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USATRECOM TECHNICAL REPORT 64-55

XV-8A FLEXIBLE WING AERIAL UTILITY VEHICLE

By

H. Kredit

February 1965

U. S. ARMY TRANSPORTATION RESEARCH COMMAND

FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-121(T)

RYAN AERONAUTICAL COMPANY

SPONSORED BY:

ADVANCED RESEARCH PROJECTS AGENCY

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The current emphasis on limited warfare has indicated that a requirement exists for a simple, inexpensive, and easily maintained and operated concept for positive air cargo delivery to remote areas. The XV-8A Flexible Wing Aerial Utility Vehicle, potentially, may satisfy this requirement.

This Command concurs with the conclusions reached in this report, and recommends that further flight tests be conducted by Army research pilots and user-type organizations. Only through a user-type evaluation can the specific mission and capabilities of this type of vehicle be determined.

This report is published for the dissemination of information and for the stimulation of discussion and consideration of this concept for delivery of air cargo in remote areas.

Task 1D121401A14172
Contract DA 44-177-AMC-121(T)
USATRECOM Technical Report 64-55
February 1965

XV-8A
FLEXIBLE WING AERIAL UTILITY VEHICLE

REPORT NO. 64B082A

This research was supported by the Advanced Research Projects Agency of the Department of Defense and was monitored by the U.S. Army Transportation Research Command (USATRECOM) under Contract DA 44-177-AMC-121(T).

Prepared by
RYAN AERONAUTICAL COMPANY
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for
U. S. ARMY TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

XV-8A Aircraft in Flight

ABSTRACT

The results of a test program of a flexible wing manned utility vehicle are presented. Discussed are performance characteristics, handling qualities and operational flight envelopes. Included is a supplemental flight test report in the Addendum which reflects configuration changes.

PREFACE

This report has been prepared by the Ryan Aeronautical Company, 2701 Harbor Drive, San Diego, California, as authorized under Contract DA 44-177-AMC-121(T).

The report discusses the XV-8A flight test program. The project was supported by the Advance Research Projects Agency of the Department of Defense and was monitored by the U. S. Army Transportation Research Command. All testing was conducted at the Yuma Proving Grounds, Yuma, Arizona between 5 February 1964 and 28 April 1964. Airborne test activity at that locale provided aircraft support and work space facilities.

This document, entitled "XV-8A FLEXIBLE WING AERIAL UTILITY VEHICLE" was authored by H. Kredit, Flight Test Engineer, and approved by P. Girard, Project Engineer. Technical Editor was B. Haldeman and Art Editor was E. Cornell.

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ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

SYMBOLS

Symbol	Units	Nomenclature
A	ft^2	Area (for XV-8A = 450 ft^2)
AR		Aspect Ratio b^2/S
a	ft/sec^2	Acceleration
b	ft.	Wing Span (for XV-8A = 33.4 ft.)
BHP	550 ft. (lb/sec)	Actual Brake Horsepower
BHP_{CH}	550 ft. (lb/sec)	Chart Brake Horsepower
CAS	MPH	Calibrated Airspeed
C_D		Drag Coefficient
C_{D_o}		Parasite Drag Coefficient
C_{D_i}		Induced Drag Coefficient
C.G.		Center of Gravity (STA.)
C_L		Lift Coefficient
D	LB	Drag
e		The Square Root of Oswald's Span Efficiency Factor
F	LB	Force
g	32.2 ft/sec^2	Acceleration Due to Gravity
H (h)	ft	Absolute Altitude (Tapeline)
H_D (h_d)	ft	Density Altitude
H_p	ft	Pressure Altitude

SYMBOLS (Continued)

Symbol	Units	Nomenclature
ΔH_{p_e}	ft.	Altimeter Position Error Correction
ΔH_{p_i}	ft.	Altimeter Instrument Error Correction
IAS	MPH	Indicated Airspeed
i_w	DEG	Wing Incidence Angle
MAP	in. Hg	Manifold Pressure
OAT	°C	Outside Air Temperature
P	in. Hg.	Atmospheric (Static) Pressure
ΔP_p	in. Hg.	Static Pressure Error
q	lb/ft^2	Dynamic Pressure
R/C	ft/min	Rate of Climb
R/D	ft/min	Rate of Descent
RPM	1/min	Engine Revolutions per Minute
S	$ft.^2$	Wing Area
S	ft.	Take-off or Landing Distance
t	Min	Time
T	°K	Temp. in Degrees Kelvin (°C + 273)
TAS	MPH	True Airspeed
V	MPH (ft/sec)	Observed Airspeed
ΔV_i	MPH (ft/sec)	Airspeed Instrument Error Correction
V_i	MPH (ft/sec)	Indicated Airspeed, $V_o + \Delta V_i$

SYMBOLS (Continued)

Symbol	Units	Nomenclature
ΔV_{p_e}	MPH (ft/sec)	Airspeed Position Error Correction
V_c	MPH (ft/sec)	Calibrated Airspeed $V_1 + \Delta V_{pe}$
ΔV_c	MPH (ft/sec)	Compressibility Correction
V_e	MPH (ft/sec)	Equivalent Airspeed $V_c - \Delta V_c$
V_T	MPH (ft/sec)	True Airspeed, $V_c/\sqrt{\sigma}$
V_g	MPH (ft/sec)	Ground Speed
W		Weight
W_f	gpm	Fuel Flow

Subscripts

a	Aileron
c	Compressibility, Corrected
ch	Chart
d	Density
e	Elevator, Equivalent
ew	Equivalent Weight Designation
g	Ground
i	Indicated, Instrument
o	Observed, or Sea Level Standard
p	Pressure, Propeller
r	Rudder

SYMBOLS (Continued)

Symbol	Units	Nomenclature
s		Standard Conditions or Standard Values
t		Total
w		Wind

Greek Symbols

Symbol	Units	Nomenclature
α	Deg.	Angle of Attack
β	Deg.	Angle of Sideslip
δ	Deg.	Control Surface Deflection
η		Propulsive Efficiency
γ	Deg.	Flight Path Angle to the Horizontal
ρ	$Slugs/ft^3$	Mass Density
σ		Density Ratio, ρ/ρ_o
ϕ	Radians	Wing Tip Helical Path Angle
ψ	Deg.	Angle of Yaw
μ		Coefficient of Friction

SUMMARY

The purpose of this test program was to determine the performance characteristics and over-all handling qualities to establish the operational flight envelope of the XV-8A Flexible Wing Aerial Utility Vehicle.

Standard performance, stability and control flight testing techniques were employed during all phases of operation. Airborne oscillograph recordings and pilot-observed instrument readings were used for data acquisition. A Fairchild Flight Analyzer camera was used to measure take-off and landing distances.

The handling characteristics of the aircraft are good. Control harmony between the longitudinal and lateral control systems is excellent, enabling the aircraft to be flown with one hand. Stability in all cases is positive with only light forces required. The flight characteristics of this airplane are similar in most respects to those found in a conventional airplane with a comparable light wing loading.

The performance capabilities of the airplane are all within predicted values. The cruise capability is such that a 100-mile mission can be flown at maximum gross weight. Take-off and landing performance proved the STOL capability of the airplane. At maximum gross weight, the take-off distance over a 50-foot obstacle is 1,000 feet. Landing distance to clear a 50-foot obstacle is 400 feet.

During the course of the test program, the airplane proved to be a reliable and easy aircraft to maintain and service. Some test operations were conducted from unprepared desert surfaces, establishing the capability for operation from areas other than regular airfields.

The operational and flying techniques are basically similar to those of lightweight conventional aircraft. The two-control system lends itself to simplicity and provides adequate control power to permit a fixed wing incidence trim setting for the entire flight including take-off, climb, cruise, descent, and landing.

CONCLUSIONS AND RECOMMENDATIONS

1. The aircraft is safe and pleasant to fly for an Army pilot of average skill. Data available indicate that, with improvements, the concept can be developed into a flying truck with reduced experience and skill requirements for the operator. Helicopter and light plane experience aids in transition to this aircraft, although such experience is by no means necessary.

2. The aircraft is capable of rough field operation with certain advantages over fixed-wing aircraft or helicopters.

3. The idea of a primitive, low-cost, low-maintenance, limited-performance but useful aerial device was clearly demonstrated. For example, only one operation out of 47 was delayed due to aircraft maintenance. This program did not represent an operational evaluation environment; however, the low maintenance and support required was very unusual for an experimental aircraft.

4. The aircraft met or exceeded all predicted performance goals and demonstrated its ability to haul bulky cargo shapes and a useful load almost equal to its empty weight.

5. Safe landing characteristics with engine power at idle were demonstrated.

6. The system is highly sensitive to turbulence and rough air which is uncomfortable, but is self-damping to a high degree. The wing appears to lose lift in some conditions of turbulence, causing some degradation of climb and descent performance.

7. Crosswind operation investigations were continuously conducted. The results suggest that limitations will eventually be established that are quite compatible with light aircraft of about the same weight.

8. The ability of the aircraft to operate as a light STOL utility vehicle with a 100-mile range was established.

9. The concept of piloted, powered, flexible-wing vehicles appears very promising as a result of this program. Continued development by the U. S. Army also appears desirable, considering present requirements and the fast-moving conceptual changes in air mobility.

10. D. R. Simon, a U. S. Army TRECOM pilot, was checked out in the XV-8A in a three-day period at the end of the test program. His 2-hour-and-50-minute flight time included operation throughout the flight envelope, and a number of taxi runs and landings under wind conditions from calm to 15 knots. This pilot's experience and reactions established the relative ease with which the system lends itself to pilot qualification.

INTRODUCTION

The XV-8A aircraft (designated FLEEP) resulted from Ryan Aeronautical Company design studies of the application of the Rogallo flexible-wing concept to a manned aircraft. This aircraft is an improved version of the origional Ryan flexible-wing manned test vehicle.

