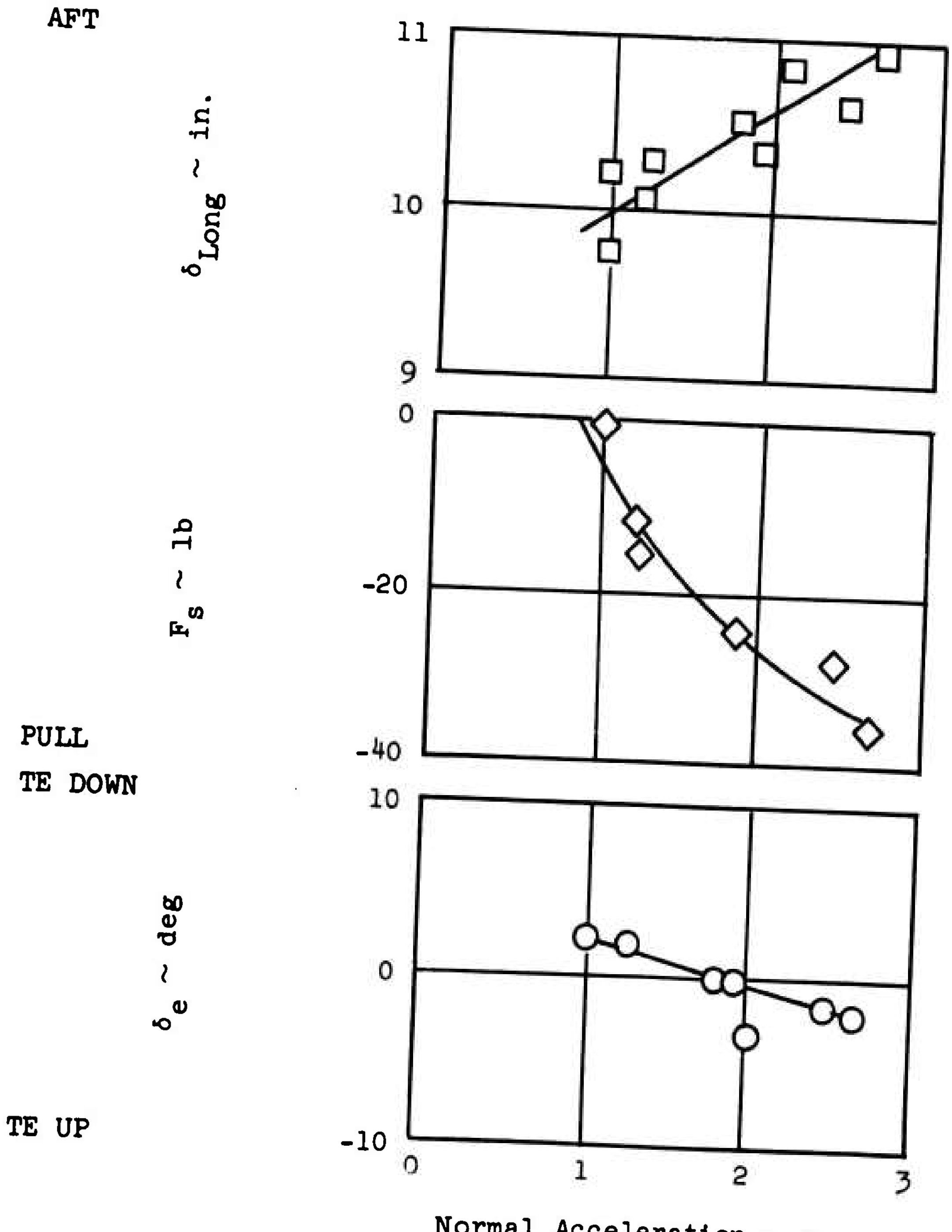

Figure 30. Maneuvering Flight Characteristics, 0° Camber, PLF at 122 Knots, BLC ON, AFT CG, 3000 Feet Altitude.

NOTE: Right- and Left-Turn Test Data

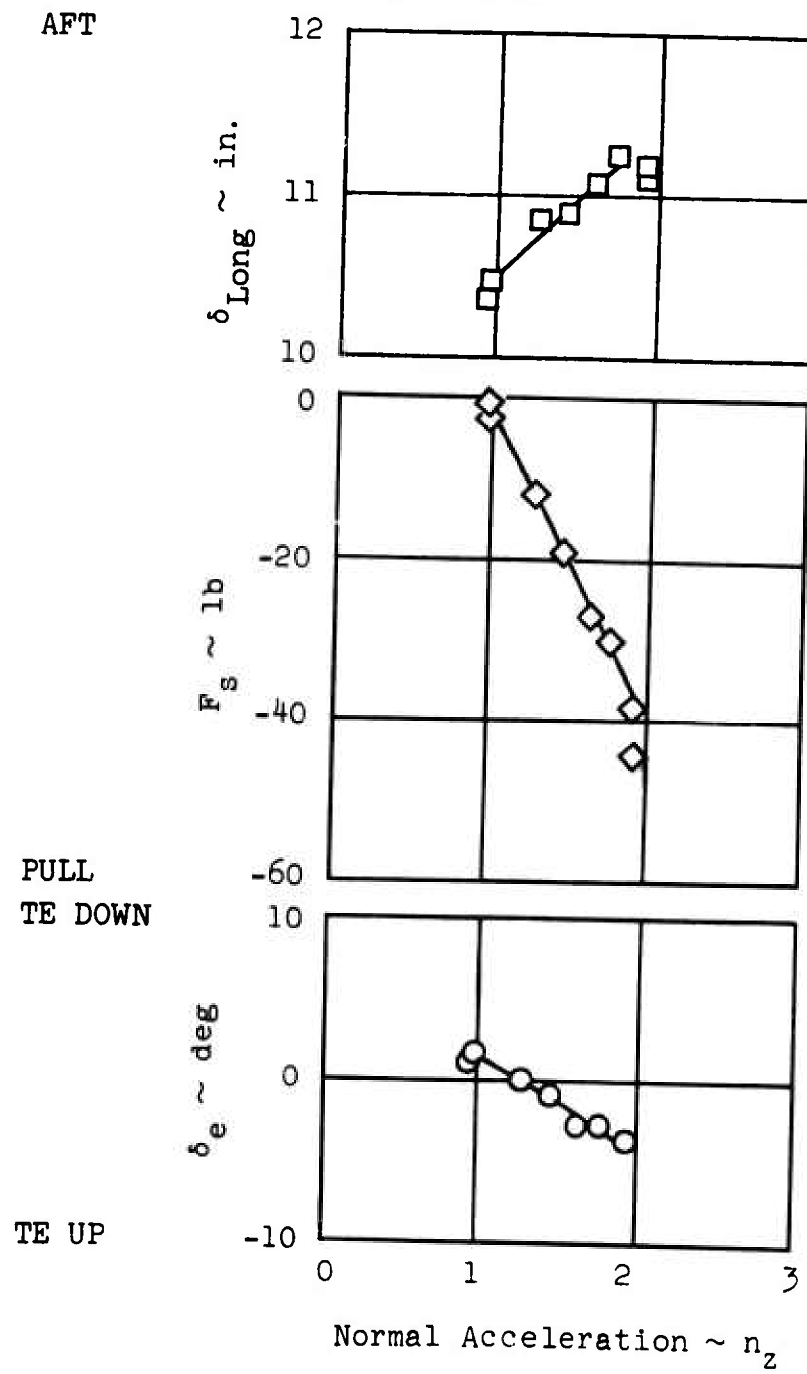

Figure 31. Maneuvering Flight Characteristics, 0° Camber, PLF at 122 Knots, BLC ON, FWD CG, 3000 Feet Altitude.

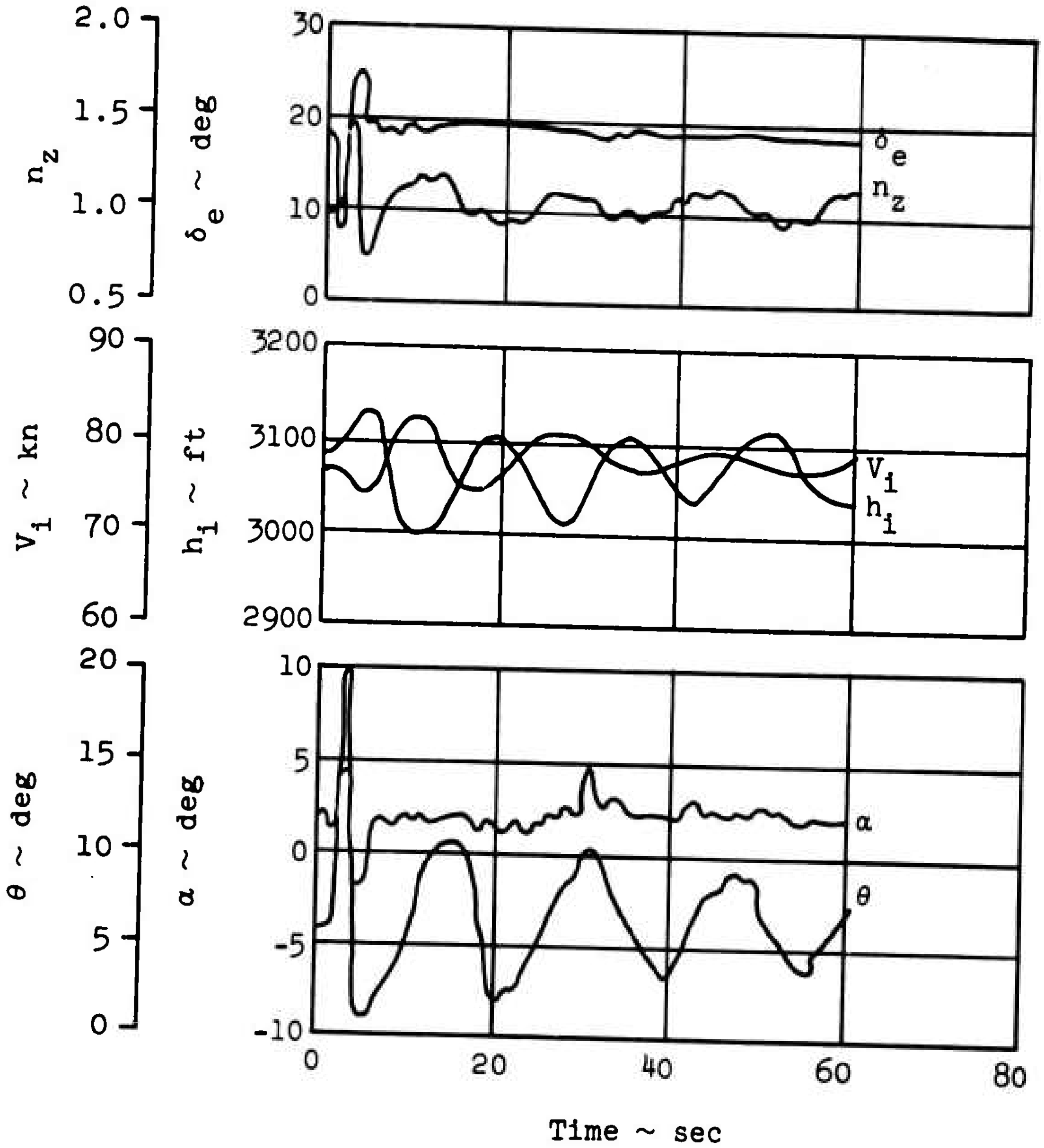

Figure 32. Longitudinal Dynamic Stability, Phugoid, 30° Camber, BLC ON, AFT CG, 71 Knots.

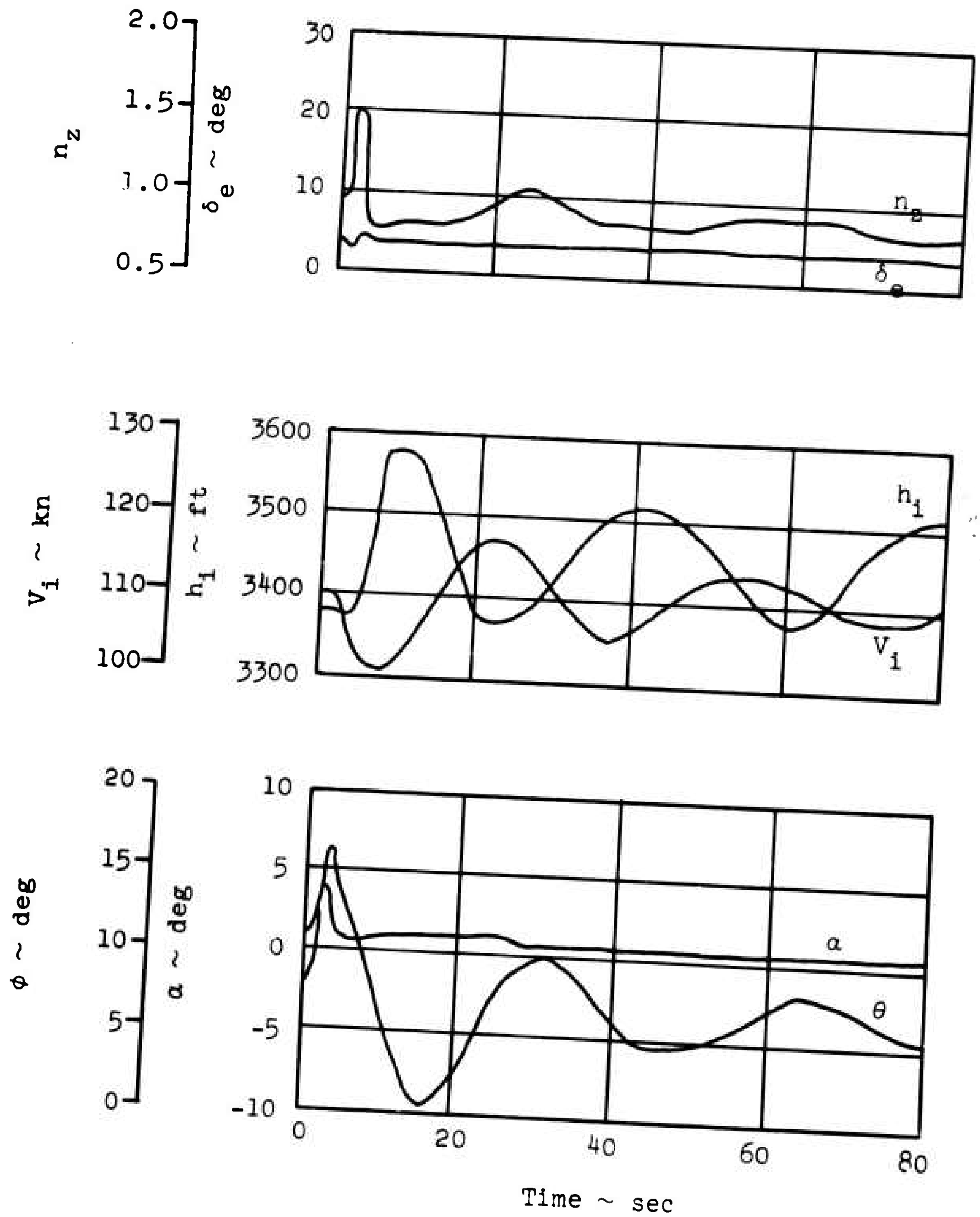

Figure 33. Longitudinal Dynamic Stability, Phugoid, 0° Camber, BLC ON, AFT CG, 122 Knots.

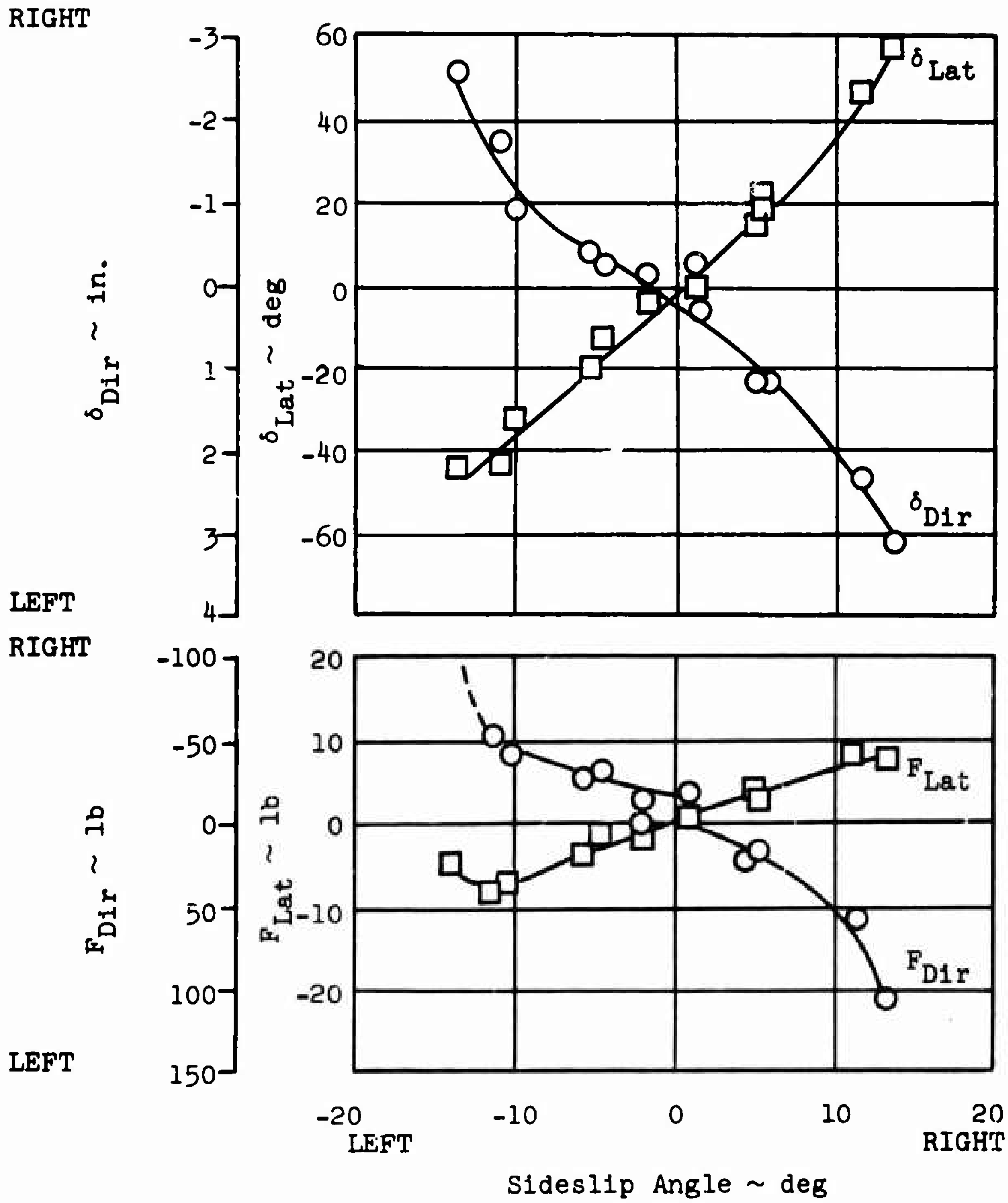

Figure 34. Lateral/Directional Static Stability, 0° Camber BLC ON, 71 Knots, 4000 Feet Altitude.