The aircraft was designed as a single-place, lightweight utility vehicle, capable of carrying a 1000-pound payload and having short-field take-off and landing characteristics.

The primary purpose of the test program was to determine the flight characteristics and performance capabilities and to establish an operational flight envelope for the aircraft. Special attention was directed toward determining the adequacy of the longitudinal control system for perfomring the landing flare maneuver with idle power.

DESCRIPTION OF XV-8A VEHICLE

GENERAL

The description of the XV-8A aircraft is divided into four major categories:

1. Wing
2. Fuselage/gear
3. Tail
4. Power plant

and four minor additional categories:

5. Control system
6. Fuel system
7. Electrical system
8. Cockpit instruments

A three-view general arrangement drawing is shown in Figure 1. As noted on the drawing, the wing pitch pivot point was moved forward 12 inches from Fuselage Station 115.5 to Fuselage Station 103.5. This modification was made prior to the start of the test program.

WING

The wing is composed of three main structural members: a rigid center keel, and rigid right and left leading edges. The two leading edges join the keel at the apex and form a near-triangular wing planform. The keel runs longitudinally aft from the apex along the center line of the wing. The flexible membrane, made of Dacron with a polyester coating, is continuously attached to the leading edges and keel. The leading edges have a 50-degree sweep angle. The total wing area in flat planform is 450 square feet.

The wing support structure is a truss system made of aluminum tubing. The streamlined aluminum spreader bar and supporting structure are so designed and hinged as to permit the leading edges to be folded aft along the keel to facilitate ground handling and storage.

The wing is capable of being rolled ±7-1/2 degrees laterally, and moved from 0 to 30 degrees incidence angle relative to the platform.

FUSELAGE/GEAR

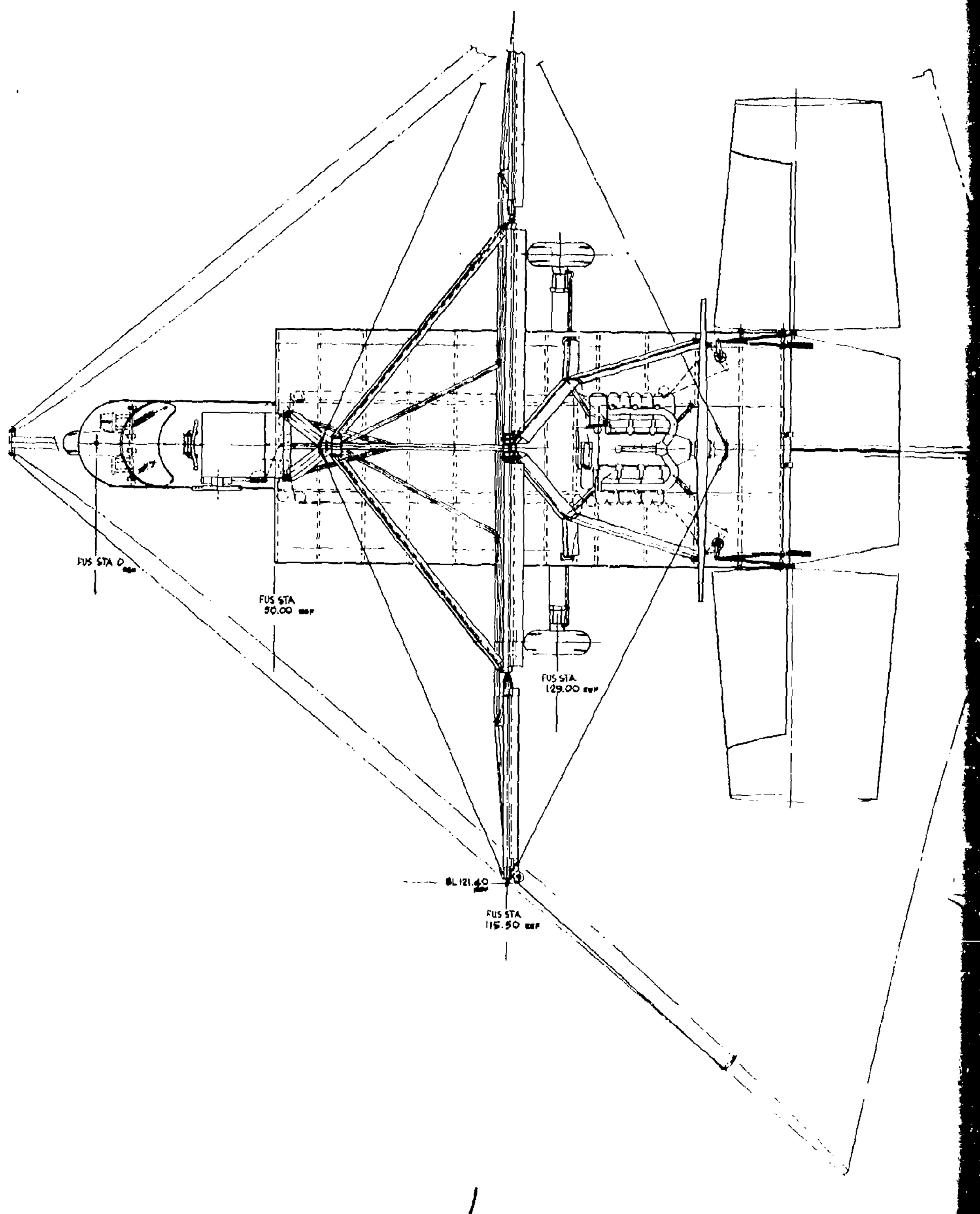
The fuselage is basically a rectangular platform of conventional riveted sheet metal construction. The platform supports the wing support structure, engine, pilot cockpit, and landing gear. The platform has a cargo loading area of 36.75 square feet. The main landing gear suspension is a single leaf spring of Fiberglas construction, semi-cantilever mounted from the cargo platform. The nose gear mounted forward at the pilot's cockpit is steerable and has a conventional oleo-type shock absorber. Brakes are provided on both main wheels and are actuated by a single toe-operated pedal mounted atop the right rudder pedal.

TAIL

The tail is a U-type with a 35-degree dihedral, and it is cantilever-mounted on the outer edges of the aft extension of the cargo bed. The stabilizers are hinged at the platform so that they can be folded inboard for ground handling and storage. The movable surfaces attached to the stabilizers incorporate an overhung balance system. In addition, a horizontal elevator is attached to the aft end of the fuselage. The total tail area is 62.93 square feet, with a total movable surface area of 46.70 square feet.

POWER PLANT

A Continental IO-360A fuel injection engine rated at 210 brake horsepower at 2800 rpm is mounted on a tubular frame in a pusher installation on the aft end of the platform. The engine is equipped with a 7-foot-diameter, two-bladed all metal, Model BHC-C2YF-1A Hartzell constant-speed propeller operated in fixed pitch. No starter or generator is installed on the engine.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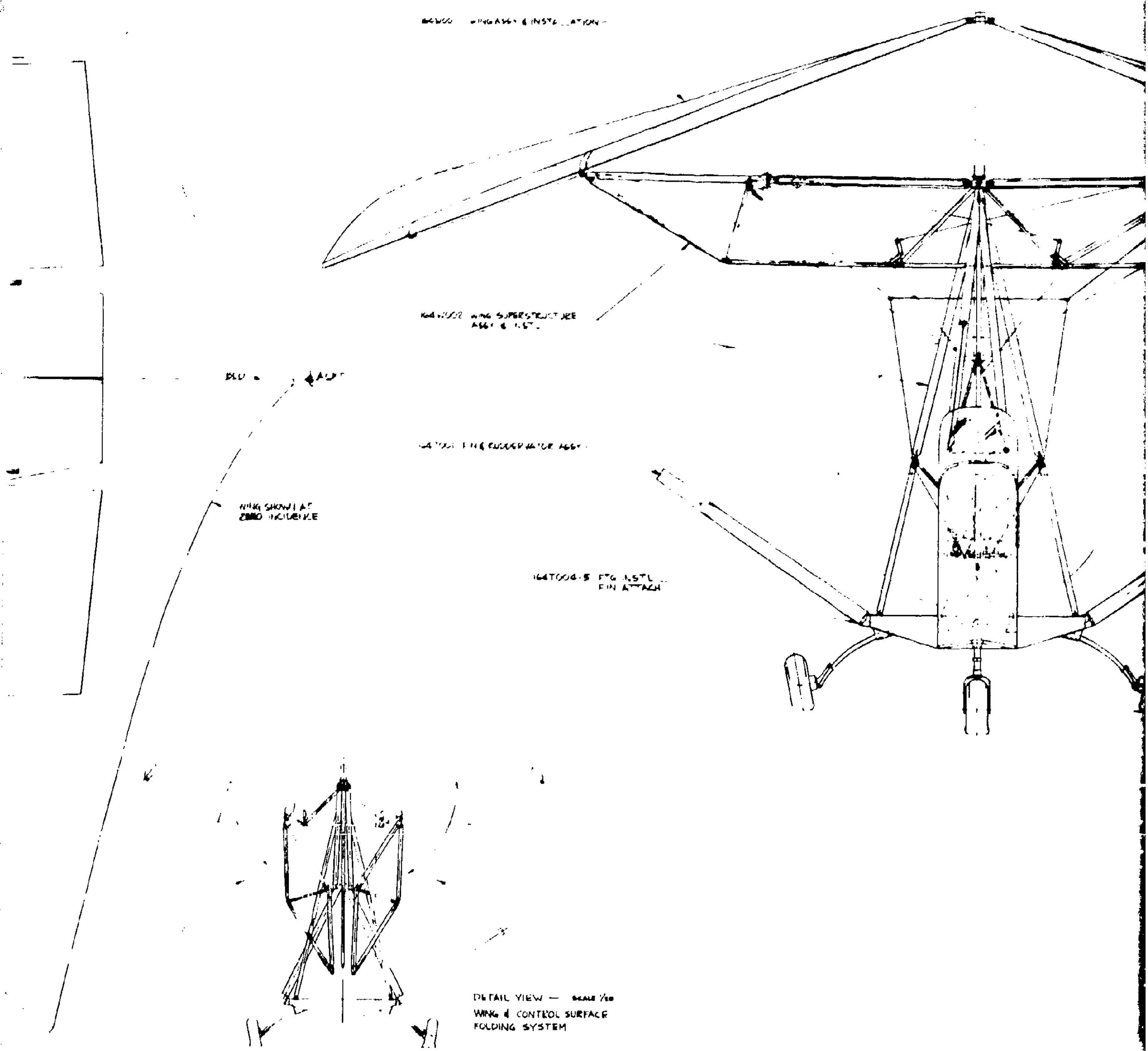
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WING & CONTROL SURFACE FOLDING SYSTEM