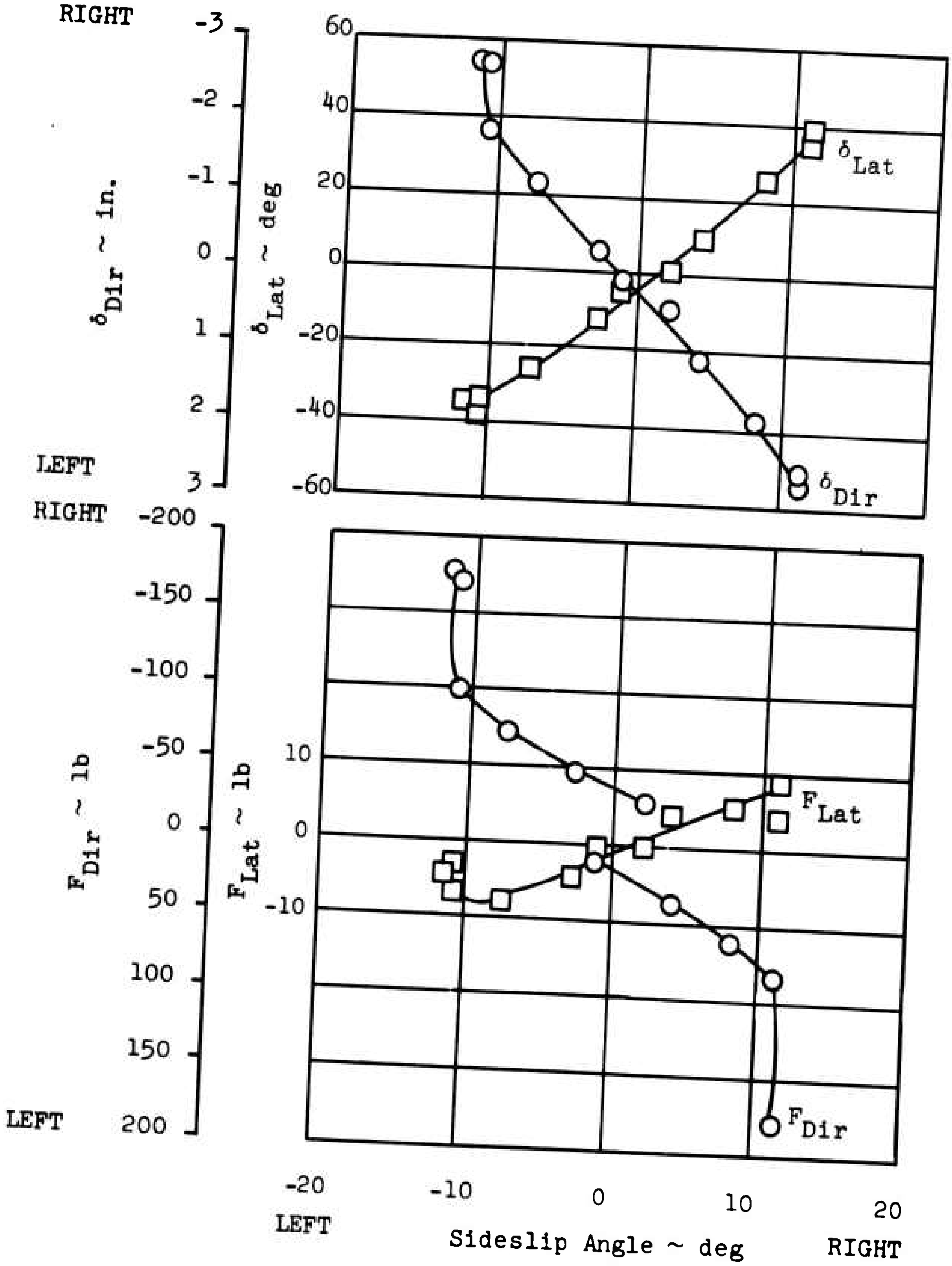

Figure 35. Lateral/Directional Static Stability, 15° Camber, BLC ON, 71 Knots, 4000 Feet Altitude.

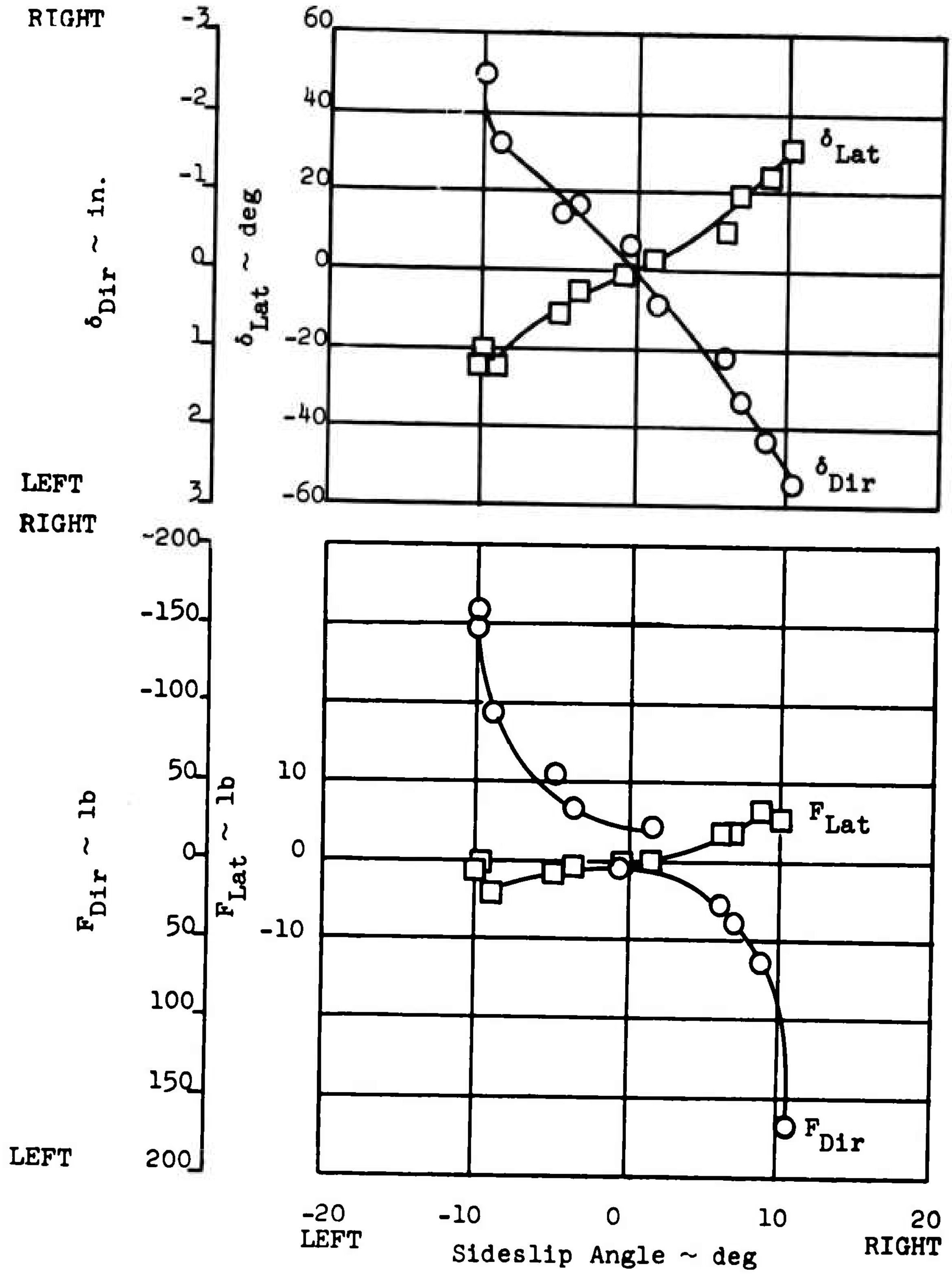

Figure 36. Lateral/Directional Static Stability, 30° Camber, BLC ON, 71 Knots, 4150 Feet Altitude.

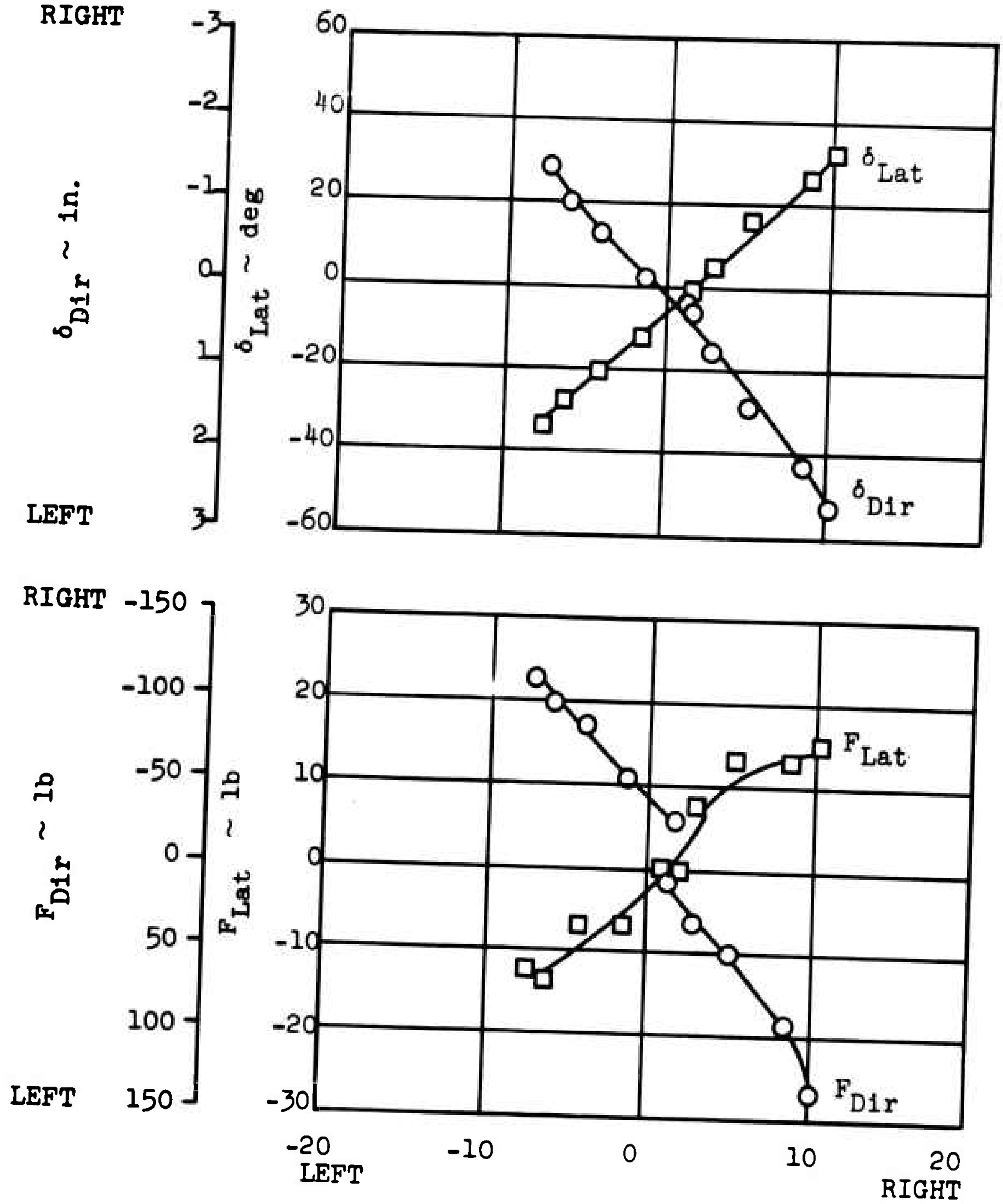

Figure 37. Lateral/Directional Static Stability, 0° Camber, BLC ON, 122 Knots, 4000 Feet Altitude.

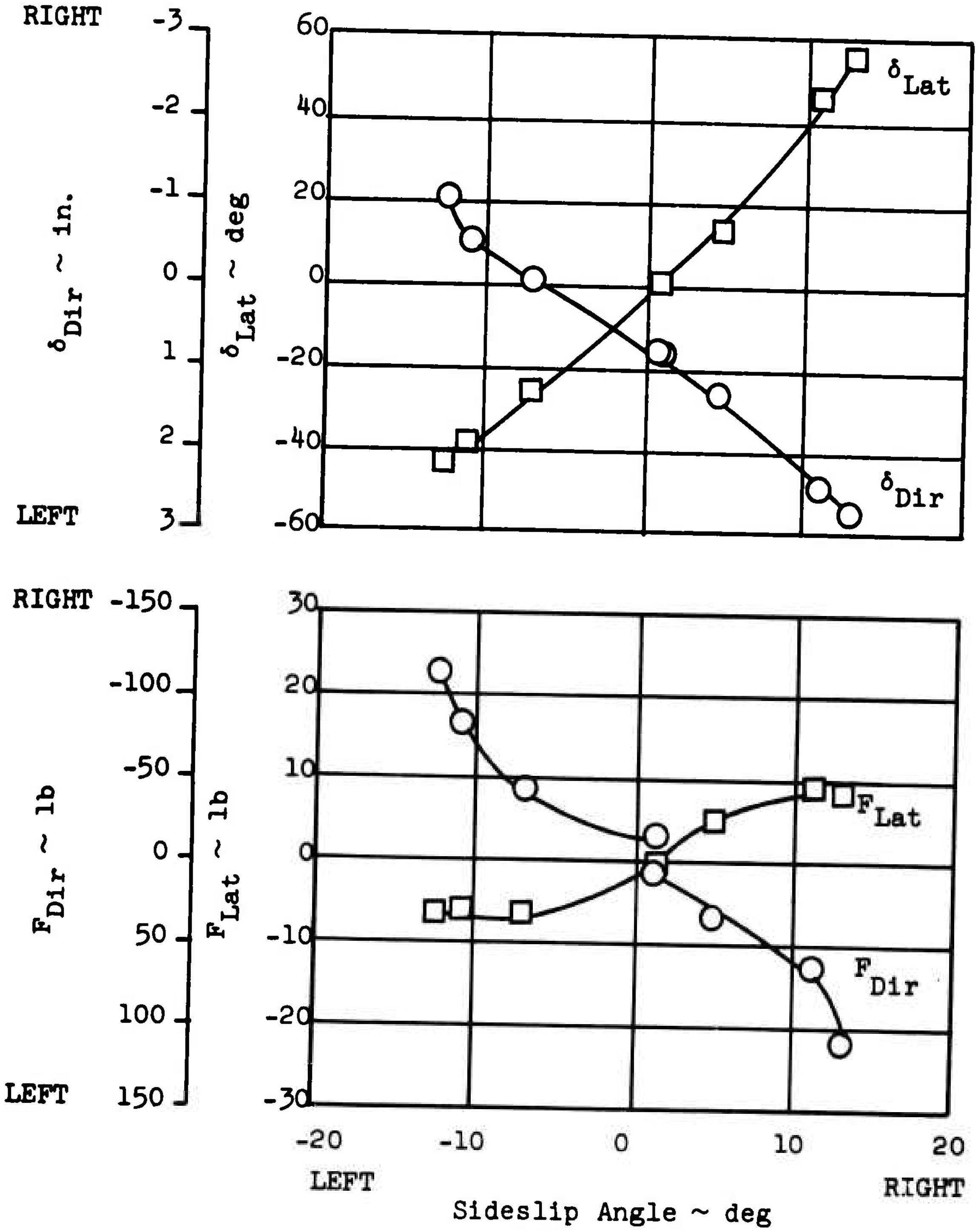

Figure 38. Lateral/Directional Static Stability, 0° Camber, BLC OFF/SEALED, 83 Knots, 3000 Feet Altitude.

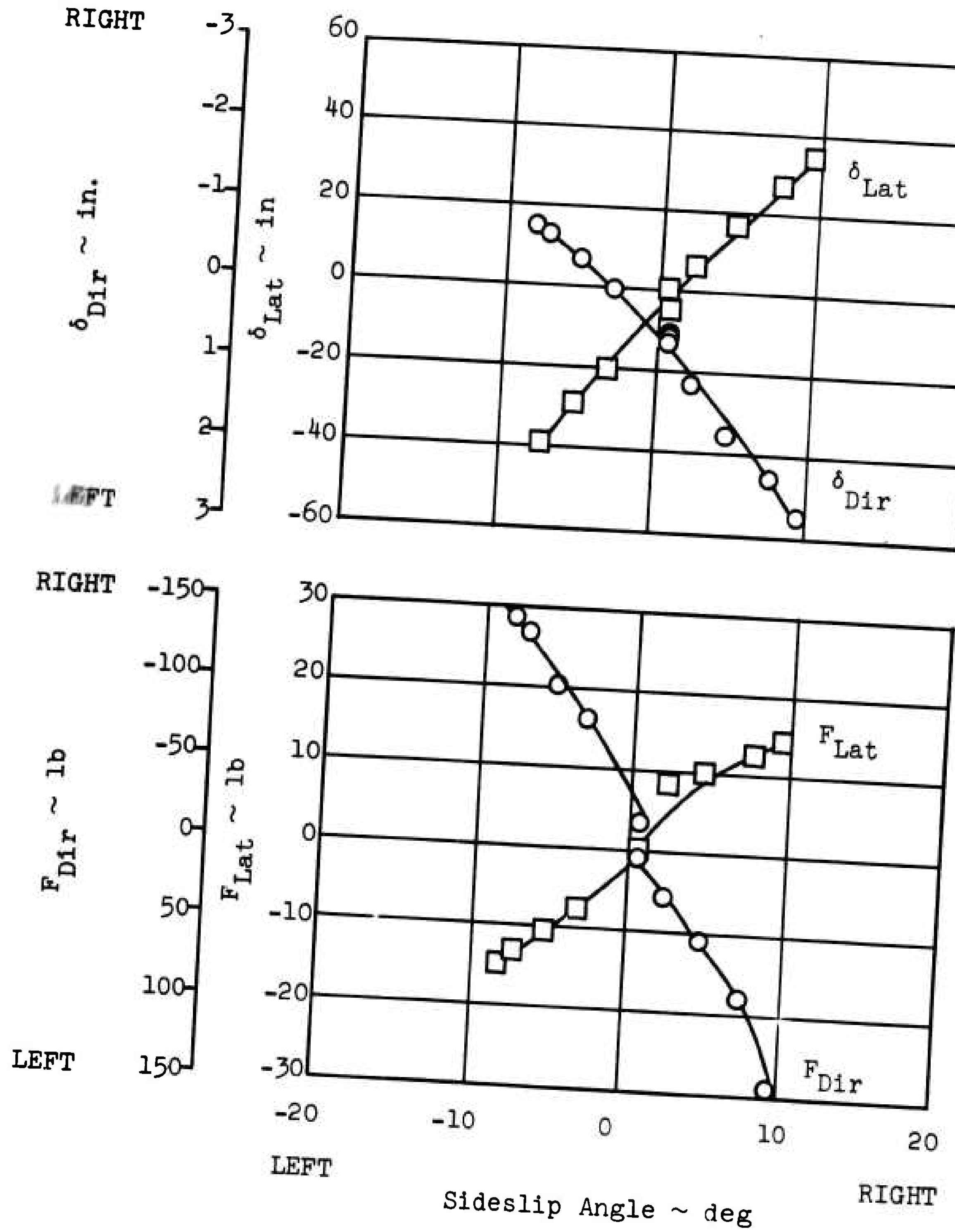

Figure 39. Lateral/Directional Static Stability, 0° Camber BLC OFF/OPEN, 122 Knots, 2800 Feet Altitude.

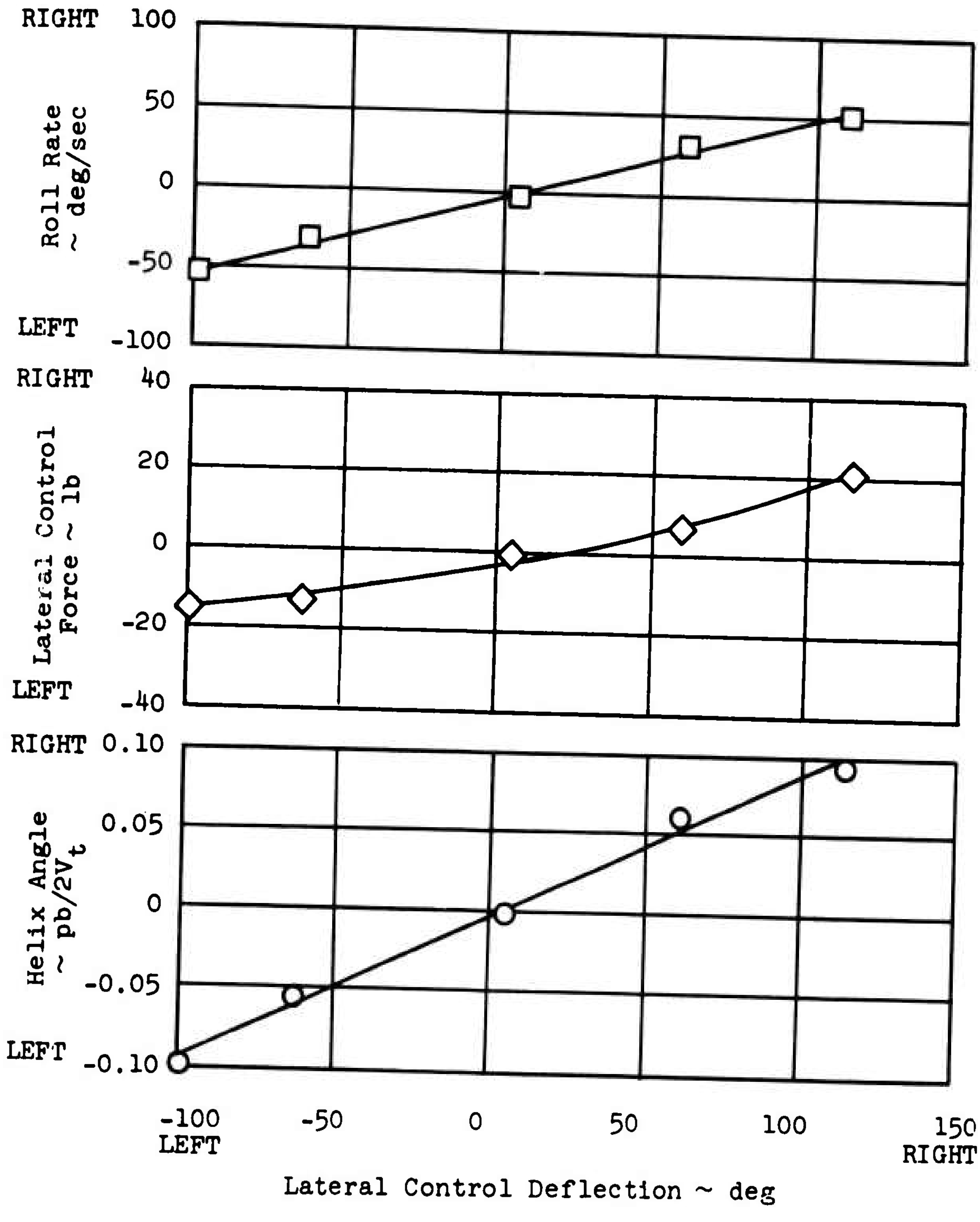

Figure 40. Lateral Control Power, 0° Camber, 71 Knots, BLC ON, 2900 Feet Altitude.

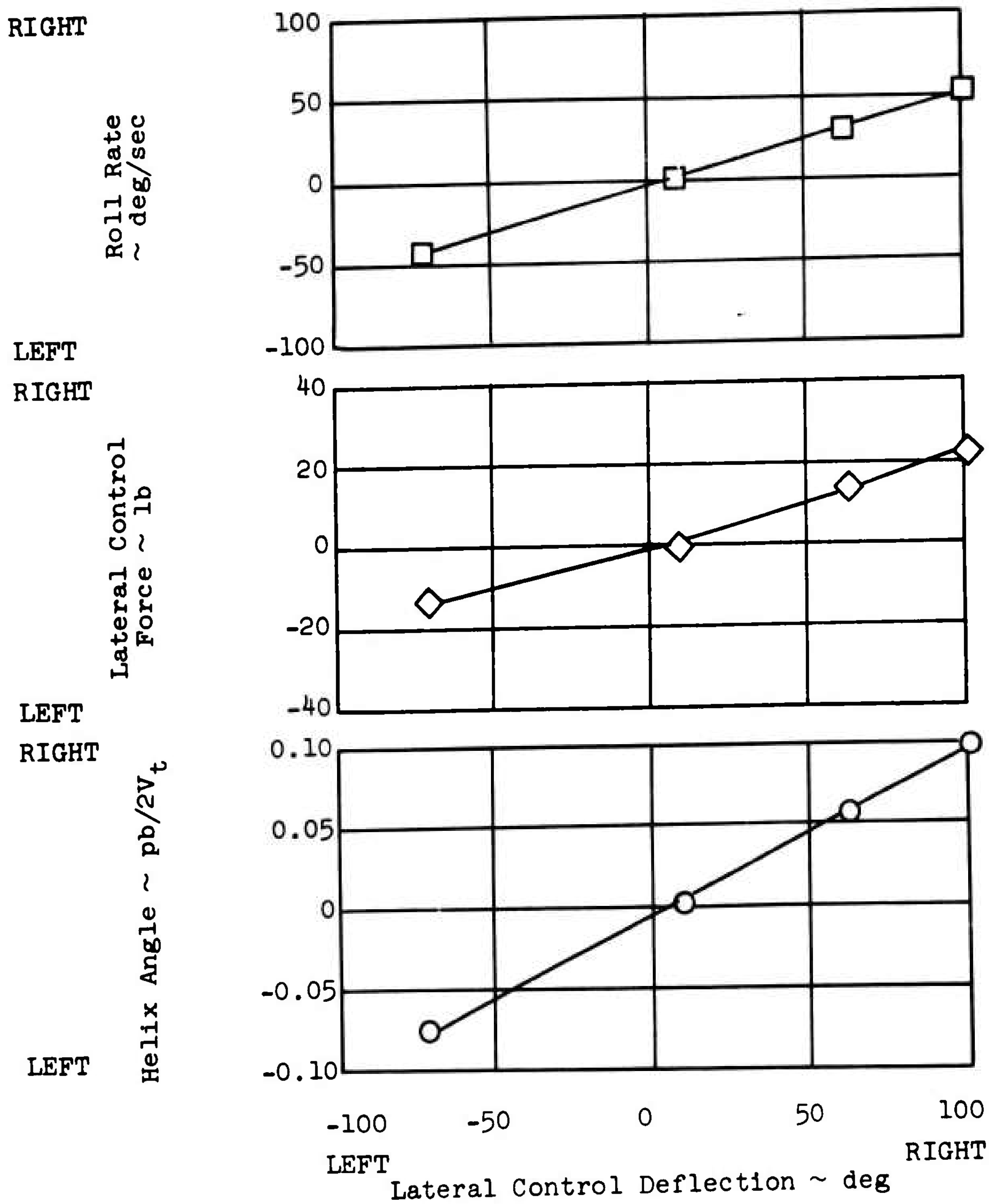

Figure 41. Lateral Control Power, 30° Camber, 71 Knots, BLC ON, 3150 Feet Altitude.

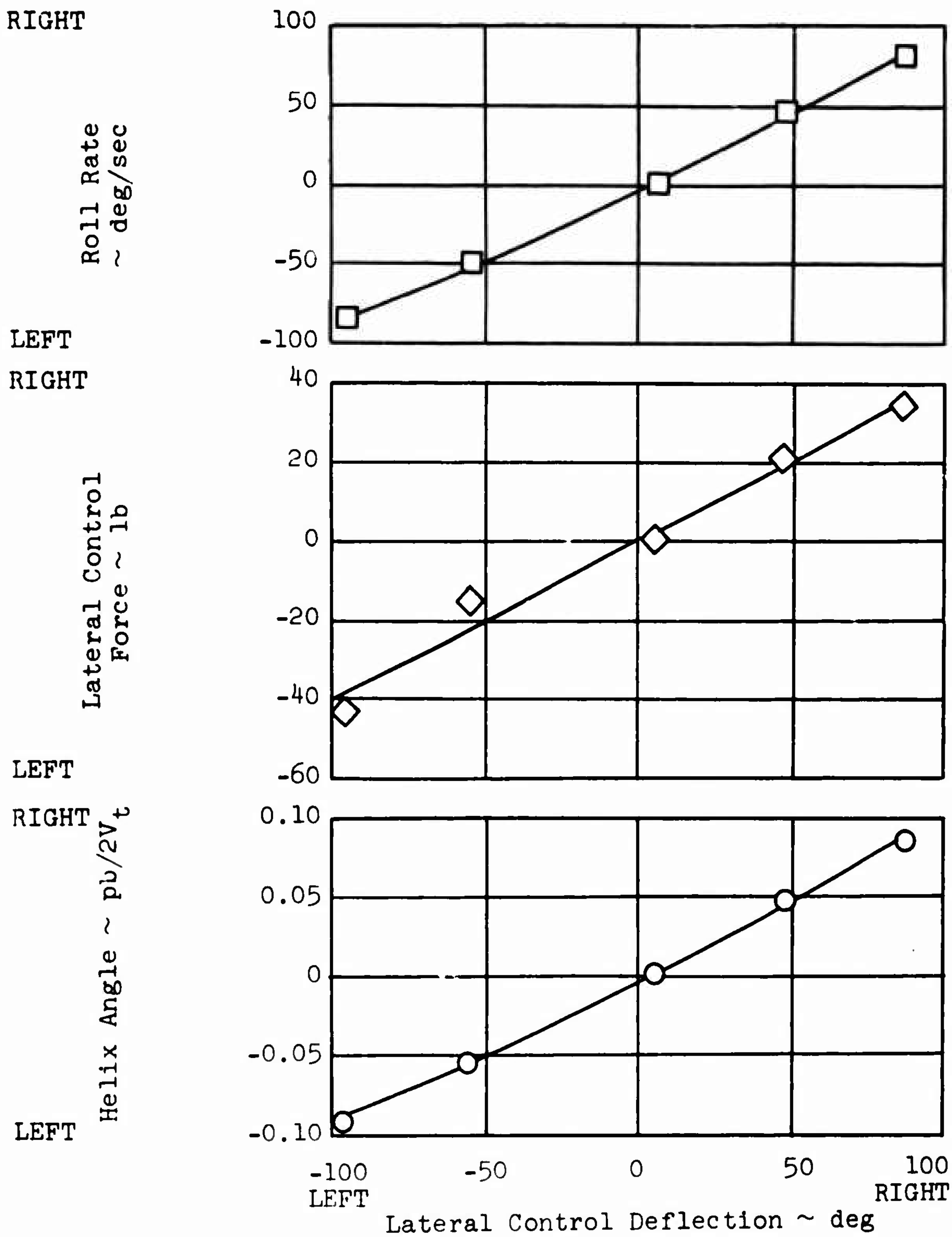

Figure 42. Lateral Control Power, 0° Camber, 122 Knots, BLC ON, 2900 Feet Altitude.

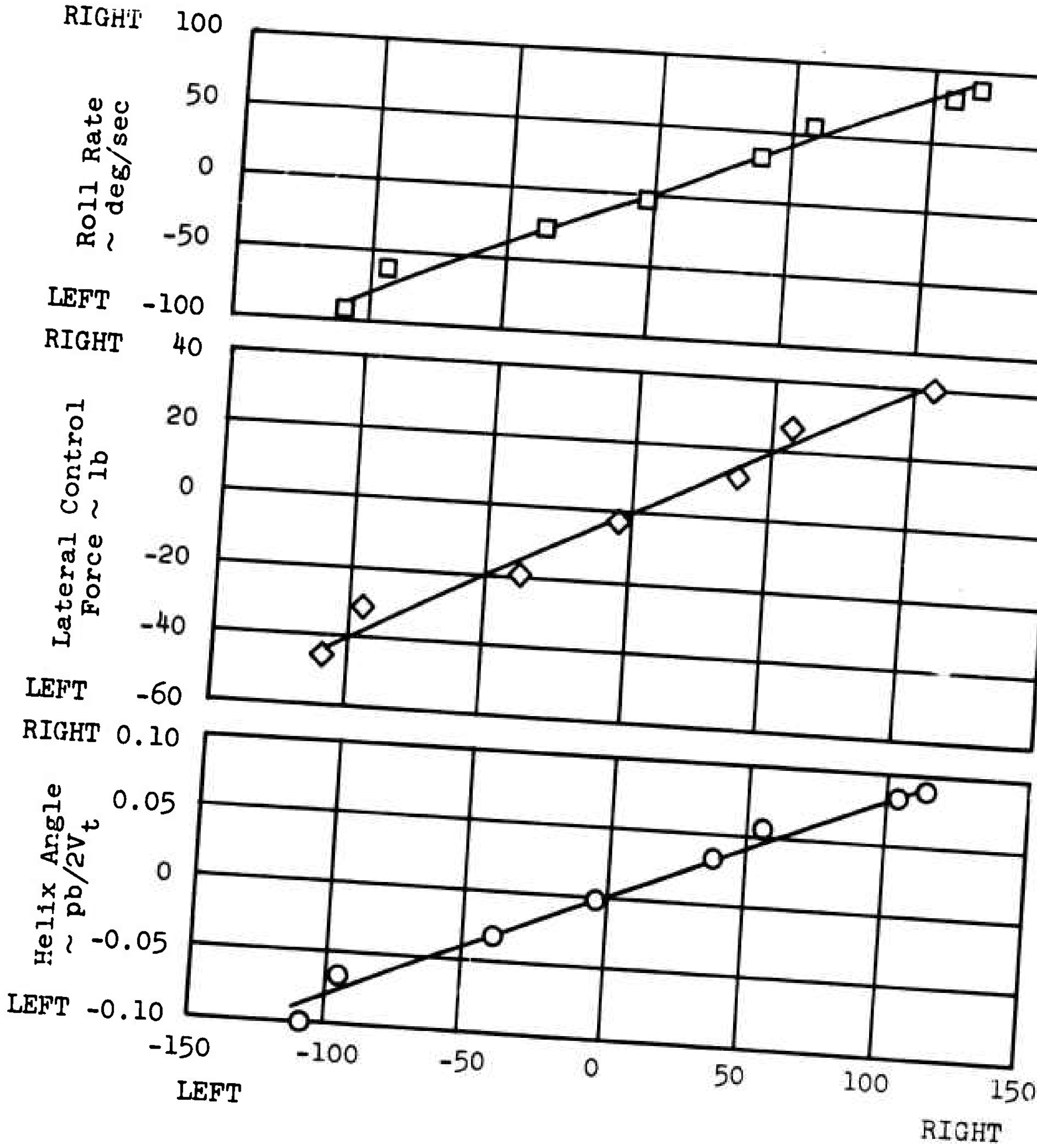

Figure 43. Lateral Control Power, 0° Camber, 122 Knots, BLC OFF/OPEN, 2900 Feet Altitude.