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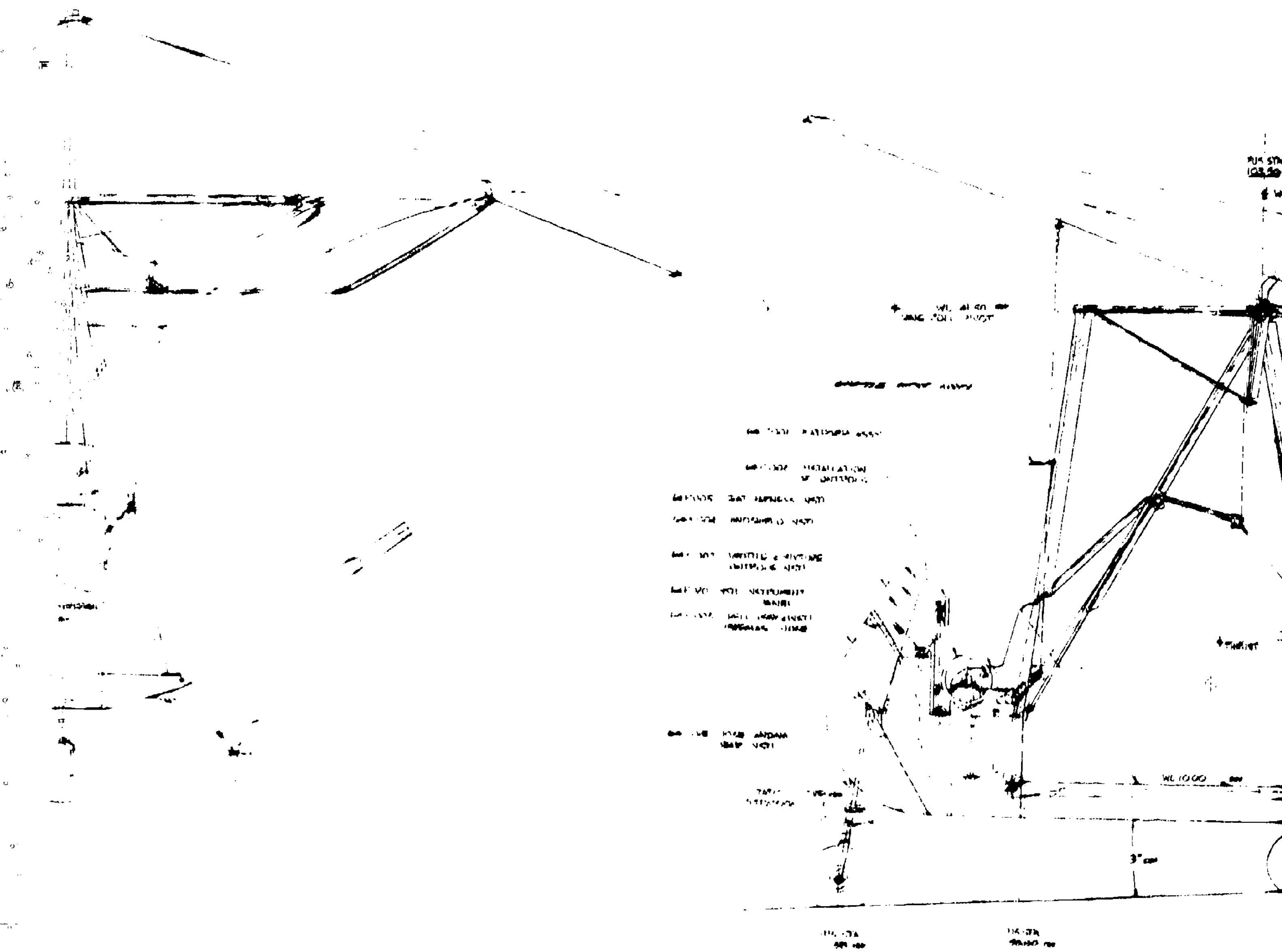
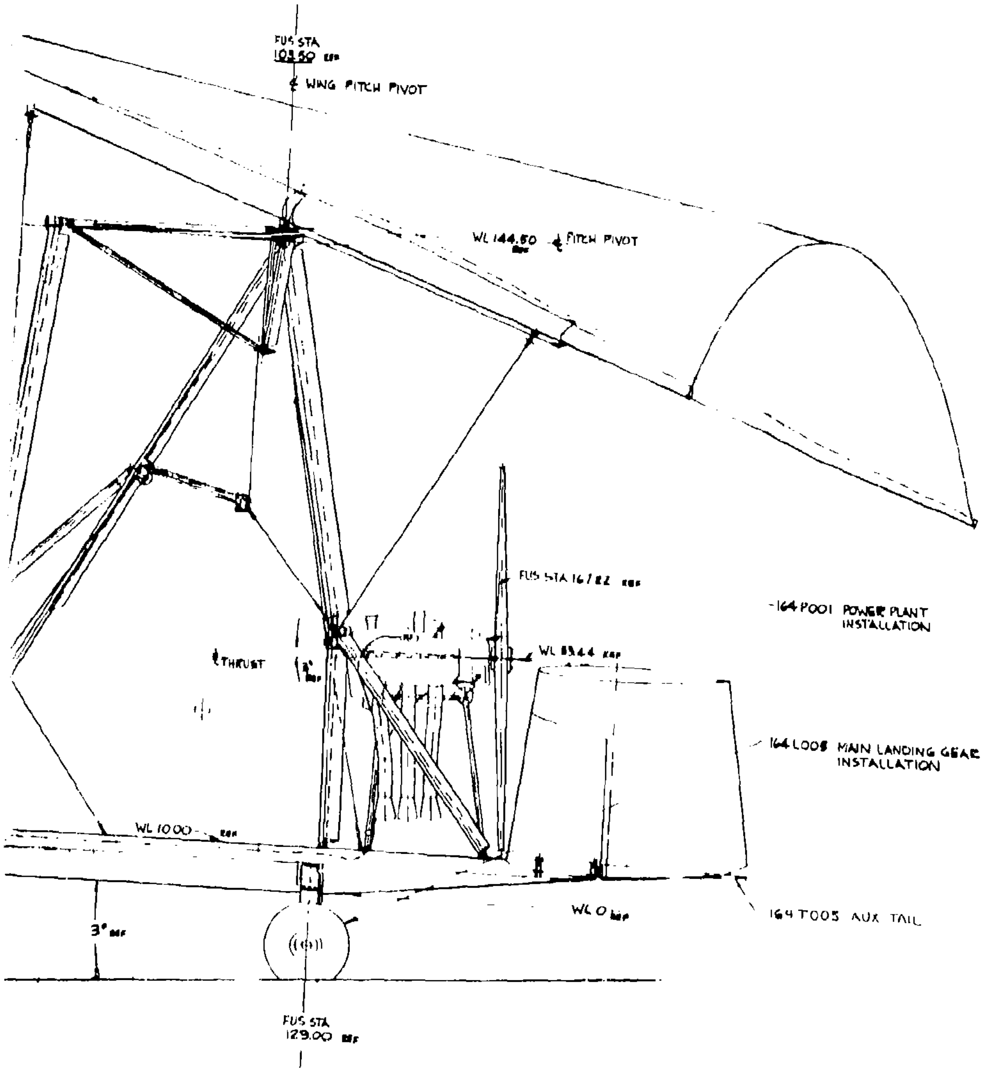




GENERAL INFORMATION

WING AREA	= 450 SQ FT
KEEL ℄ L.E LENGTH	= 26.0 FT
MAX SPAN	= 33.4 FT
L.E. SWEEP ANGLE	= 50°
T.E. SCALLOP	= 6% WING AREA
TOTAL TAIL AREA (TRUE)	= 51.0 SQ FT
MOVABLE SURFACE AREA (TOTAL)	= 34.78 SQ FT
AIRFOIL SECTION	= NACA 0012
DIHEDRAL	= 36°
POWER PLANT	= IO-360-A (CONTINENTAL)
PROP DIAMETER	= 7.0 FT
LANDING GEAR WHEELBASE	= 127.60 INS
TRACK	= 108.00 INS
TIRE SIZE	= 700x6 TYPE III (L.P)

Figure 1. XV-8A Three-View Drawing

CONTROLS

The airplane is equipped with a two-control system with the capability of converting to a three-control system. This entire test program was flown with the two-control system.

Longitudinal trim is provided by changing the wing incidence angle with respect to the fuselage platform. A trim wheel located on the left side of the pilot's cockpit allows for pilot actuation during flight. The pitch setting is automatically locked when not in use.