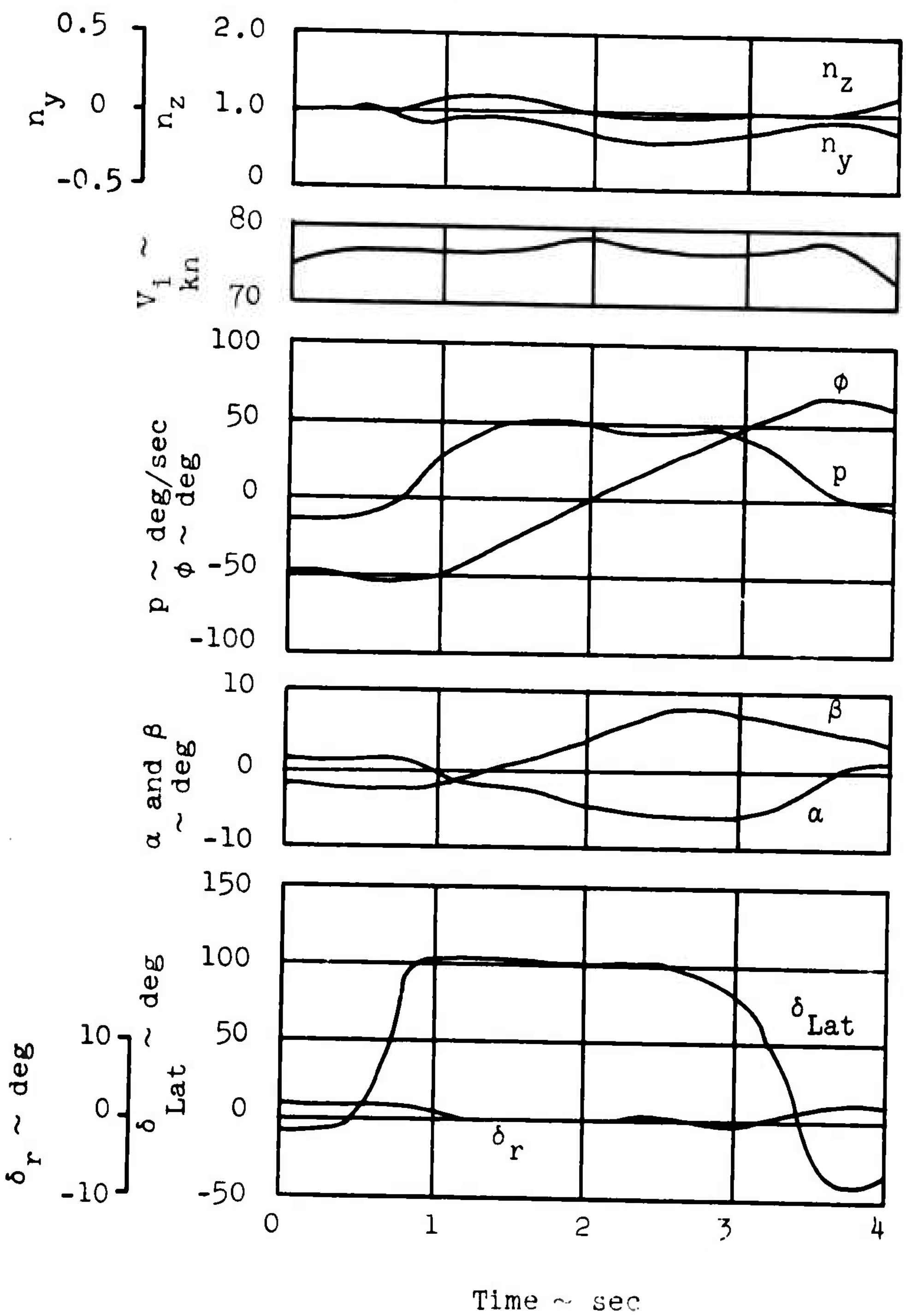

Figure 44. Lateral Control Time History, 30° Camber, BLC ON, 71 Knots.

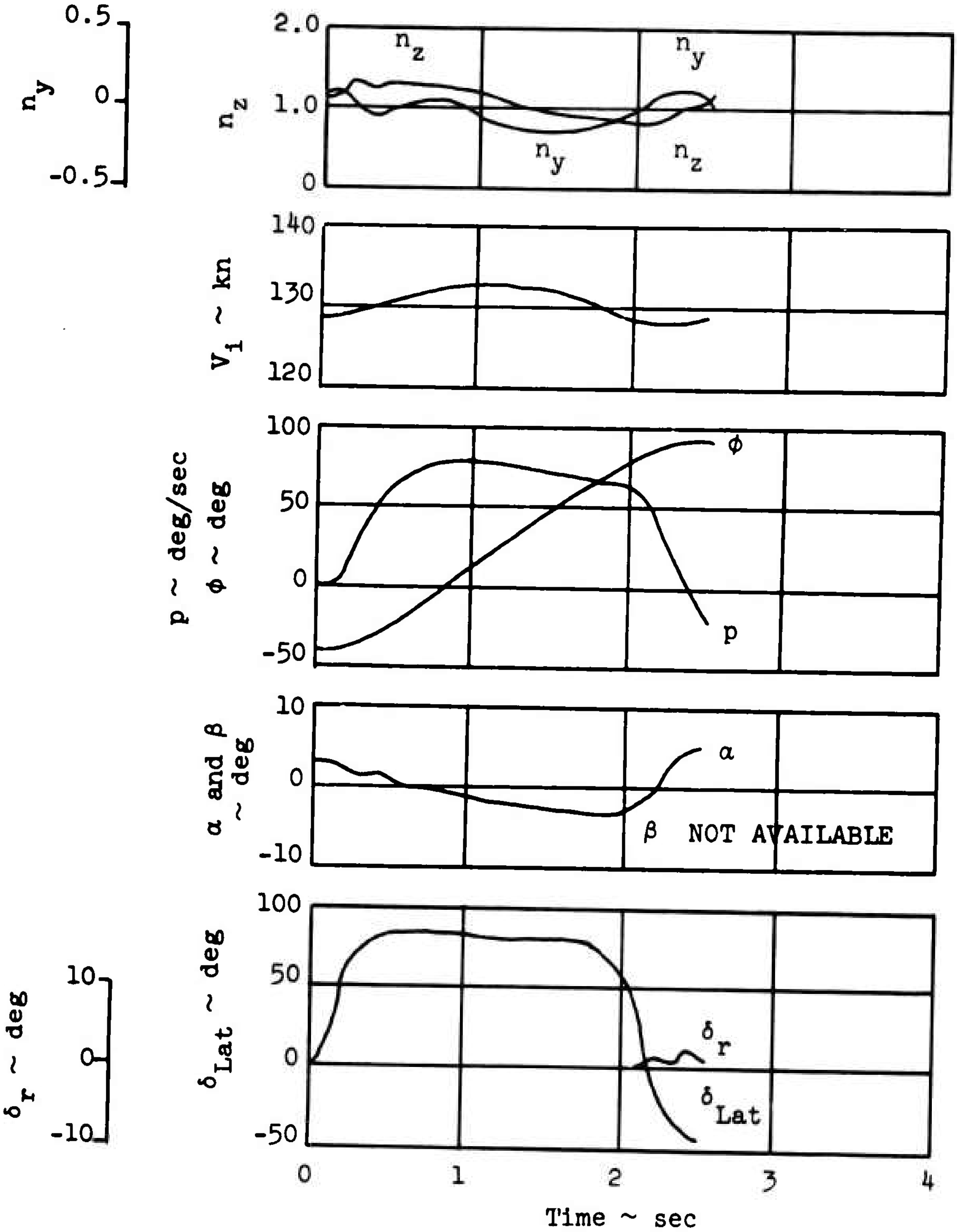

Figure 45. Lateral Control Time History, 0° Camber, BLC ON, 122 knots.

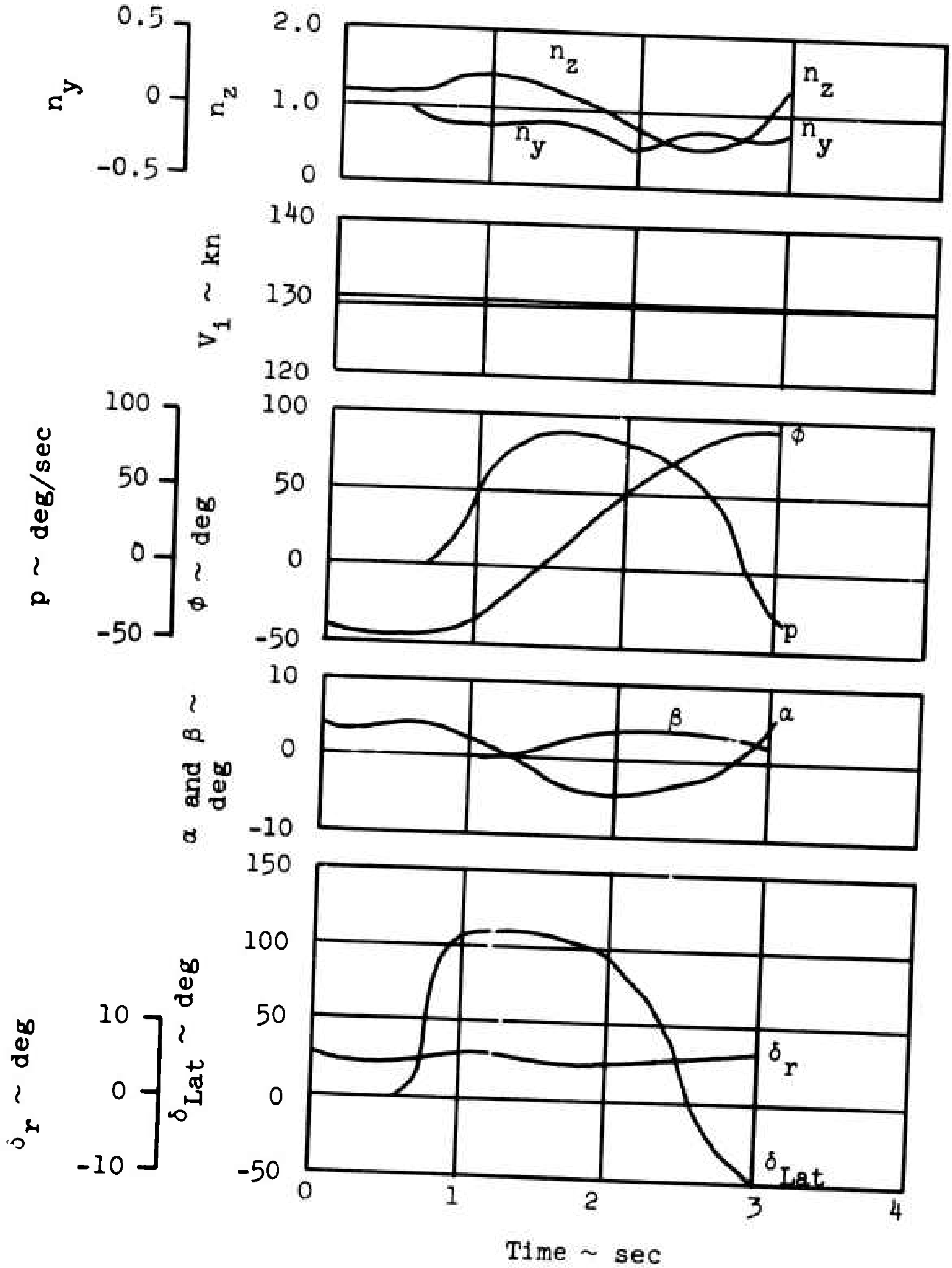

Figure 46. Lateral Control Time History, 0° Camber, BLC OFF/OPEN, 122 Knots.

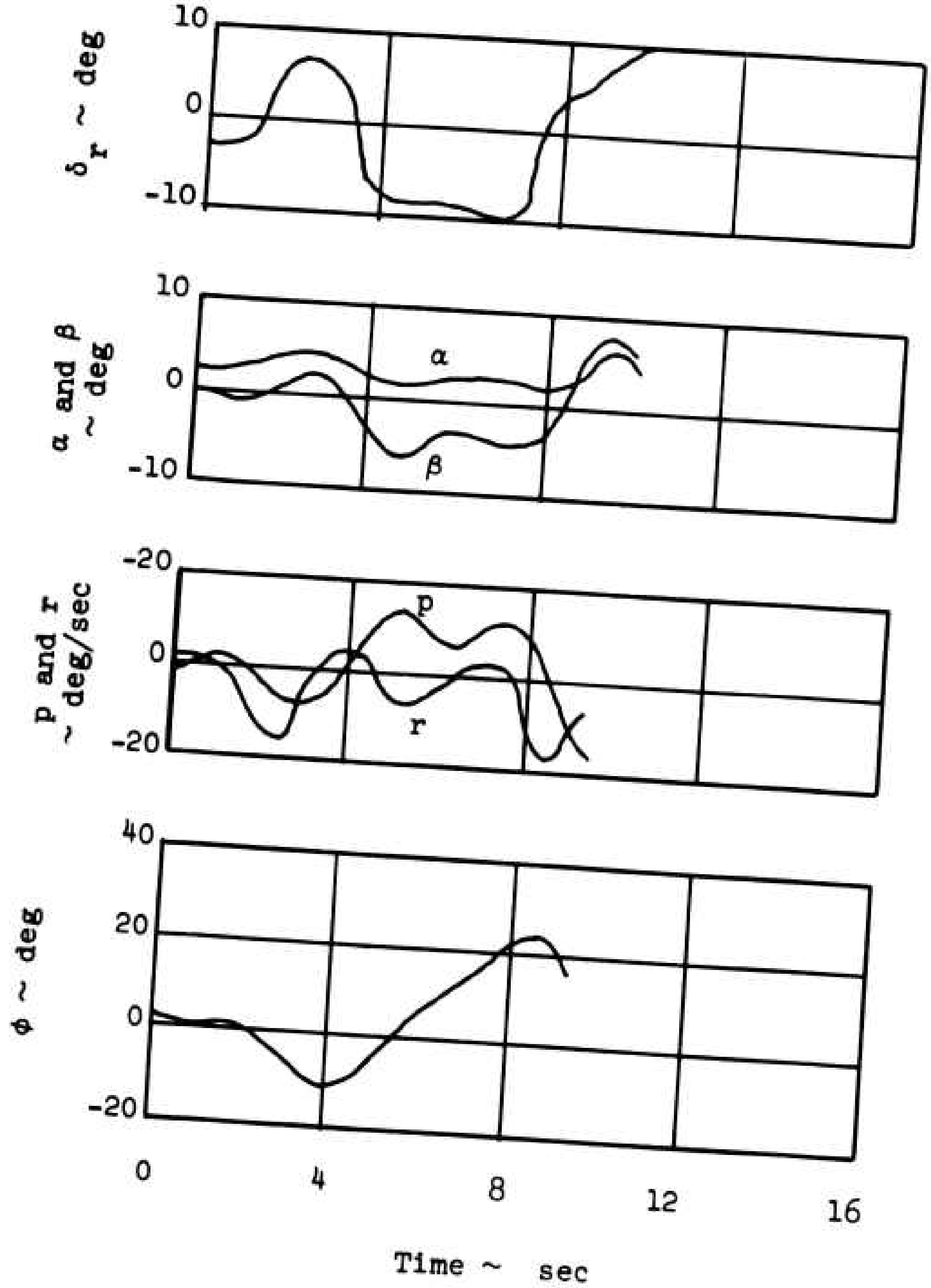

Figure 47. Lateral/Directional Stability, Rudder Induced Roll, 30° Camber, BLC ON, 71 Knots.

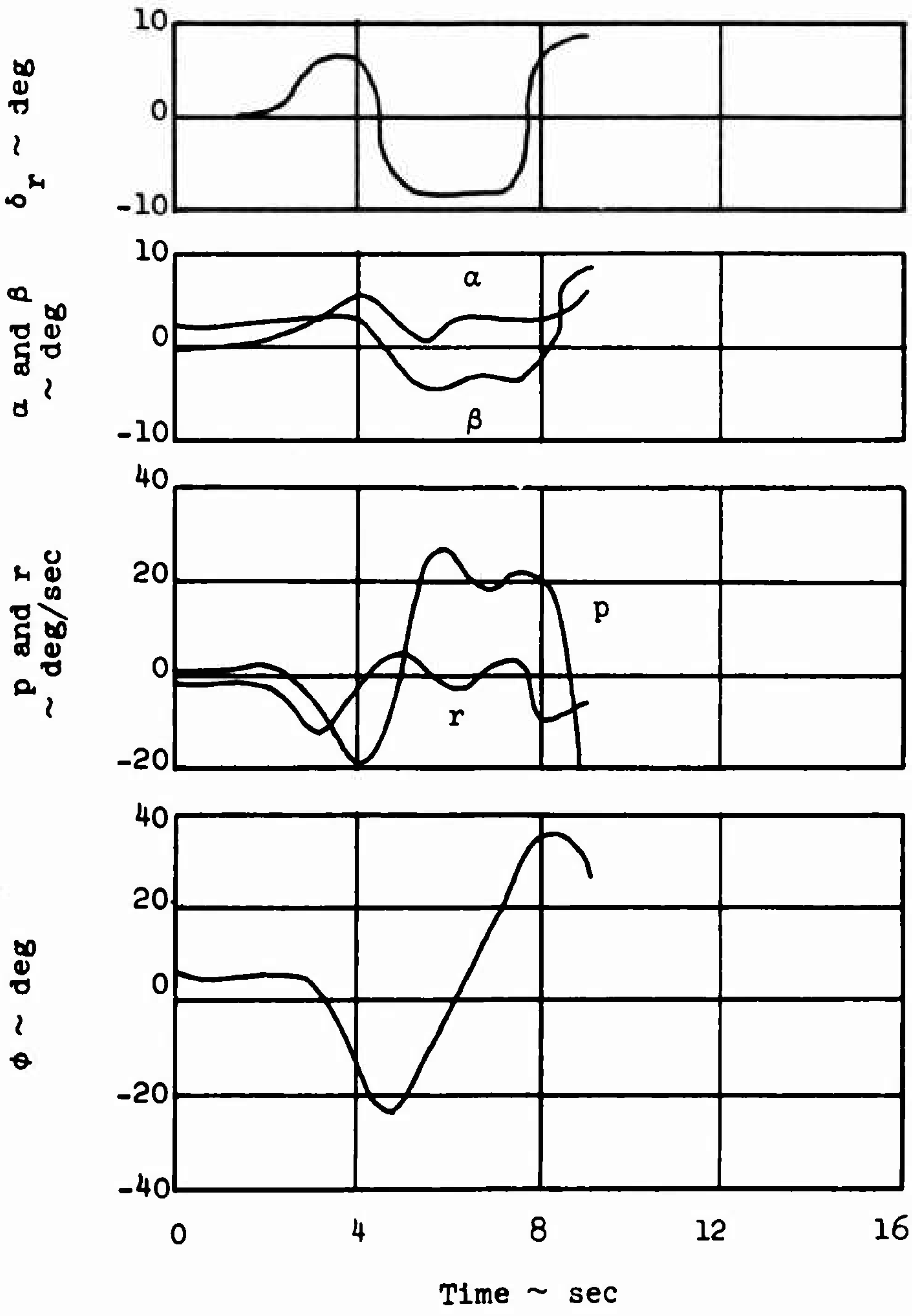

Figure 48. Lateral/Directional Stability, Rudder Induced Roll, 0° Camber, BLC ON, 122 Knots.

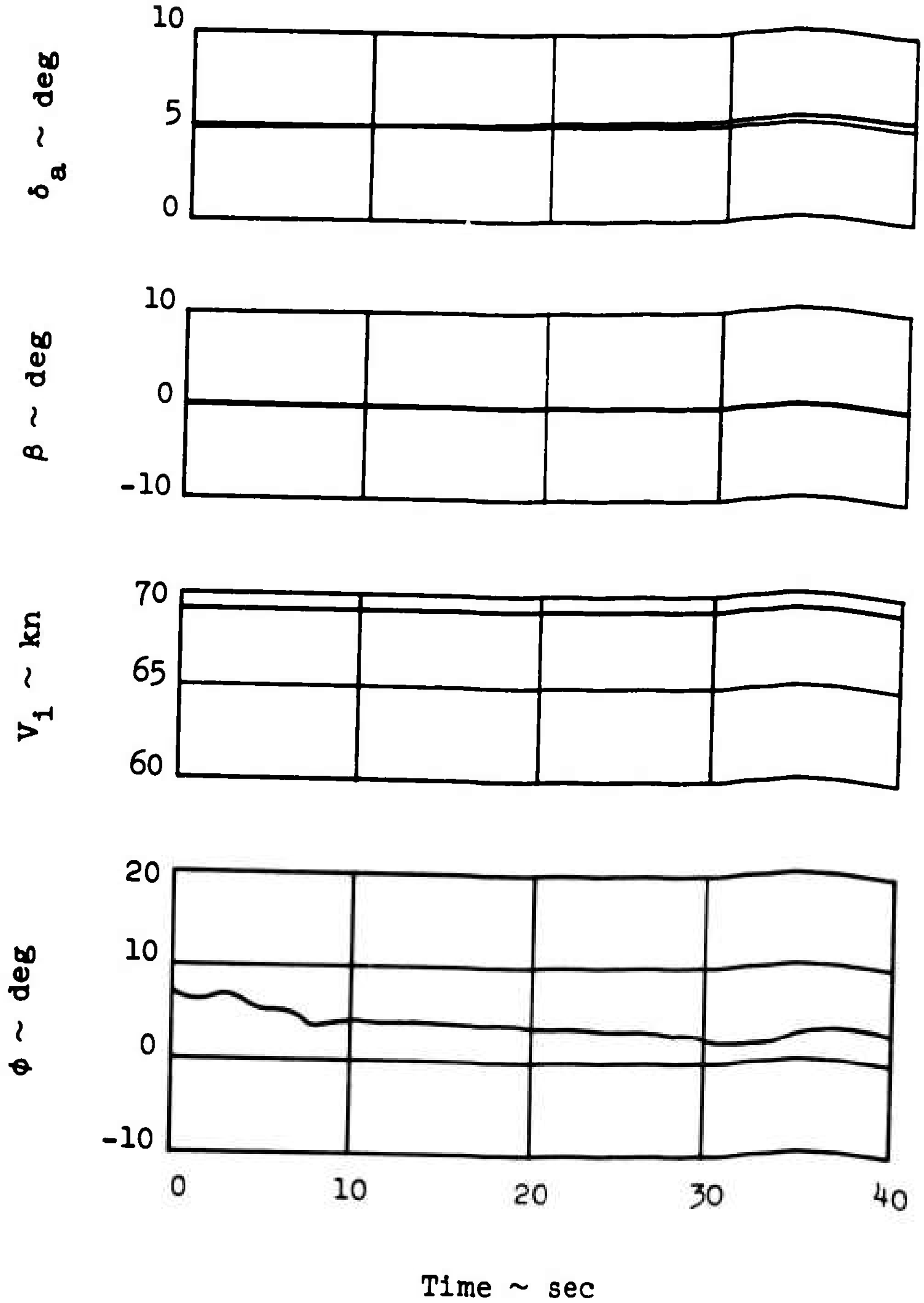

Figure 49. Lateral/Directional Stability, Spiral Stability, 30° Camber, BLC ON, 71 Knots, Right Turn.

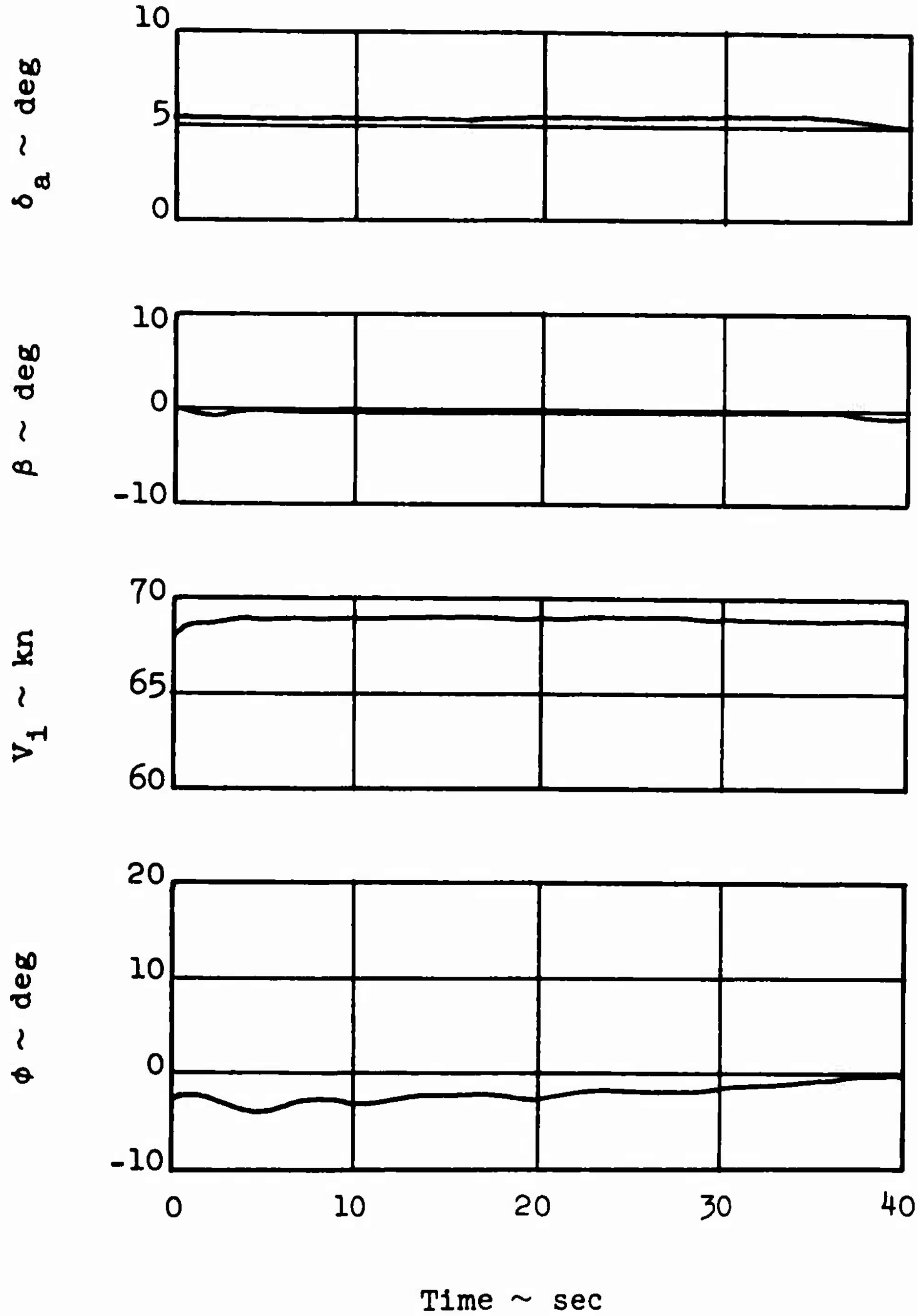

Figure 50. Lateral/Directional Stability, Spiral Stability, 30° Camber, BLC ON, 71 Knots, Left Turn.

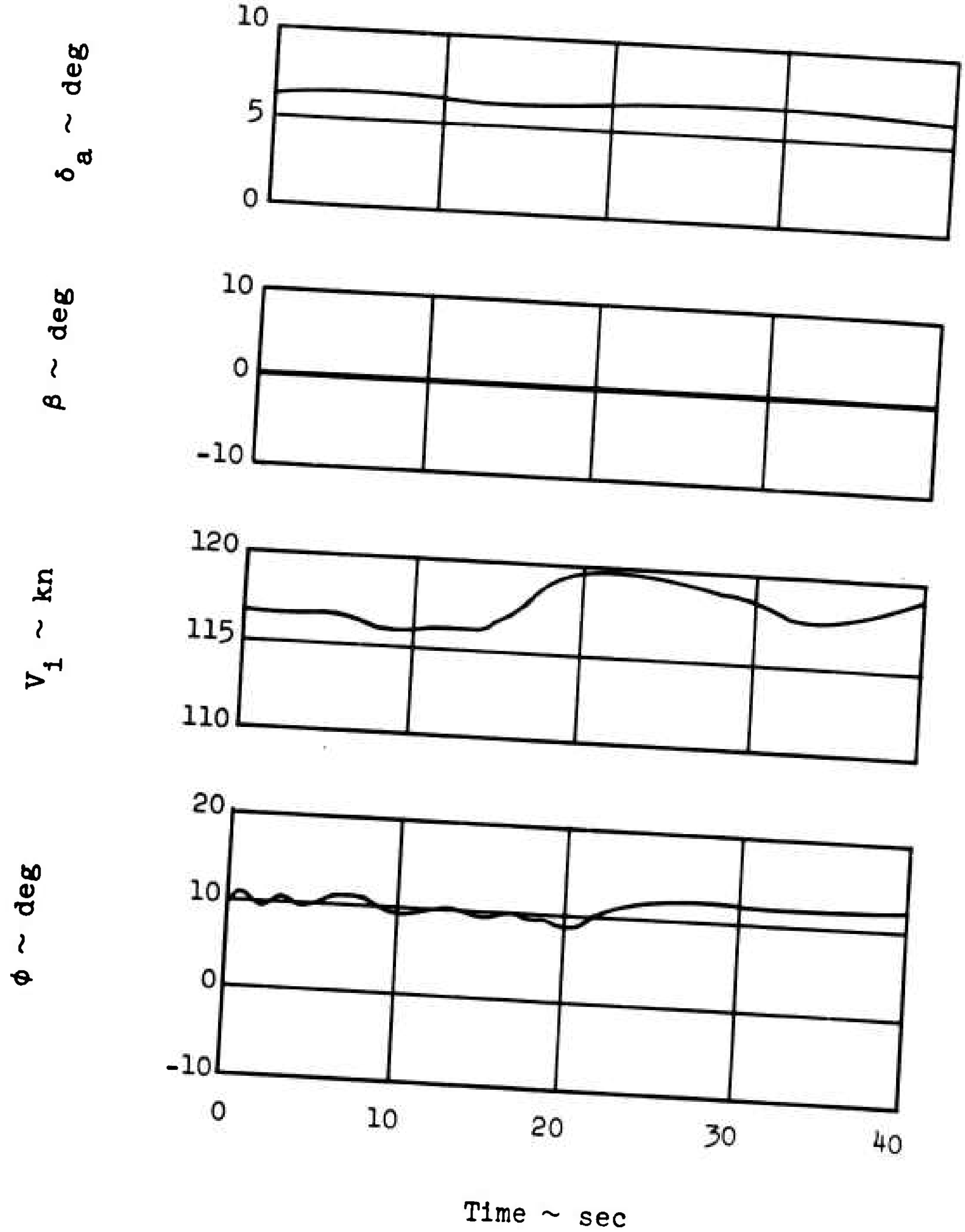

Figure 51. Lateral/Directional Stability, Spiral Stability, 0° Camber, BLC ON, 122 Knots, Right Turn.

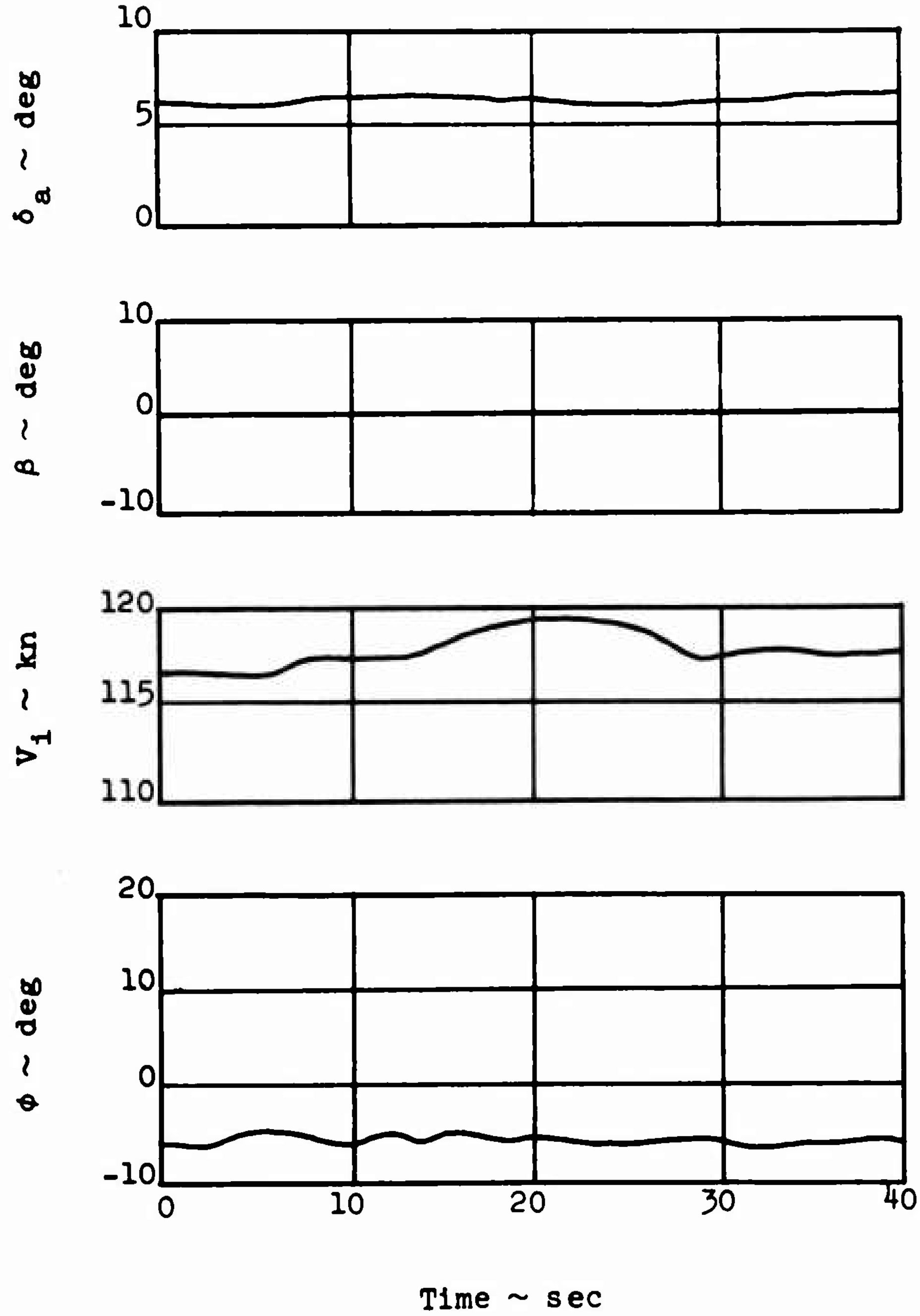

Figure 52. Lateral/Directional Stability, Spiral Stability, 0^o Camber, BLC ON, 122 Knots, Left Turn.