The lateral control system is actuated by a control wheel mounted on the upper end of the control column. The first 25 degrees of wheel deflection actuate the hinged tips of the wing leading edges. Further control wheel deflection moves the entire wing laterally with respect to the fuselage. A ground adjustable bolt rope running through the trailing edge of the wing fabric is the only means of lateral control trim.

FUEL SYSTEM

A 28-gallon fuel tank is located in the interior of the center section of the platform. An engine-driven fuel pump is used to supply fuel to the engine. An emergency fuel shut-off valve is located in the pilot's compartment. Throttle and mixture controls are located on the left side of the cockpit.

ELECTRICAL SYSTEM

There is no electrical system on the aircraft other than the engine magnetos. Electrical power for the instrumentation system and radio was provided for by conventional storage batteries which were part of the instrumentation system.

COCKPIT INSTRUMENTS

The following engine and flight instruments are located in the cockpit: oil and fuel pressure, oil and cylinder head temperature, tachometer, manifold pressure, outside air temperature, airspeed, altimeter, and rate-of-climb indicator.

TABLE 1
GENERAL INFORMATION - XV-8A AIRCRAFT

Wing Area	=	450 sq. ft.
Keel and L. E. Length	=	26.0 ft.
Maximum Span	=	33.4 ft.
L. E. Sweep Angle	=	50°
T. E. Scallop	=	6% Wing Area
Total Tail Area (true)	=	62.93 sq. ft.
Movable Surface Area (total)	=	46.70 sq. ft.
Airfoil Section (Tail)	=	NACA 0012
Dehedral	=	35°
Power Plant	=	IO-360-A (Continental)
Prop Diameter		7.0 ft.
Landing Gear Wheelbase	=	27.60 in.
Track	=	108.00 in.
Tire Size	=	700 x 6 Type III (L. P.)
Empty Weight	=	1115 lb.
Useful Load	=	1185 lb.

INSTRUMENTATION SYSTEM

GENERAL

The objective of the instrumentation task on the XV-8A aircraft was to measure dynamic loading of principal structural members of the aircraft as well as to obtain data for flight performance analysis. This was accomplished by the use of strain gages fastened to the structural members to measure stress and by the use of linear potentiometers to measure relative motion of wing, tail, and other control surfaces. Accelerometers measured g forces, and a vertical gyro measured aircraft attitude. Measurements were recorded on two oscillograph recorders.

RECORDING EQUIPMENT

Points, methods of measurement, and record readability are described in Table 2.

Two standard 26-channel Consolidated Engineering Corporation oscillographs were used to record the structural and flight data. Power for the recorders was supplied by two storage batteries. The recorders operated for approximately one hour, which was sufficient time for each flight operation.

Signals were processed through six signal conditioner boxes located adjacent to the recorders.

Figure 2 shows the recording equipment layout on the aircraft. The transducer signal voltage was supplied from the storage batteries through a special 16-volt regulator (Figure 3). The signal conditioner boxes provided signal attenuation, signal balance, and resistance calibration for the signals from the transducers.

Record identification and test event marks were made with a telephone dial connected to each recorder. By dialing the flight numbers and event numbers on the telephone dial, pulses appeared on both recording tapes at the start of each flight and record event. An event switch located on the left side of the pilot's wheel was used to indicate periodic events during a test run. The event voltage also indicated the level of the reference or signal voltage.

MEASURING INSTRUMENTS

Foil-type strain gages were secured with epoxy cement to the aircraft. Multi-pin connectors were placed in the signal cables to allow the aircraft to be disassembled without cutting wires.

Special brackets held the potentiometers for incidence angle, roll angle, elevator angle, and aileron angle. An 8-foot boom, which protruded beyond the nose of the aircraft, was secured to the left side of the aircraft platform. The boom was used for static and pitot pressure to record airspeed as well as to support the wind vanes and the potentiometers for angle of attack and sideslip. Special strain gage force rings measured pitch and roll cable loads.

Three linear accelerometers were installed on a special mounting and were located at the C.G. of the aircraft. These accelerometers measured vertical, lateral, and longitudinal forces on the aircraft during flight as well as during take-off and landing operations.

The vertical gyro was self-erecting, with potentiometer output from the pitch and roll gimbals. The gyro was carefully aligned with the longitudinal axis of the aircraft to eliminate cross-coupling effects with the roll channel when pitch was introduced in flight.

INSTRUMENTATION SYSTEM ACCURACY

Accuracy of the instrumentation system is a function of the calibration of each transducer; it is also a function of the accuracy of the signal voltage. Each transducer was calibrated by direct loading or by bridge resistance substitution. The deflection of the signal at each recorder was adjusted to give a voltage excursion which established the record readability found in Table 2.

TABLE 2
MEASUREMENTS, RECORDING METHODS, AND RECORDING READABILITY

Measurement	Transducer	Recording Method	Record Readability
Bend. Keel, Pivot	Strain Gage	Oscillograph	<340 psi
Shear Keel, Aft Pivot	Strain Gage	Oscillograph	<340 psi
Bend. Keel, Vert Apex	Strain Gage	Oscillograph	<340 psi
Comp/Ten. Keel, Apex	Strain Gage	Oscillograph	<340 psi
Shear Keel, Fwd Pivot	Strain Gage	Oscillograph	<340 psi
Bend. Lead. Edge, Pivot	Strain Gage	Oscillograph	<270 psi
Compression/Tension Lead. Edge, Pivot	Strain Gage	Oscillograph	<270 psi
Comp/Ten. Spread. Bar, Horiz.	Strain Gage	Oscillograph	<100 psi
Comp/Ten. Spread. Bar, Diag.	Strain Gage	Oscillograph	<100 psi
Comp/Ten. Cent. Strut	Strain Gage	Oscillograph	<240 psi
Comp/Ten. Fwd "V" (R)	Strain Gage	Oscillograph	<240 psi
Comp/Ten. Fwd "V" (L)	Strain Gage	Oscillograph	<240 psi
Comp/Ten. Aft "V" (R)	Strain Gage	Oscillograph	<240 psi
Comp/Ten. Aft "V" (L)	Strain Gage	Oscillograph	<240 psi
Angle Wing, Pitch	Potentiometer	Oscillograph	.10 deg.
Angle Wing, Roll	Potentiometer	Oscillograph	.80 deg.
Angle Wing, Tip	Potentiometer	Oscillograph	.32 deg.
Angle Ruddervator	Potentiometer	Oscillograph	.23 deg.
Angle Attack	Potentiometer	Oscillograph	.28 deg.
Angle Sideslip	Potentiometer	Oscillograph	.28 deg.
Angle Roll, Free Space	Potentiometer	Oscillograph	<1.0 deg.
Angle Pitch, Free Space	Potentiometer	Oscillograph	<1.0 deg.
Acceleration Platform "X"	Strain Gage	Oscillograph	.02 g

TABLE 2 (Continued)
MEASUREMENTS, RECORDING METHODS, AND RECORDING READABILITY

Measurement	Transducer	Recording Method	Record Readability
Acceleration Platform "Y"	Strain Gage	Oscillograph	.02 g
Acceleration Platform "Z"	Strain Gage	Oscillograph	.04 g
Vibration, Tail Surface	Strain Gage	Oscillograph	.02 g
Load Pitch Cable	Strain Gage	Oscillograph	<1.0 lb.
Load Roll Cable (R)	Strain Gage	Oscillograph	<1.0 lb.
Load Roll Cable (L)	Strain Gage	Oscillograph	<1.0 lb.
Flow Fuel Line	Freq. Meter	Oscillograph	<.285 gpm
Pressure, Oil	Panel Inst.	Pilot	<2 lb.
Temp, Oil	Panel Inst.	Pilot	<5 deg.
Temp, Outside Air	Thermo.	Pilot	<1.0 deg.
Temp, Cylinder Head	Panel Inst.	Pilot	<5 deg.
Pressure, Altitude	Panel Inst.	Pilot	<25 ft.
Pressure, Airspeed	Panel Inst.	Oscillo & Pilot	<.5 mph
RPM, Engine Speed	Panel Inst.	Pilot	<50 rpm
Rate, Rate of Climb	Panel Inst.	Pilot	<100 fpm
Pressure, Manifold	Panel Inst.	Pilot	<.1 in.hg.
Force, Roll Control	Strain Gage	Oscillograph	<1.0 lb.
Force, Pitch Control	Strain Gage	Oscillograph	<.5 lb.
Force, Pitch Trim	Strain Gage	Oscillograph	<1.0 lb.
Position, Roll Control	Dial	Pilot	<2.0 deg.
Position, Pitch Control	Dial	Pilot	<2.0 deg.