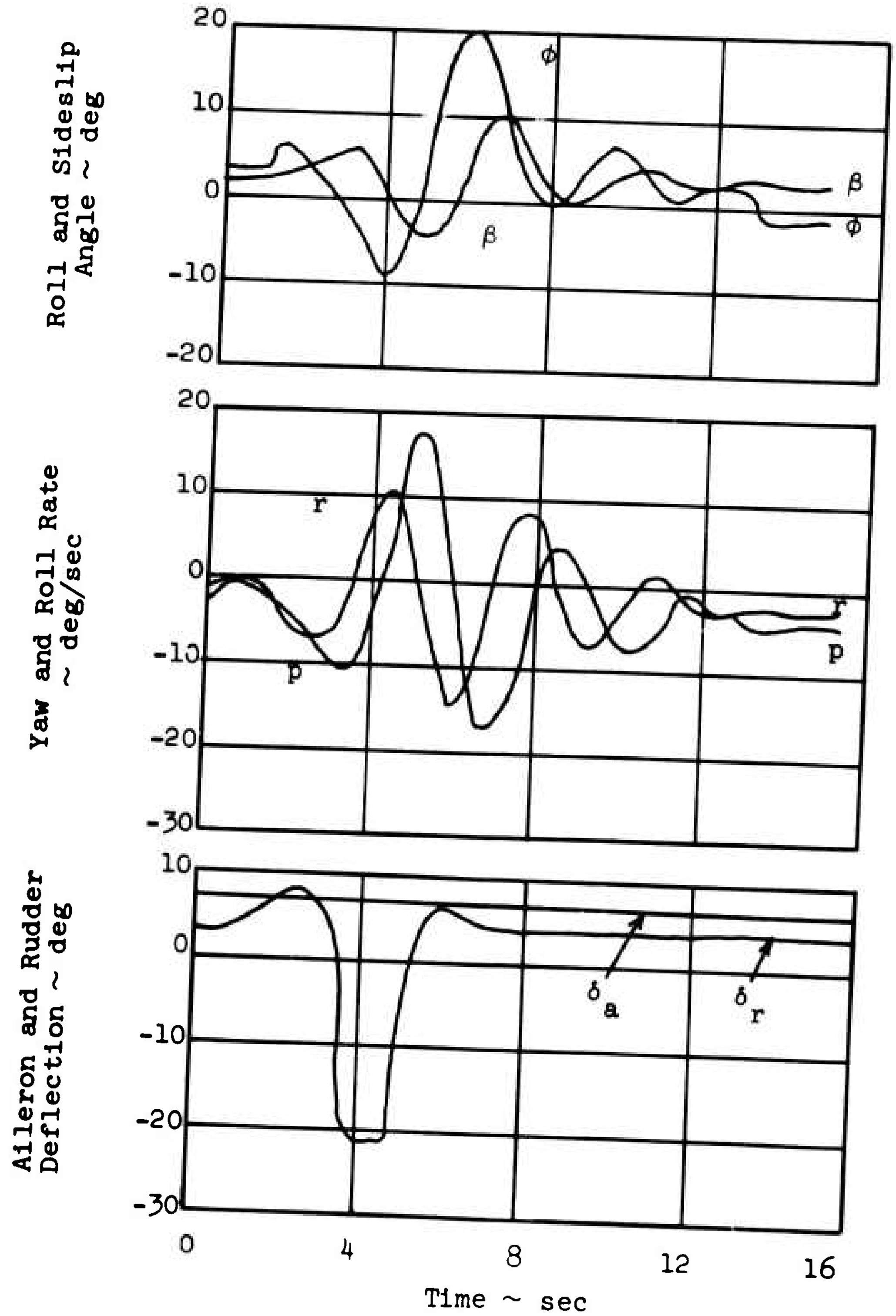

Figure 53. Dynamic Lateral/Directional Stability, Dutch Roll, 0° Camber, 71 Knots Trim Speed, BLC ON.

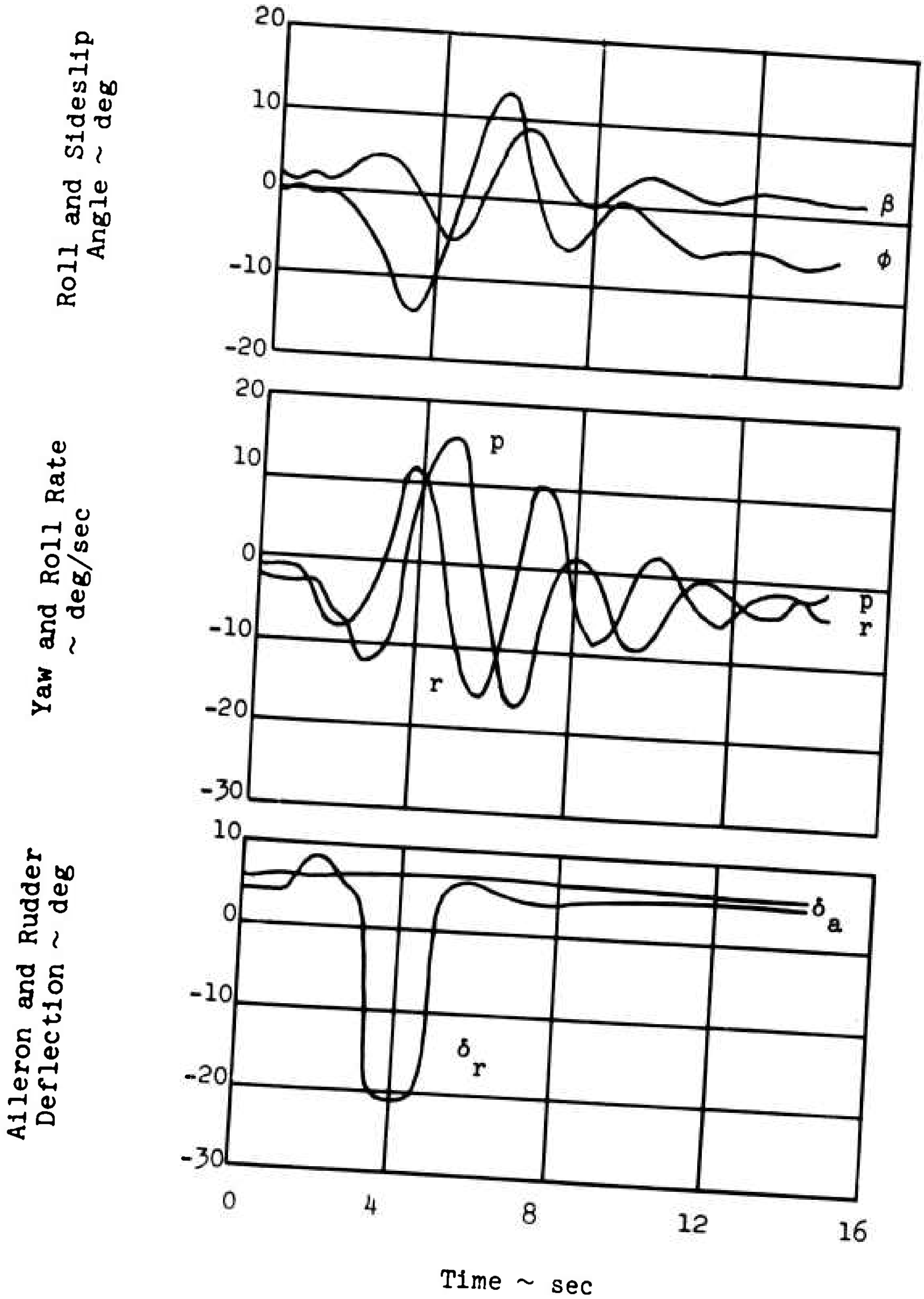

Figure 54. Dynamic Lateral/Directional Stability, Dutch Roll, 15° Camber, 71 Knots Trim Speed, BLC ON.

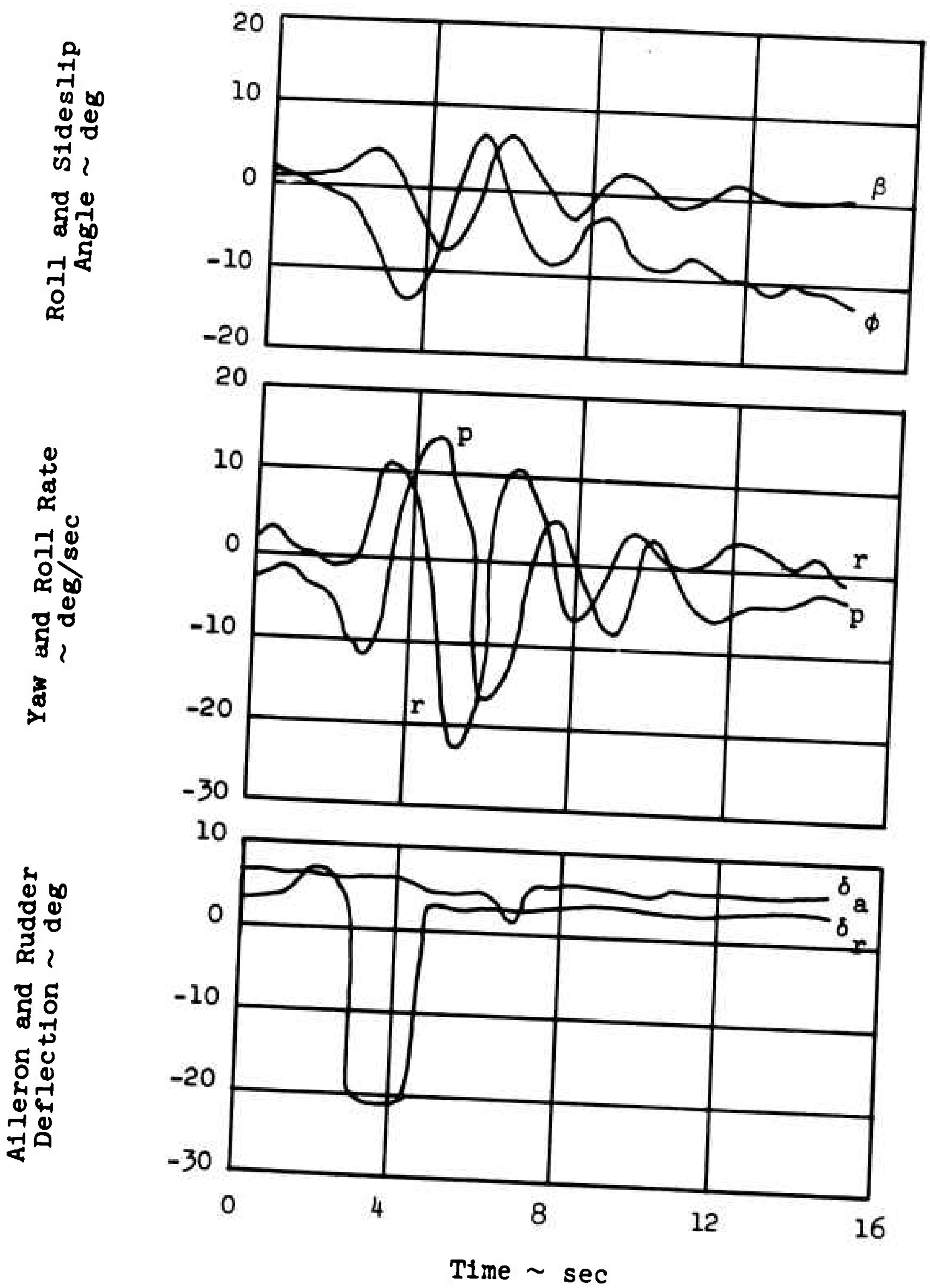

Figure 55. Dynamic Lateral/Directional Stability, Dutch Roll, 30° Camber, 71 Knots Trim Speed, BLC ON.

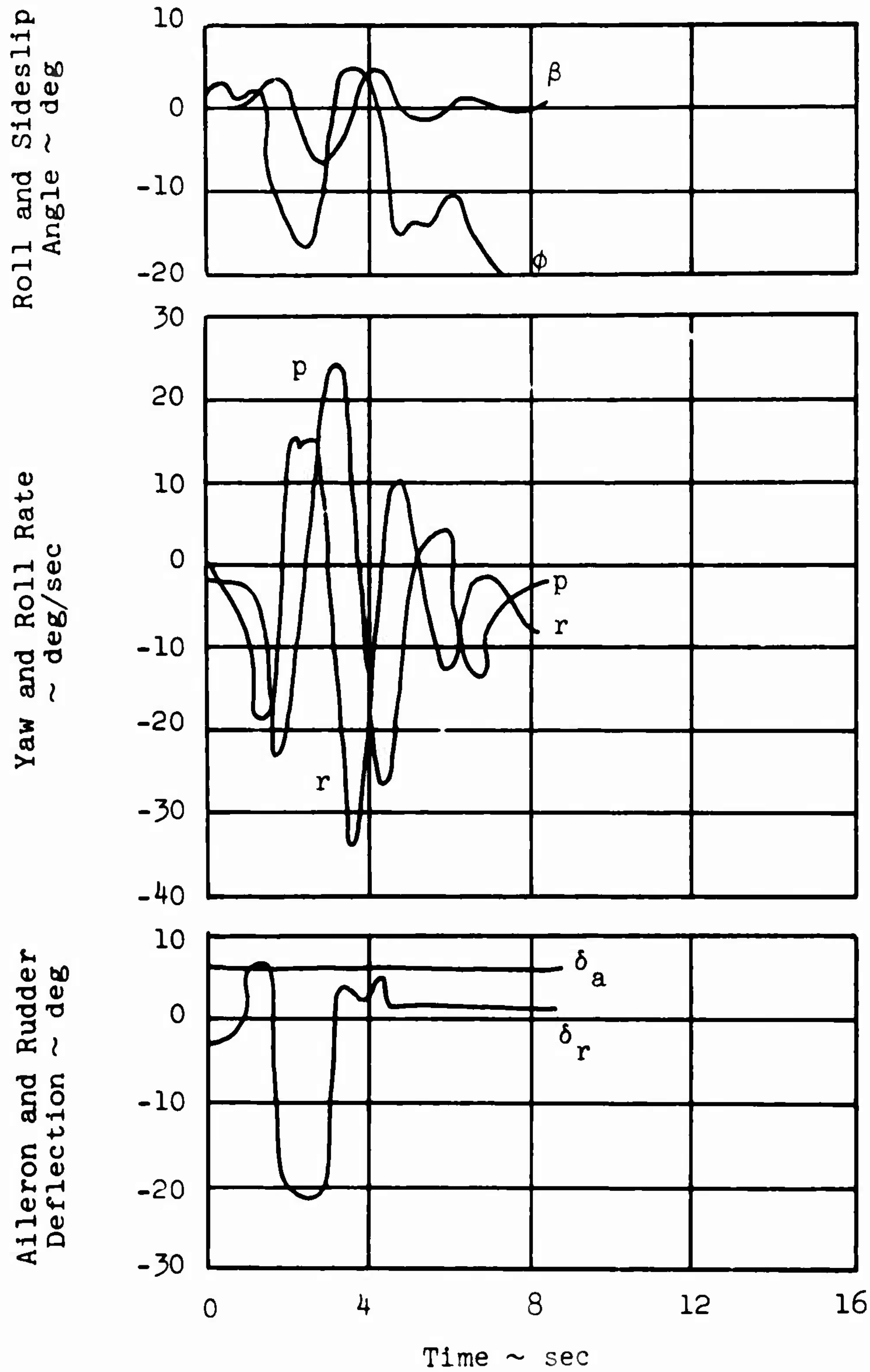

Figure 56. Dynamic Lateral/Directional Stability, Dutch Roll, 0° Camber, 122 Knots Trim Speed, BLC ON.

NOTE: Power Increase From 108 SHP to 199 SHP, No Change in Trim Setting

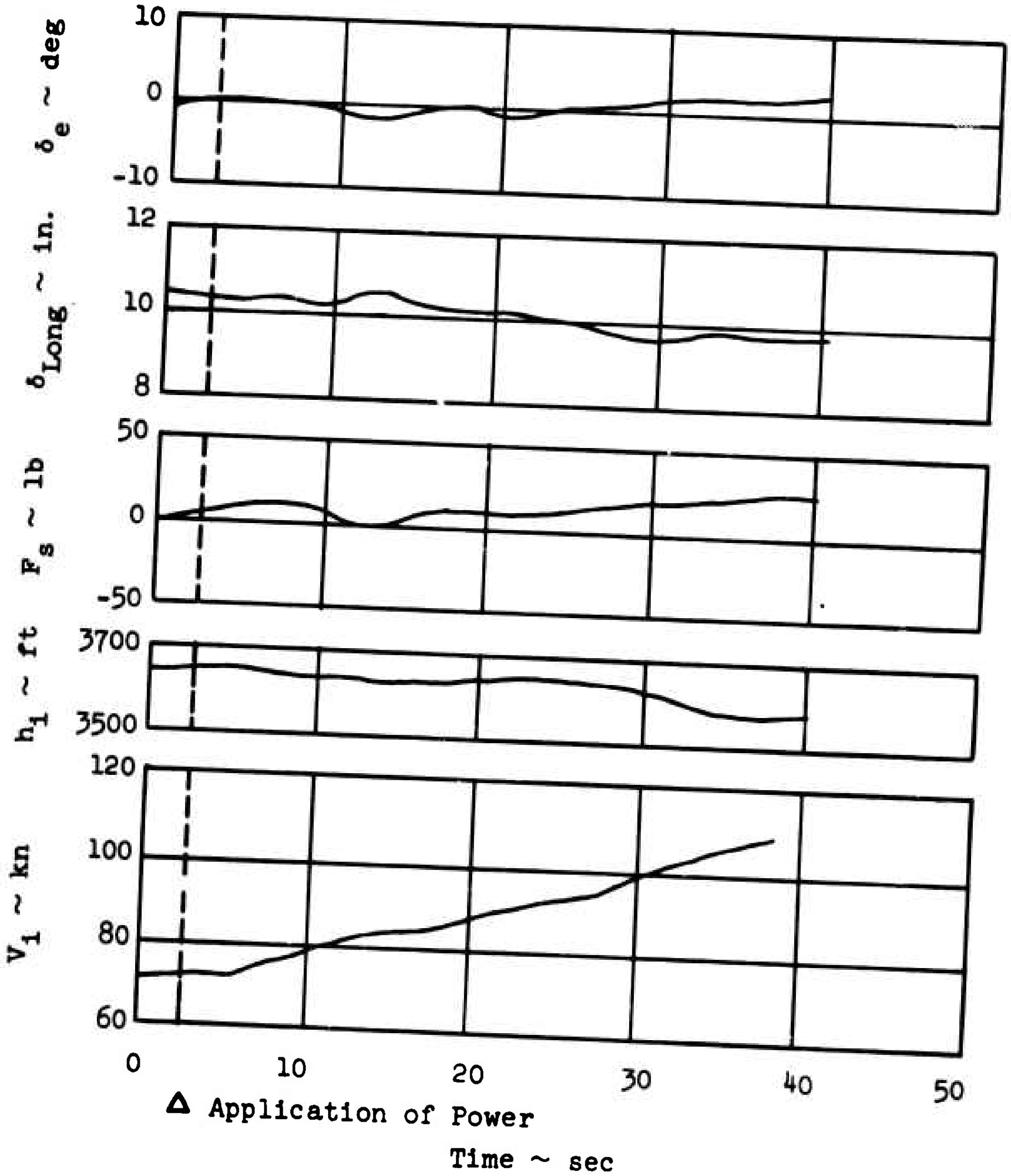

Figure 57. Out-of-Trim Characteristics, Power Effect, 0° Camber, BLC ON, AFT CG, 71 Knots Trim Speed.

NOTE: Power Increase From 146 SHP to 233 SHP, No Change in Trim Setting

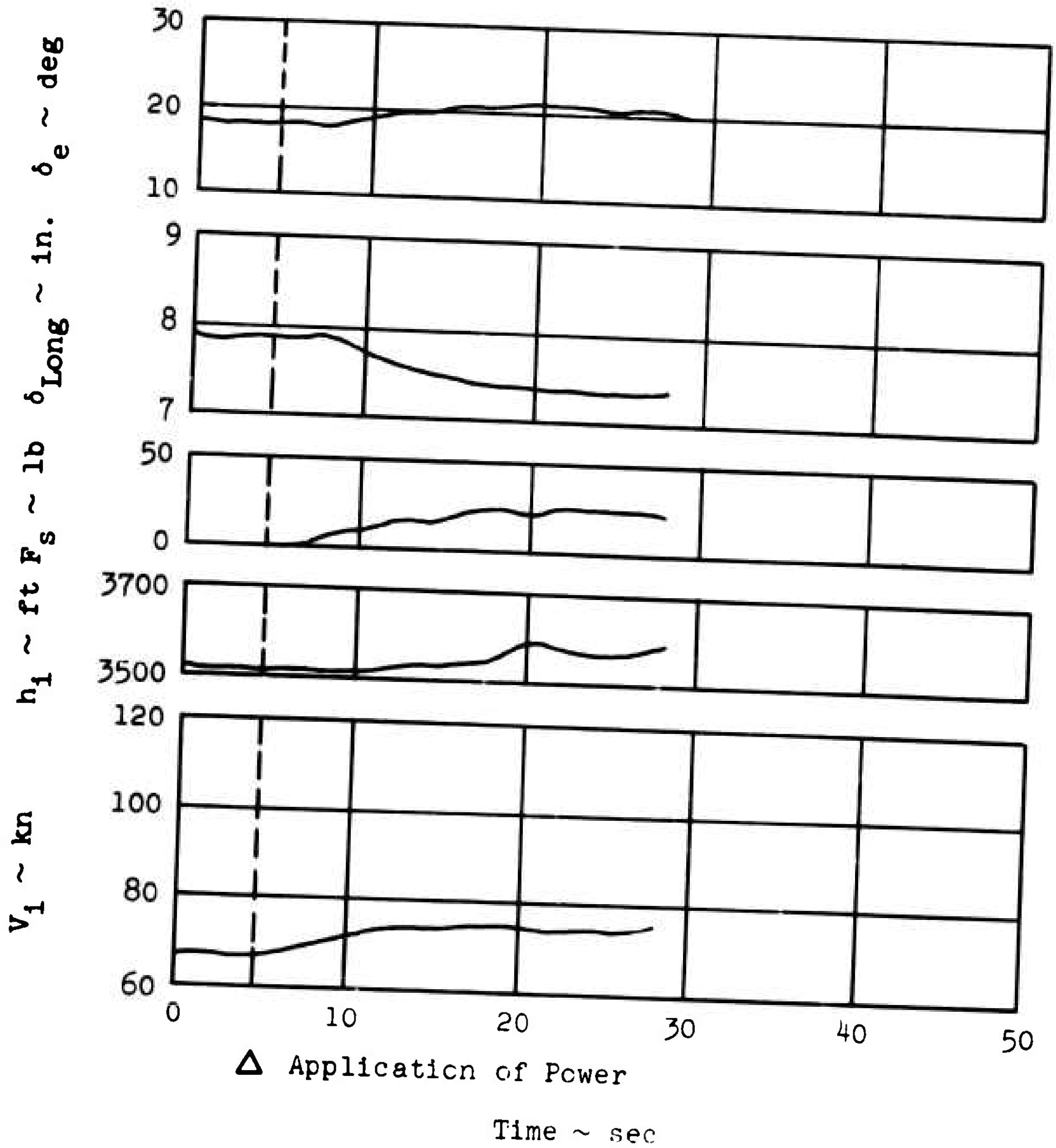

Figure 58. Out-of-Trim Characteristics, Power Effect, 30° Camber, BLC ON, AFT CG, 71 Knots Trim Speed.

NOTE: No Change in Trim Setting

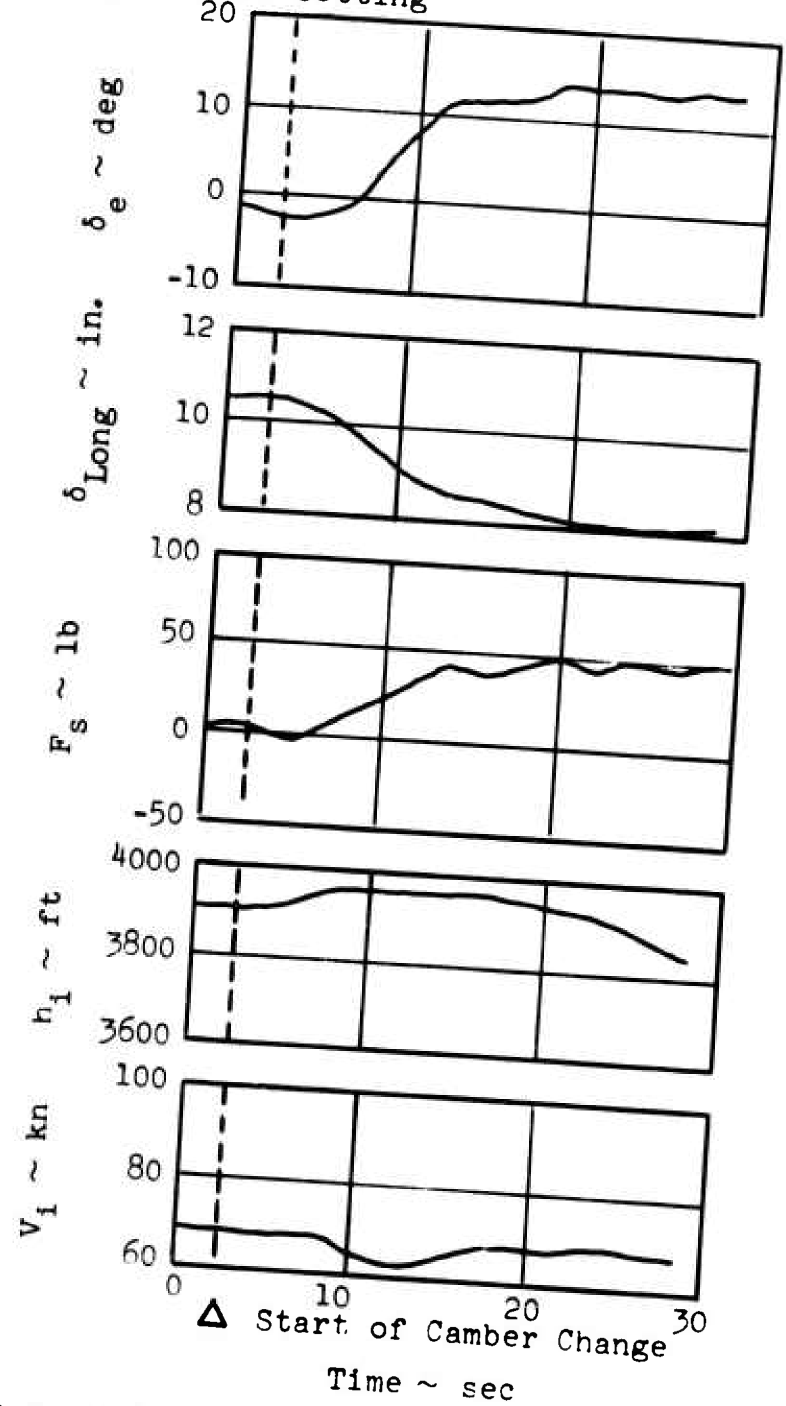

Figure 59. Out-of-Trim Characteristics, Camber Effect, 0° to 30° Camber, BLC ON, AFT CG, 71 Knots Trim Speed.

NOTE: No Change in Trim Setting

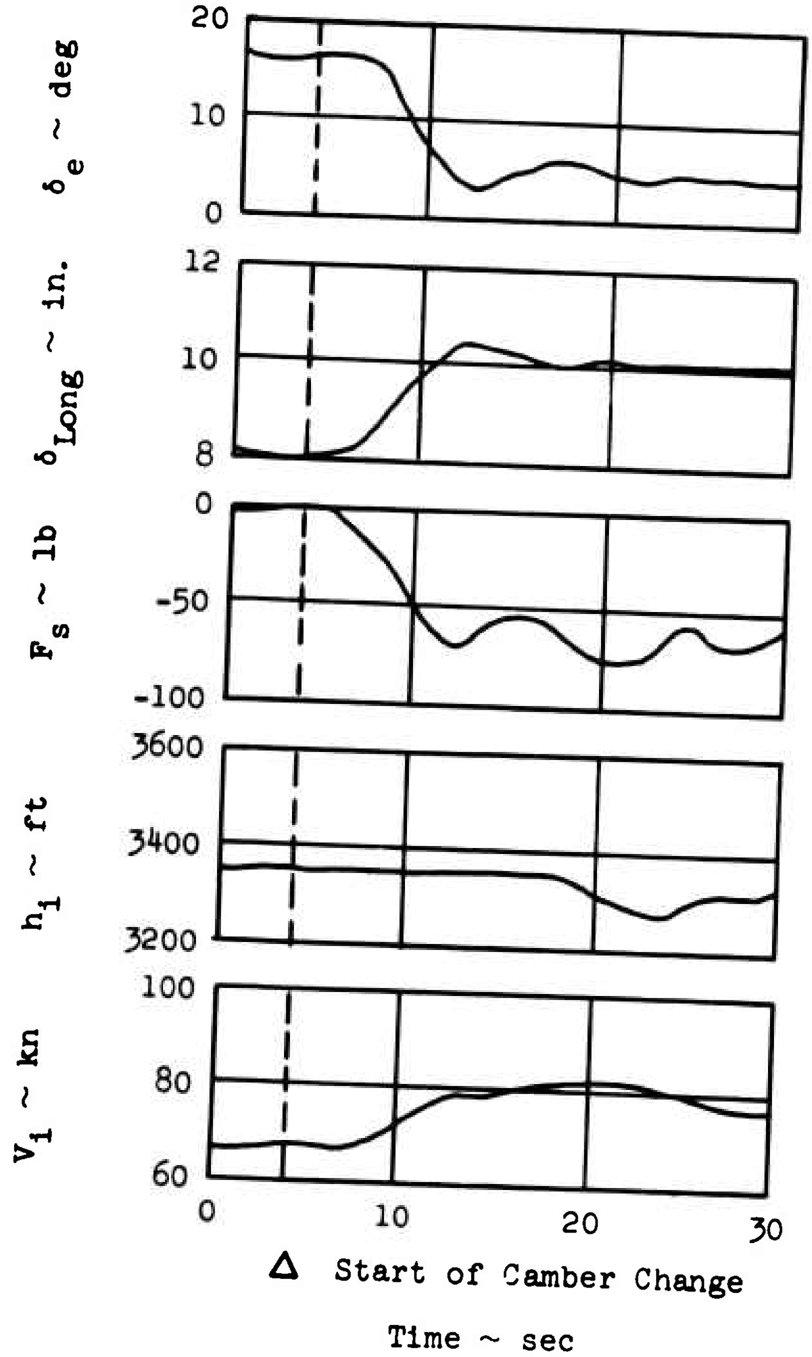

Figure 60. Out-of-Trim Characteristics, Camber Effect, 30° to 0° Camber, BLC ON, AFT CG, 71 Knots Trim Speed.

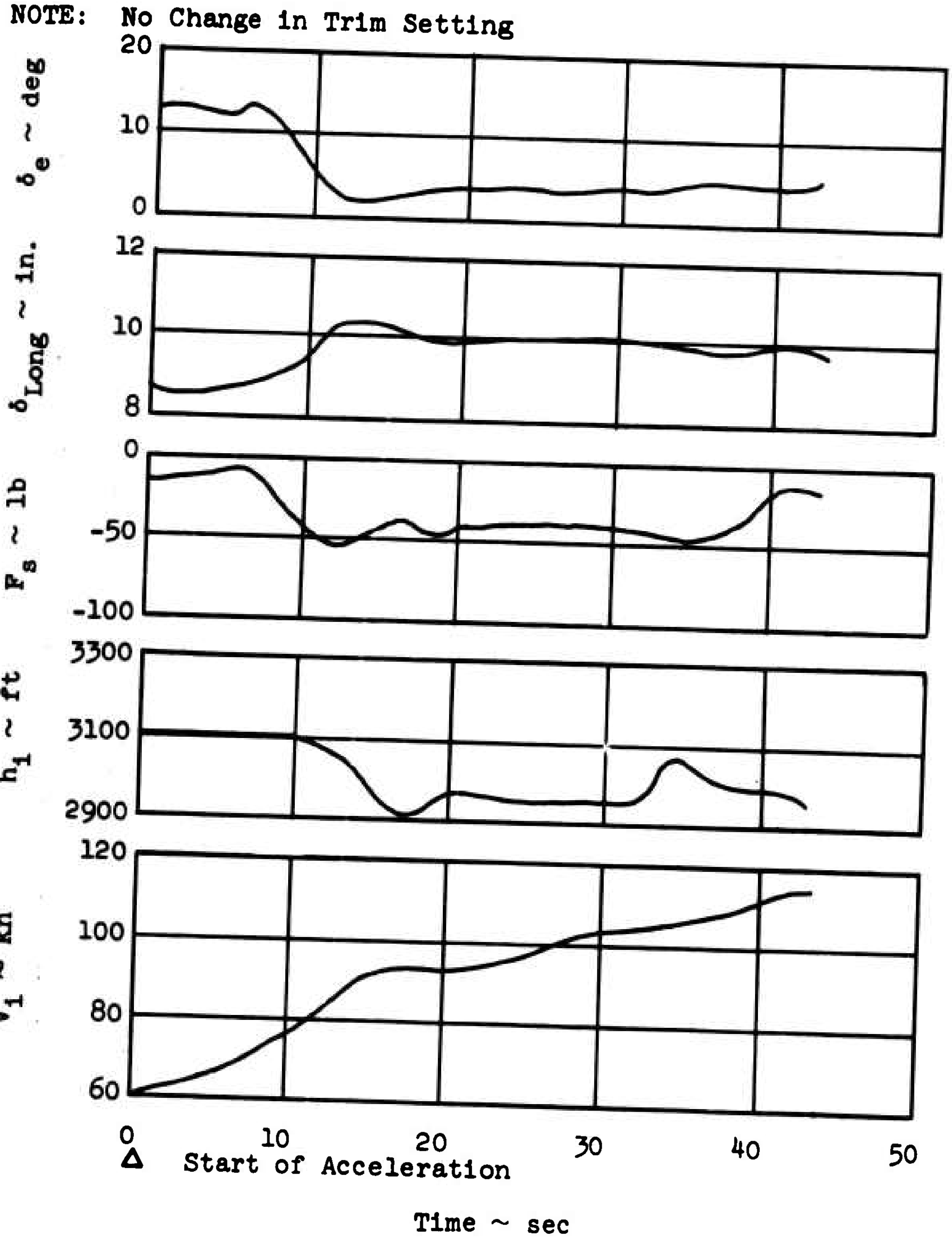

Figure 61. Out-of-Trim Characteristics, Acceleration, 30° to 0° Camber, BLC ON, AFT CG, 62.2 Knots Trim Speed.