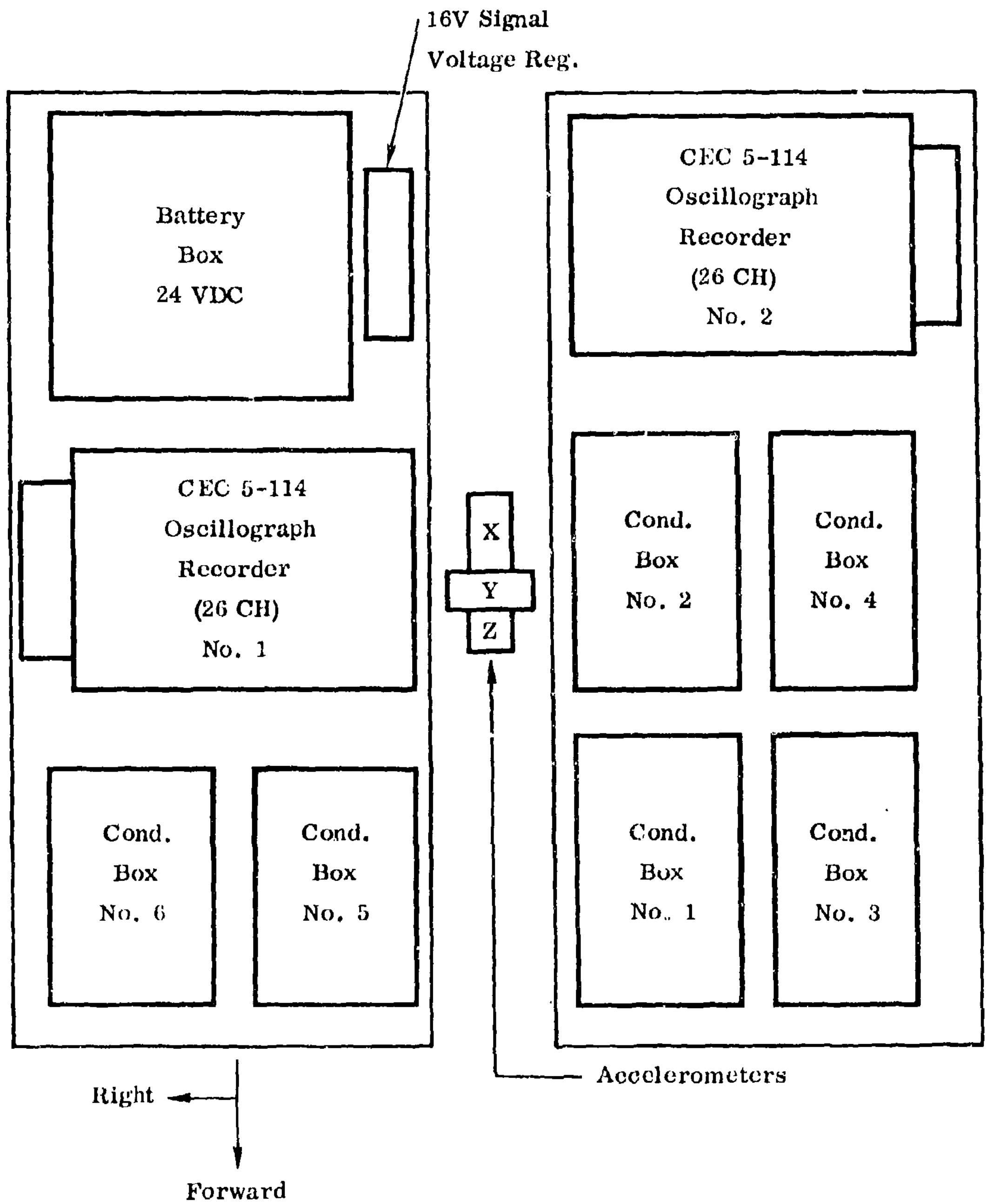
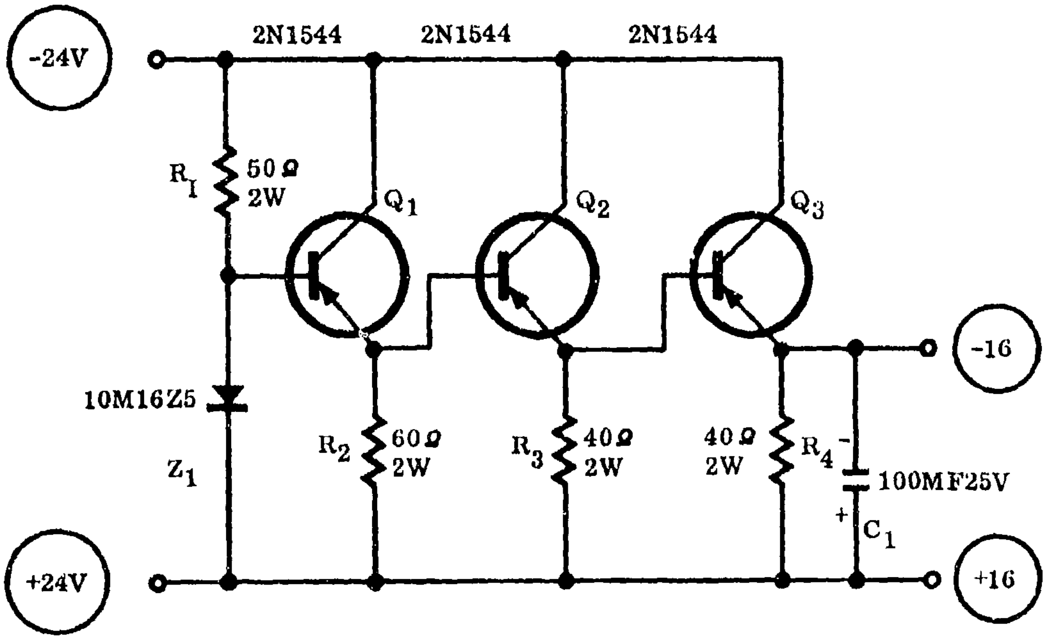


Figure 2. Recording Equipment



Note: R_2 may be divided to change output voltage.

Figure 3. Strain Gage Signal Regulator

TEST PROCEDURES

The flight test procedures used throughout the flight test program were mostly standard and are applicable to all low-speed, lightweight aircraft. On-board oscillographs recorded stability and control data including loads, forces, deflections, etc. The majority of performance data was obtained from pilot-observed records.

Due to the small speed range of the airplane (10 to 15 miles per hour), normal data scatter would often mask an attempted 2- to 4-miles-per-hour airspeed change. Consequently, test results reflect fewer data points throughout the speed range than are normally obtained when testing a more conventional type aircraft with a larger speed envelope.

Weight and balance checks were made by weighing the aircraft on three platform scales. The center of gravity was controlled by shifting the instrumentation pallet fore and aft on the cargo platform.

TEST RESULTS

PERFORMANCE

Airspeed Calibration

Airspeed position error correction was obtained by using the ground speed course method. Constant speed runs were made over a known course length, and true speed was obtained from time and distance data. Calibrated airspeed obtained from true airspeed was compared to the airspeed indicated by the aircraft pitot-static system, thereby obtaining the airspeed position error correction. From this data, the static pressure error was obtained which was used to determine the corresponding altimeter position error. Figures 4 and 5 show the airspeed and altimeter position error corrections respectively.

An attempt was made to verify this data by using the altimeter depression or tower fly-by method for determining the static pressure error. Since this method lends itself more favorably to high speed aircraft, poor correlation and an excessive amount of data scatter were obtained. Consequently, the position error correction curves as presented reflect only the results of the ground speed course method.

XV-8A PERFORMANCE SUMMARY

S. L. Std. Cond. - Design Gross Wt.

	Calculated	Actual
T.O. Ground	240 ft	570 ft
Ldg Ground Roll (No Wind)	120 ft	120 ft
R/C (Low Cargo)	250 ft/min	550 ft/min
(High Cargo)		425 ft/min
R/S (Idle Power)		890 ft/min
S/C		7800 ft
Max. Range Speed	55 MPH	57 MPH
Range (10% Reserve)	115 stat mi	108 stat mi (3000 ft alt)
Max. Speed (BHP Avail. Extrapolated)	64 MPH	70 MPH
(Aileron Oscillation Limit)		63 MPH
Stall Speed (Nom. C.G.)	40 MPH	47 MPH
Endurance		2.1 hrs
Pay Load	1000 lb	850 lb
Specific Range	.82 stat mi/lb	.85 stat mi/lb (3000 ft alt)
L/D	3.9	4.7

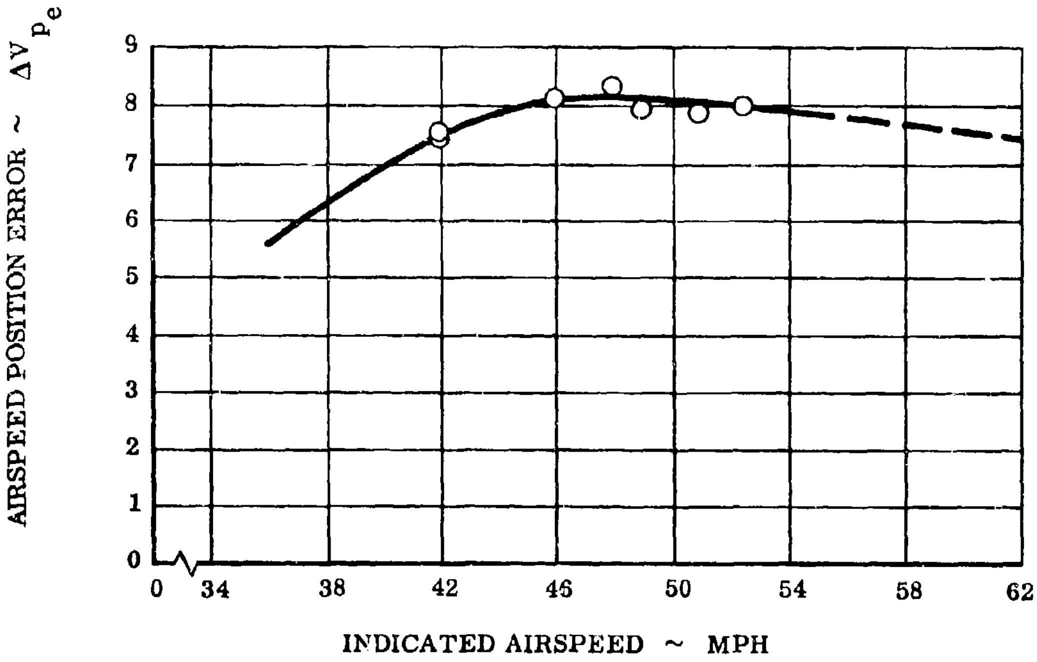


Figure 4. Airspeed Position Error Correction

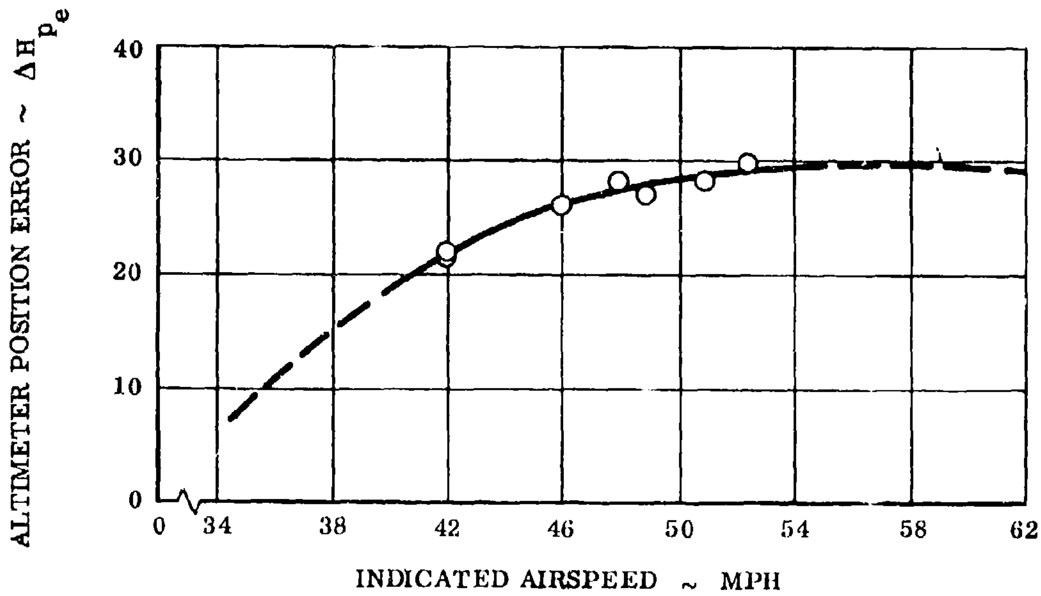


Figure 5. Altimeter Position Error Correction

Take-Off and Landing Performance

The XV-8A take-off and landing performance tests were conducted at two gross weights: 2300 pounds and 2000 pounds. A Model IV Fairchild Flight Analyzer was used as the principal source of data acquisition for the establishment of take-off and landing distances over a nominal 50-foot obstacle.

All testing was conducted from Runway 35 at the Yuma Proving Ground. Balloons indicated the 50-foot obstacle height to the pilot. The Fairchild Flight Analyzer was set up on a concrete aircraft parking pad at an offset distance of 1300 feet from the centerline of Runway 17-35.

A total of seven take-offs were made, three at heavy gross weight and four at light gross weight. A total of eleven landings were photographed, three at heavy gross weight and eight at light gross weight. In all cases, the air was calm and the ambient pressure and temperature produced a resulting density altitude close to standard sea level conditions. This was considered to be sea level standard, without the need for application of corrections to standard conditions.

Plotted time histories of the take-off and landing flight paths were made from the Fairchild Flight Analyzer records. The ground run and air distances were determined for a nominal 50-foot obstacle clearance.

Tabular data for take-off and landing performances are presented in Tables 3 and 4 respectively. A single summary presentation of take-off and landing performance is presented in Figure 6.

Take-off distance required for lift-off and for clearance of a 50-foot obstacle is presented as a function of aircraft gross weight at a take-off power of 2800 rpm. The resultant speed at a 50-foot altitude is 60 miles per hour, based on Analyzer data, and the rate of climb is 900 feet per minute. The distances shown in Figure 6 are for zero wind and standard sea level conditions.

Total landing distance required to clear a 50-foot obstacle and ground roll distances are presented as a function of engine rpm, over a gross weight range of 2000 to 2300 pounds at a wing incidence angle of 23 degrees. The approach speed at the 50-foot obstacle is 57 mph, based on Analyzer data, and the accompanying rate of descent is 900 feet per minute.

The distances determined from the Flight Analyzer data are correct. The camera timing indicator, and consequently speed, is believed to be in error by approximately 10 percent. Difficulty was experienced in regulating the voltage of the timer system on the Analyzer. A higher voltage had to be applied for timer operation, thereby increasing the speed of the timing system. Comparison of these speeds with observed and recorded airspeed data shows the speeds high by 10 percent. Consequently, rates of climb and descent would also be in error; but in all cases, the numbers for each run are relative.

Figures 7 through 10 show typical Fairchild Analyzer photographs of take-off and landing obtained during test.

The best technique for maximum performance take-off and landing, as determined during the performance tests, is as follows: Maximum engine rpm with brakes held firmly; stick neutral during acceleration to 35 miles per hour indicated; brisk rotation at 35 miles per hour by pulling stick one-half to three-fourths back. As the aircraft rotates, airspeed rapidly increases to 40 to 42 mph indicated. With rapid additional airspeed, it increases to about 50 as the stick is returned to neutral. Aircraft then trims out to $F_e = 0$ climb speed of about 47 to 49 mph indicated.

Landing approach is made at idle rpm which will produce a $F_e = 0$ airspeed of about 42 miles per hour indicated. The stick is eased full forward to gain 4 to 5 miles per hour airspeed at about 100 feet above the ground in order to provide enough elevator control power for flare. Full-back stick is briskly applied just before ground contact, at about 10 feet altitude. Average time from stick pull to touchdown is 3.5 seconds, with a high rate of attitude change and main-gear-only contact. The stick should be held back and full brakes applied. It is possible to scrape the ground with the elevator if the stick is not held back, especially if a complete stall with pitch-up is induced. After some practice, it is possible to stop in less than 30 feet using such a landing technique.

TABLE 3 SUMMARY TAKE-OFF DATA								
Run No.	Gr. Wt. (Lb.)	i_w (Deg.)	γ 50' (Deg.)	ΣS_H (50 Ft.)	S_H (Gnd Ft.)	S_H (Air Ft.)	R/C 50' (Ft/Sec)	V_H 50' (MPH)
38-1	2300	23	8	1050	570	480	-	-
38-3	2300	23	7.5	1015	550	465	15	60
38-5	2300	23	7	1040	575	465	13	60
39-1	2000	23	9	870	400	470	15	57
39-5	2000	23	9	815	370	445	13	61
39-7	2000	24	9.5	770	335	435	15	60
39-9	2000	24	8	770	330	440	12	60

TABLE 4
SUMMARY LANDING DATA

Run No.	Gr. Wt. (Lb.)	RPM	γ 50' (Deg.)	L/D 50'	ΣS_H (50 Ft.)	S_H (Gnd Ft.)	S_H (Air Ft.)	R/S 50' (Ft/Sec)	V_H 50' (MPH)
38-2	2300	1500	11	5.15	555	215	340	16	58
38-4	2300	1400	12	4.70	555	260	295	11	58
38-6	2300	1300	10	5.67	590	265	325	14	56
39-2	2000	1600	10	5.67	625	240	385	15	58
39-4	2000	1500	10	5.67	575	220	355	13	53
39-6	2000	1400	11.5	4.92	585	265	320	16	57
39-8	2000	1300	12.5	4.51	580	260	320	14	58
39-10	2000	1100	12	4.70	455	165	290	16	55
39-12	2000	1000	14	4.01	520	190	330	17	57
39-14	2000	800	14	4.01	350	60	290	22	55
39-16	2000	800	13	4.33	435	160	275	16	55