NOTE: No Change in Trim Setting

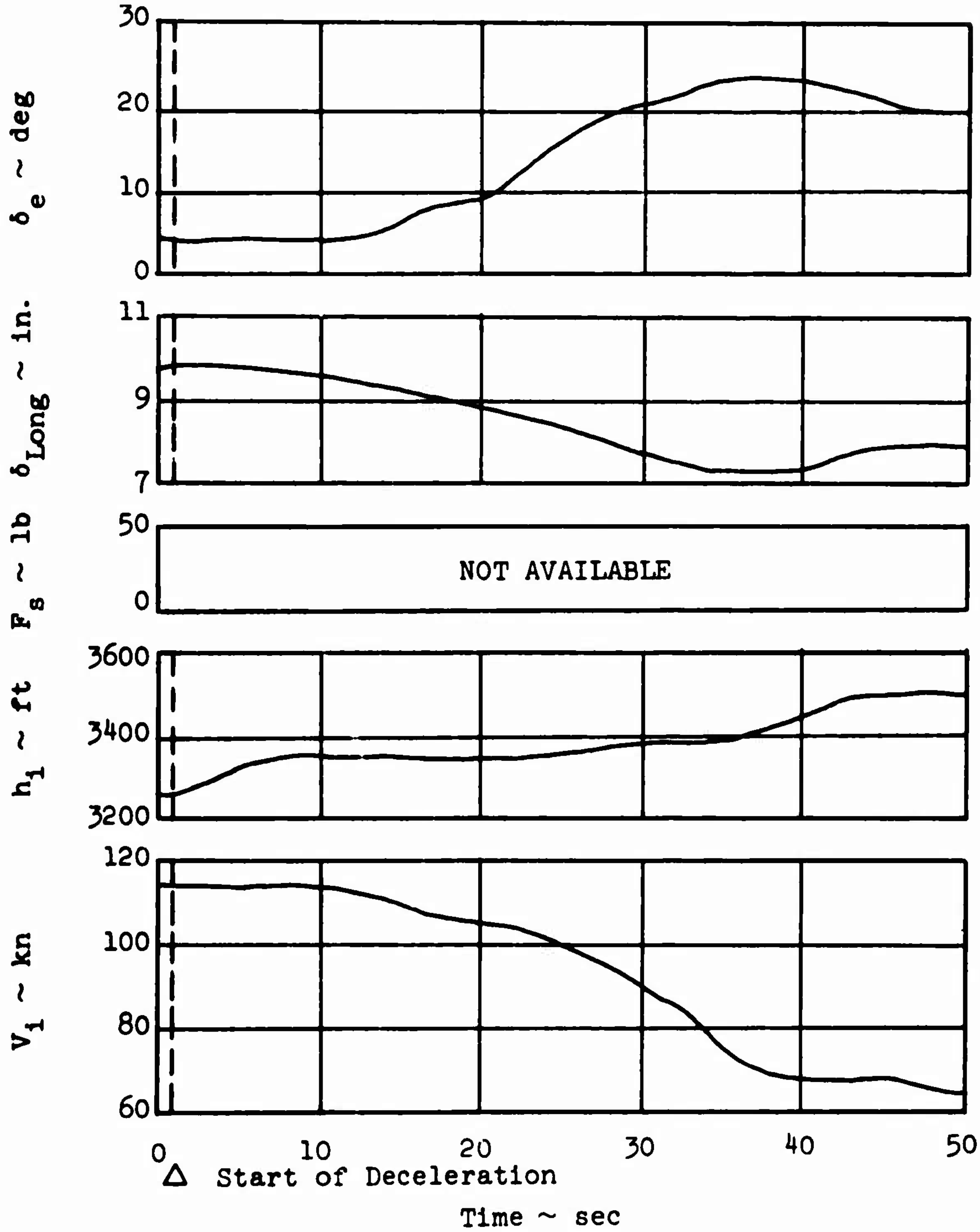

Figure 62. Out-of-Trim Characteristics, Deceleration, 0° to 30° Camber, BLC ON, AFT CG, 122 Knots Trim Speed.

NOTE: Trim Tab Change From -3.5° to -19.8°

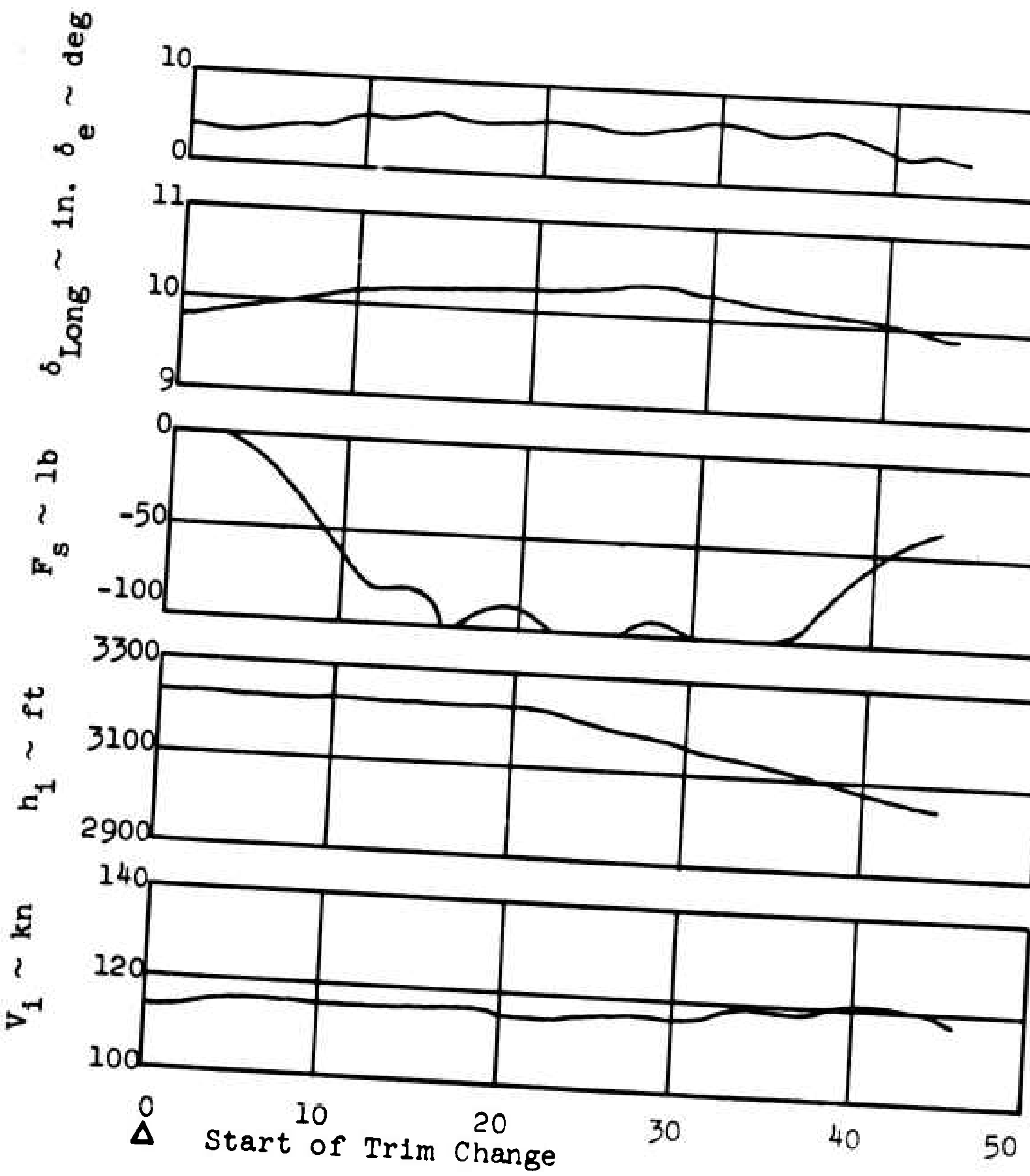

Figure 63. Out-of-Trim Characteristics, Trim Tab Effect, Nose-down Trim, 0° Camber, BLC ON, AFT CG, 122 Knots Trim Speed.

NOTE: Trim Tab Change From -3.5° to +8.0°

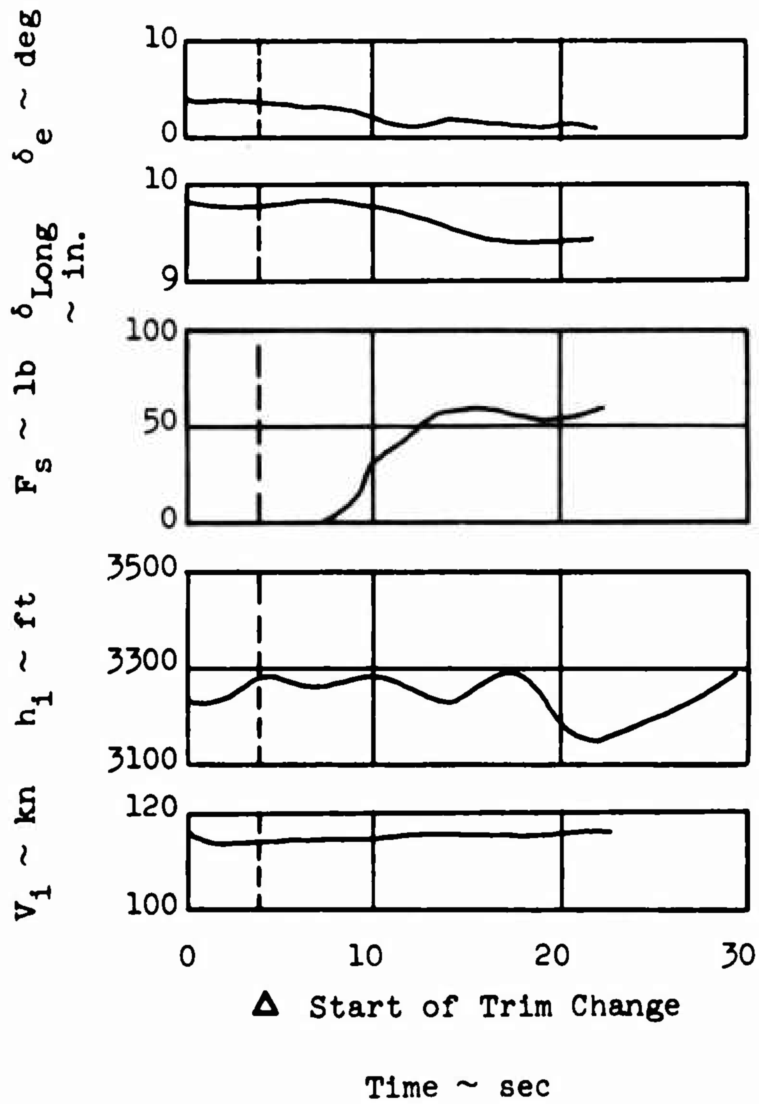

Figure 64. Out-of-Trim Characteristics, Trim Tab Effect, Nose-Up Trim, 0° Camber, BLC ON, AFT CG, 122 Knots Trim Speed.

APPENDIX
AIRCRAFT GEOMETRY

Areas

Wing:	
Area, total, including ailerons and 19.8 square feet of fuselage	105.9 square feet
Area directly affected by variable camber	58.14 square feet
Aileron area, both sides, aft of hinge line	11.13 square feet
Shroud:	
Projected area not including control surfaces, struts or fairings	15.58 square feet
Longitudinal control surface:	
Fixed portion of surface excluding shroud and including 0.50 square foot of propeller hub	7.38 square feet
Elevator, both sides, aft of elevator hinge line, including elevator balance and 0.77 square foot of trim tab and not including propeller hub	8.67 square feet
Directional control surfaces:	
Fixed portion of surface including 0.50 square foot of propeller hub	2.92 square feet
Rudders aft of rudder hinge line not including propeller hub	4.83 square feet
Wetted areas:	
Fuselage	234.40 square feet
Wing	177.40 square feet
Landing gear	54.76 square feet
Shroud	94.60 square feet

Shroud strut, upper	5.95 square feet
Shroud strut, lower	0.63 square foot
Longitudinal control surfaces	32.00 square feet
Directional control surfaces	14.93 square feet

<u>Dimensions and General Data</u>

Wing:	
Span	314.40 inches
Chord:	
At fuselage center line	58.20 inches
At theoretical tip	38.80 inches
Mean geometric chord located 76.36 inches outboard of fuselage center line	49.15 inches
Airfoil section:	
NACA 63615 modified by raising the trailing edge of the airfoil aft of the 35-percent point by 2.1 percent of the chord and holding the leading-edge radius to a constant 1.25 inches along wing span	
Airfoil maximum thickness:	
At root (BL 25.25 inches)	15 percent
At theoretical extended tip (BL 157.20 inches)	15 percent
Incidence:	
At root (BL 25.25 inches)	1.000 degree
At theoretical tip (BL 157.20 inches)	-1.117 degrees
Sweepback at 25-percent chord	0.750 degree
Dihedral (at chord line)	1.750 degrees
Aspect ratio	6.48

Taper ratio (C_T/C_R)	.667
Ailerons:	
Span (total in percent of wing span)	26.7 percent
Chord (percent wing chord)	45.4 percent
Distance from plane of symmetry to centroid of aileron area	128.70 inches
Aerodynamic balance	0 percent
Static balance	100 percent
Camber change:	
Span (total in percent of wing span)	53.5 percent
Percent of wing chord geometrically changed by camber change	65 percent
Shroud:	
Diameter - outside	75.00 inches
- inside	66.00 inches
Chord excluding fairing at fuselage attachment	30.00 inches

Airfoil section - 15-percent-thick section developed by Mississippi State University's Dept. of Aerophysics and Aerospace Engineering closely resembling the NACA 64_3-615 airfoil section. The shroud airfoil section chord line makes an angle of -1 degree, 20 minutes with respect to the propeller center line.

Incidence - Shroud and propeller center line makes an angle of -45 minutes with respect to the fuselage reference line (TE up).

Longitudinal control surface:
Stabilizer (including elevator):
Span including 1-foot width

of propeller hub	114.00 inches
Chord:	
At outer surface of propeller hub (BL 6 inches) and inner surface of shroud (BL 35 inches)	18.00 inches
At outer surface of shroud (BL 36 inches)	40.00 inches
At tip	18.00 inches
Airfoil section NACA 0010, incidence	-45 minutes
Sweep of leading edge:	
Inboard of shroud	0 degrees
Outboard of shroud	52 degrees
Dihedral	0 degrees
Elevator:	
Span, one side	57.00 inches
Chord, from hinge line aft, constant across span except for 6 inches overhang at elevator balance	12.00 inches
Surface has felt seal at hinge line with essentially no aerodynamic balance	
Static balance	50 percent
Tab span	26.00 inches
Tab chord	4.25 inches
Directional control surface:	
Stabilizer (including rudder):	
Span including 12 inches of propeller hub	70.00 inches
Chord, constant across span	18.00 inches
Airfoil section NACA 0010	

Sweep of leading edge	0 degrees
Rudder:	
Span, one "side" (top or bottom)	35.00 inches
Chord	12.00 inches
Surface has felt seal at hinge line with essentially no aerodynamic balance. No static balance is provided.	
Height of highest fixed part of aircraft above ground in static, level fuselage reference line, attitude	104.25 inches
Height of wing chord plane, at wing MGC quarter chord, above ground in static, level fuselage reference line, attitude	59.50 inches
Length, fuselage reference line level	279.75 inches
Distance from wing MGC quarter chord to shroud quarter chord	132.81 inches
Distance from wing MGC quarter chord to longitudinal control surface hinge line	153.95 inches
Distance from wing MGC quarter chord to directional control surface hinge line	153.95 inches
Angle between fuselage reference line and wing zero-lift line	3.0 degrees
Rotation ground clearance (undeflected tail skid)	9.6 degrees
Wheel and tire size, all wheels	5.00 X 5
Tread to center line of wheels (empty weight)	6.5 feet
Wheelbase	46.25 inches