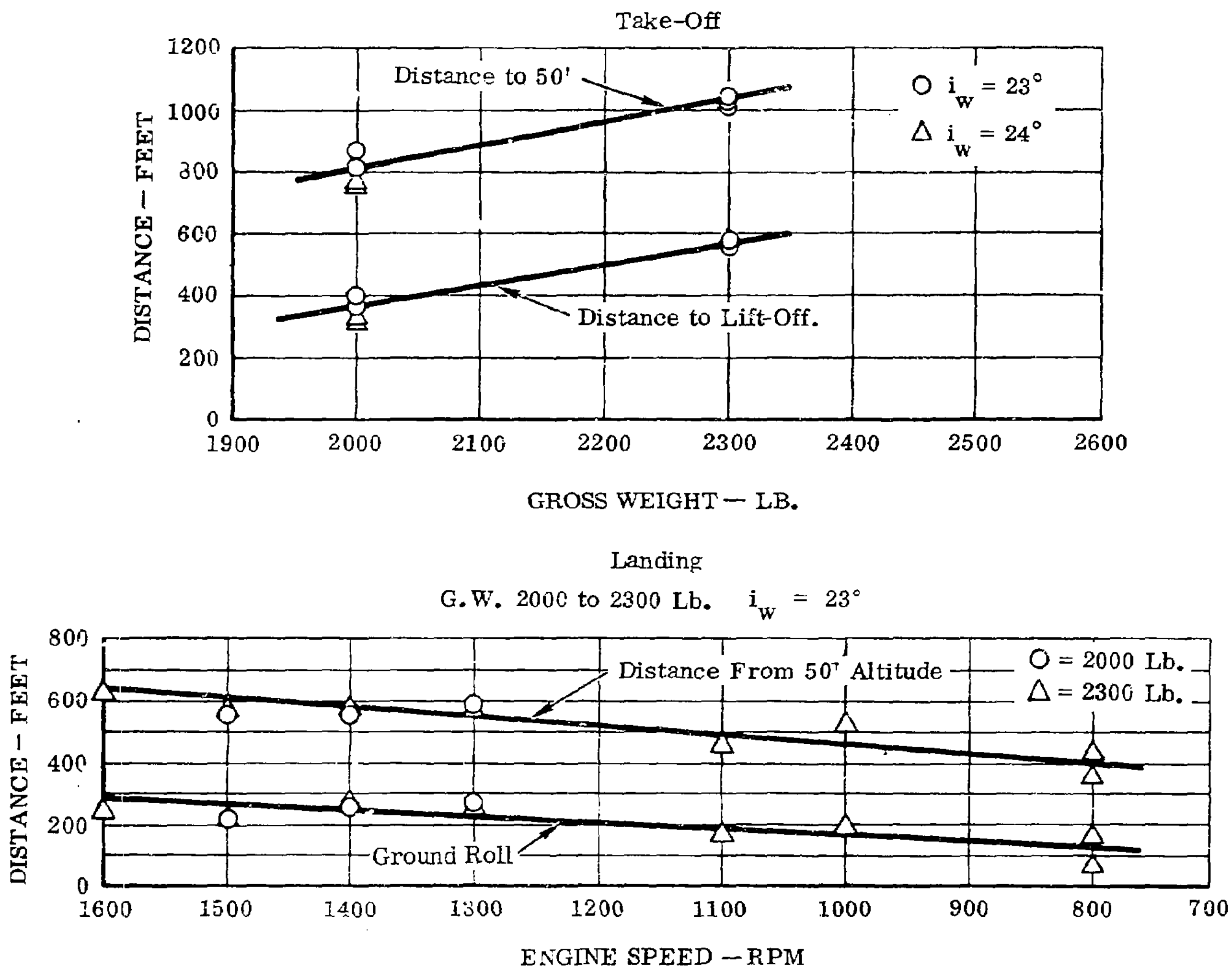


Figure 6 Take-Off and Landing Performance - Zero Wind

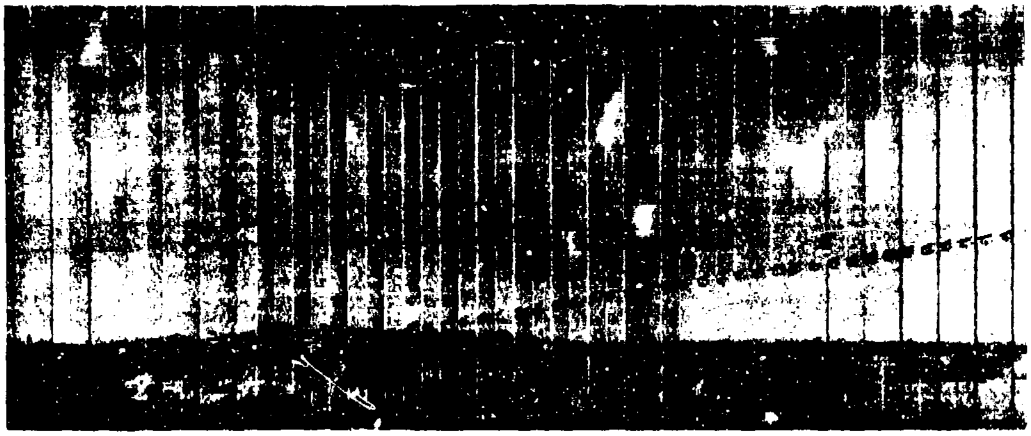


Figure 7. Take-off Flight Profile - 2000-Pound Gross Weight

Figure 8. Landing Flight Profile - 2000-Pound Gross Weight

Figure 9. Take-off Flight Profile - 2300-Pound Gross Weight

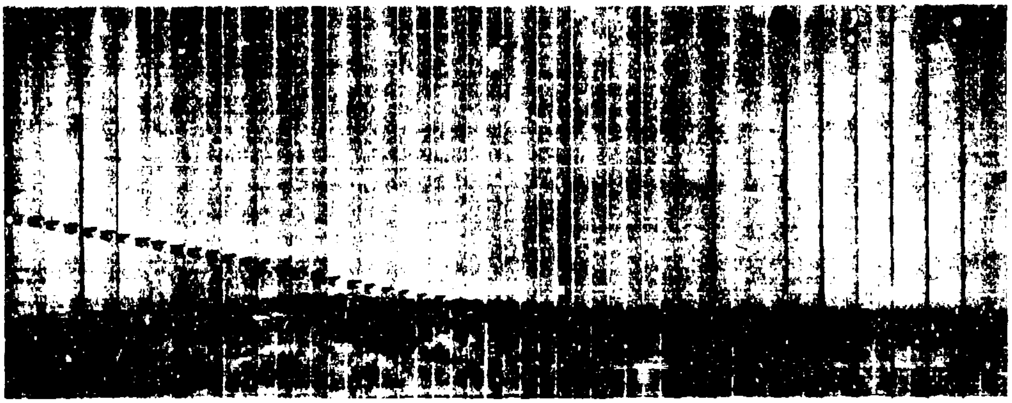


Figure 10. Landing Flight Profile - 2300-Pound Gross Weight

Climb Performance

Rate of climb data was obtained by making sawtooth climbs through a 1000-foot test altitude band. In addition, check climbs were made to verify climb schedules. The majority of climbs were made at the trim climb speed, zero stick force, for the wing incidence setting being tested.

Figures 11 and 12 show summary rate of climb data for 2000 pounds and 2300 pounds gross weight respectively. Rates of climb and the corresponding climb speed schedules are shown for both 22- and 23-degree wing incidence settings. All test climbs were made at maximum power settings with the mixture set at full rich. As shown in Figures 11 and 12, 1- to 2-mile-per-hour change in climb speed results in a 25- to 50-feet-per-minute change in rate of climb. At 2300 pounds, the sea level rate of climb exceeds the original estimate by 200 feet per minute.

A climb to maximum altitude was made at 2000 pounds take-off gross weight to determine service and absolute ceiling. The service ceiling, 100-feet-per-minute rate of climb, was 9350 feet density altitude. The absolute ceiling attained was 9900 feet density altitude. The time needed to climb to maximum altitude was 30 minutes 12 seconds.

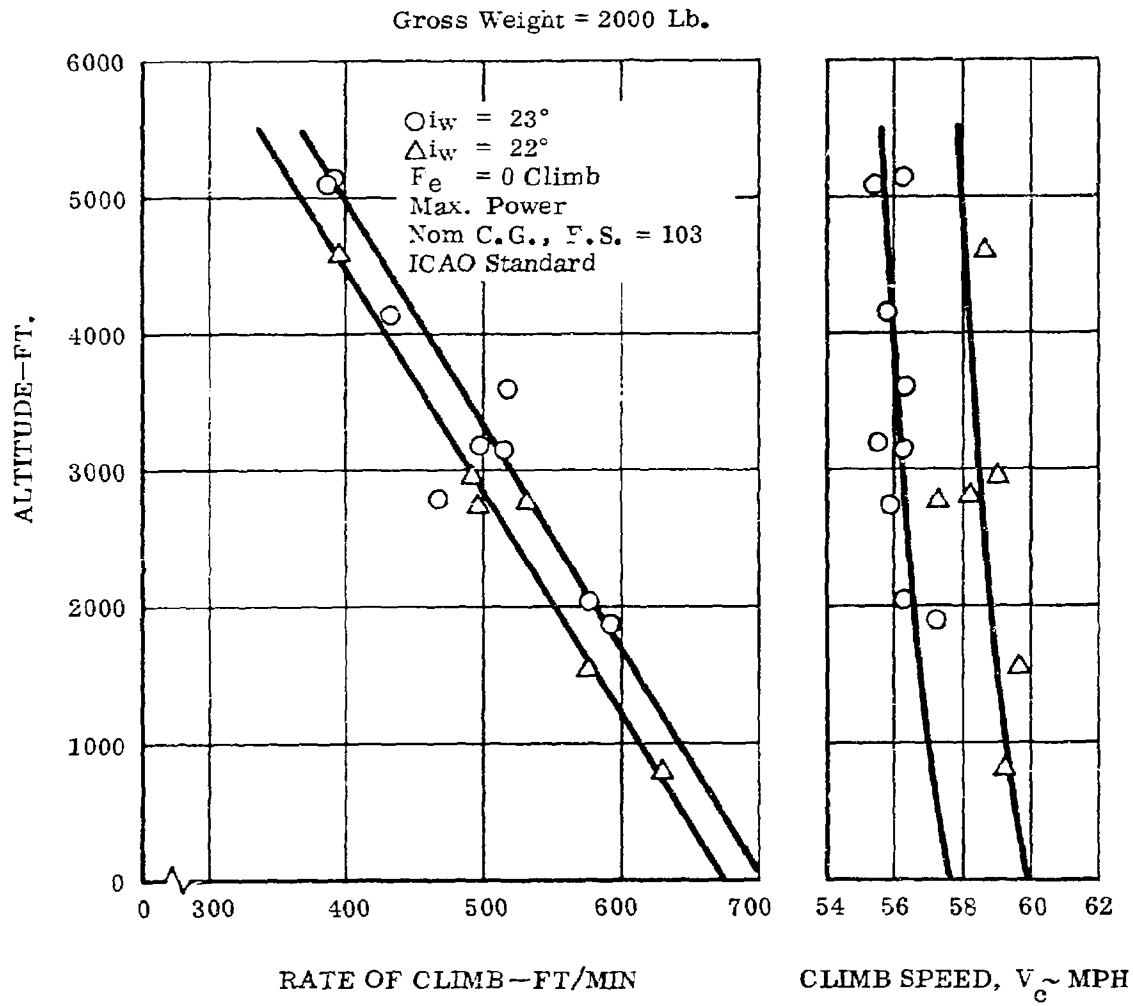


Figure 11. Rate of Climb Summary - 2000-Pound Gross Weight

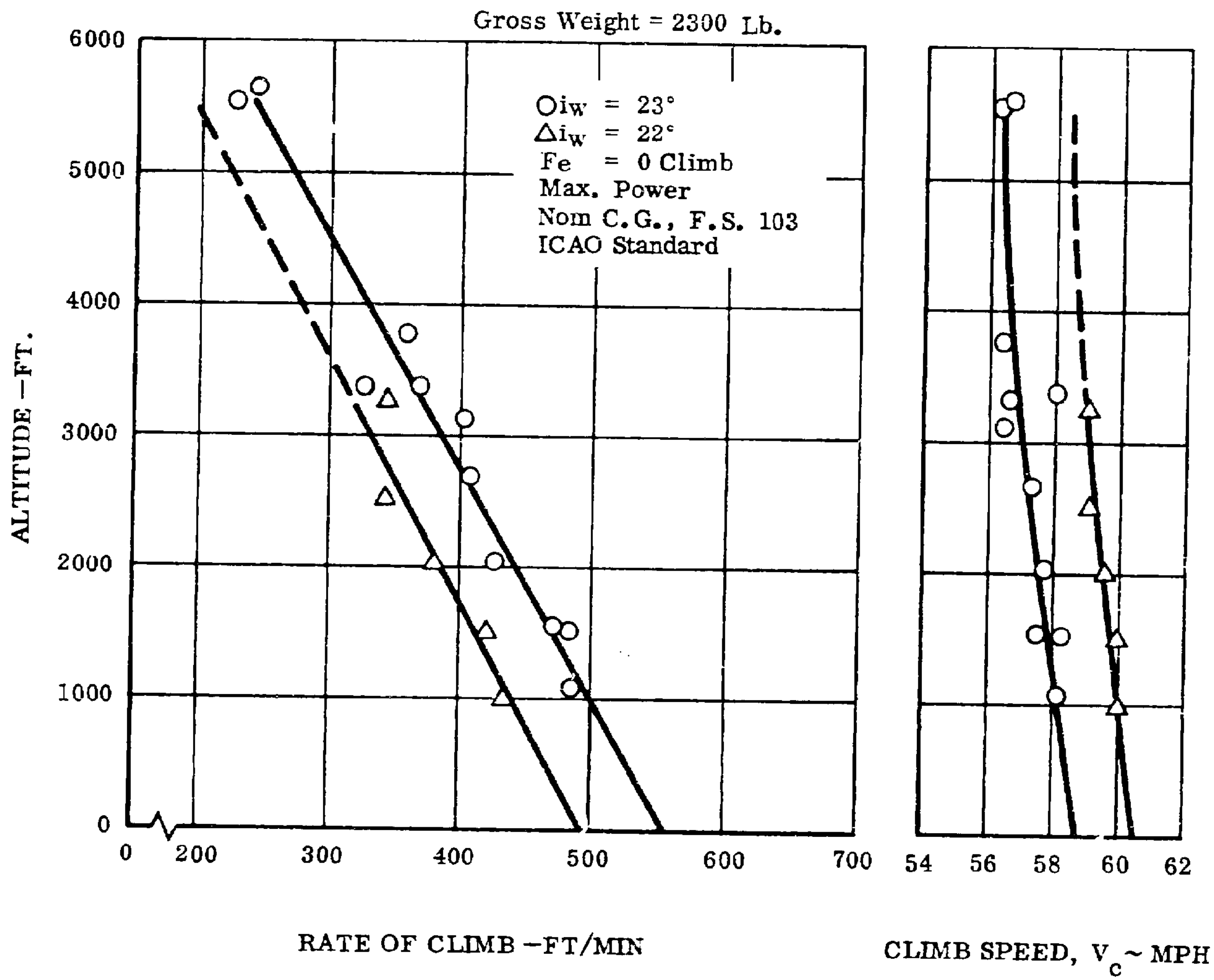


Figure 12. Rate of Climb Summary - 2300-Pound Gross Weight

Descent Performance

Considerable effort was expended in determining the descent characteristics of the aircraft. Throughout the course of the program, a wide variation in rates of descent was observed for the same test configuration, i.e., weight, C.G., wing incidence setting, and speed. Air turbulence is a major contributing factor to the variation in rate of descent. It is characteristic of this airplane to rock or roll laterally in turbulent air. This is due to the light wing loading and also to a pendulum effect caused by the fuselage and center of gravity being well below the wing. The wing, when rolled, will spill some lift and thereby will increase the rate of descent of the aircraft.

From the data obtained, the effects of speed, weight, wing incidence setting, and altitude produce minimum changes in rates of descent. The governing criterion for rate of descent is the rpm setting of the engine. Figure 13 shows rates of descents as a function of engine rpm.

The recommended descent procedure is to descend at the cruise wing incidence setting and to adjust the rate of descent with power. The most comfortable and practicable descent is with power set at 1600-1700 rpm, which is also the normal landing approach power setting.

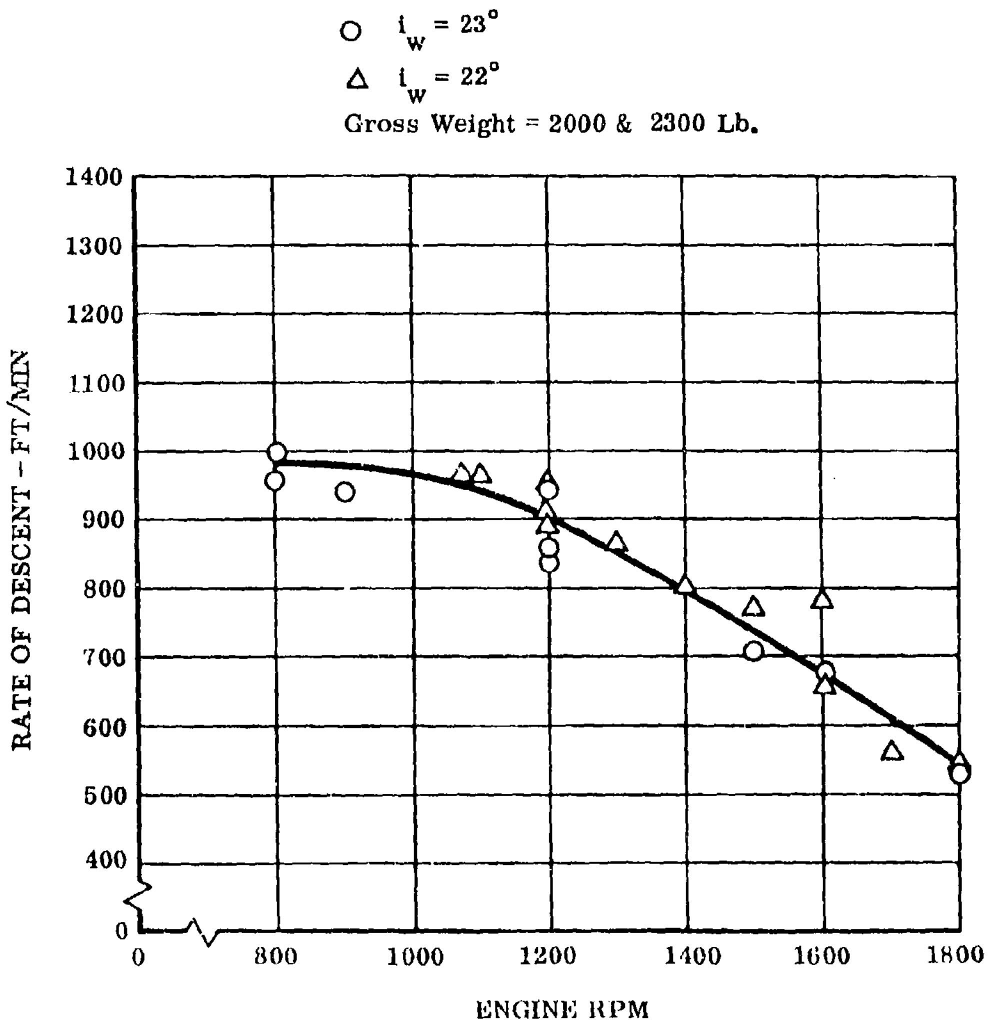


Figure 13. Average Rate of Descent vs. Engine RPM

Level Flight Performance

Level flight speed power data were taken at 3000 and 5000 feet pressure altitudes and at 2000 and 2300 pounds gross weight. A generalized power required curve as a function of velocity is presented in Figure 14. The associated generalized rpm vs. power required curve is presented in Figure 15.

All data are presented on an equivalent weight basis and reduced to sea level standard day. To obtain data for altitudes other than sea level and weights other than standard, the following relationships must be used:

$$V = \frac{V_{ew}}{\sigma\, 1/2} \times \left(\frac{W}{W_s}\right)^{1/2}$$

$$BHP = \frac{BHP_{ew}}{\sigma\, 1/2} \times \left(\frac{W}{W_s}\right)^{3/2}$$

$$RPM = \frac{RPM_{ew}}{\sigma\, 1/2} \times \left(\frac{W}{W_s}\right)^{1/2}$$

Specific range data were obtained in conjunction with the speed power tests. Figures 16 and 17 show the specific range data for 3000 and 5000 feet respectively, and in each case data for 2000 and 2300 pounds are presented. The maximum endurance and 99 percent maximum range speeds are indicated on each curve. All testing was performed with the engine mixture set at full rich. These data show that the aircraft is capable of performing a 100-mile mission with maximum payload and cruising at 3000 feet.

Take-off gross weight	2300 lb.
Total fuel	28 gal.
10 percent reserves	2.8 gal.
Useable fuel	25.2 gal.
Average R/C to 3000 feet	475 ft/min (Figure 12)

Average climb speed	58 mph (Fig. 12)
Time to climb	6.3 min.
Average climb fuel consumption	.375 gpm
Climb fuel	2.36 gal.
Distance travelled in climb (zero wind)	6 miles
Descent fuel (assumed)	1 gal.
Distance travelled in descent	0
Fuel available for cruise	21.84 gal.
Average cruise specific range	4.65 miles/gallon (Fig. 16)
Cruise distance (zero wind)	102 miles
Total distance travelled	108 miles

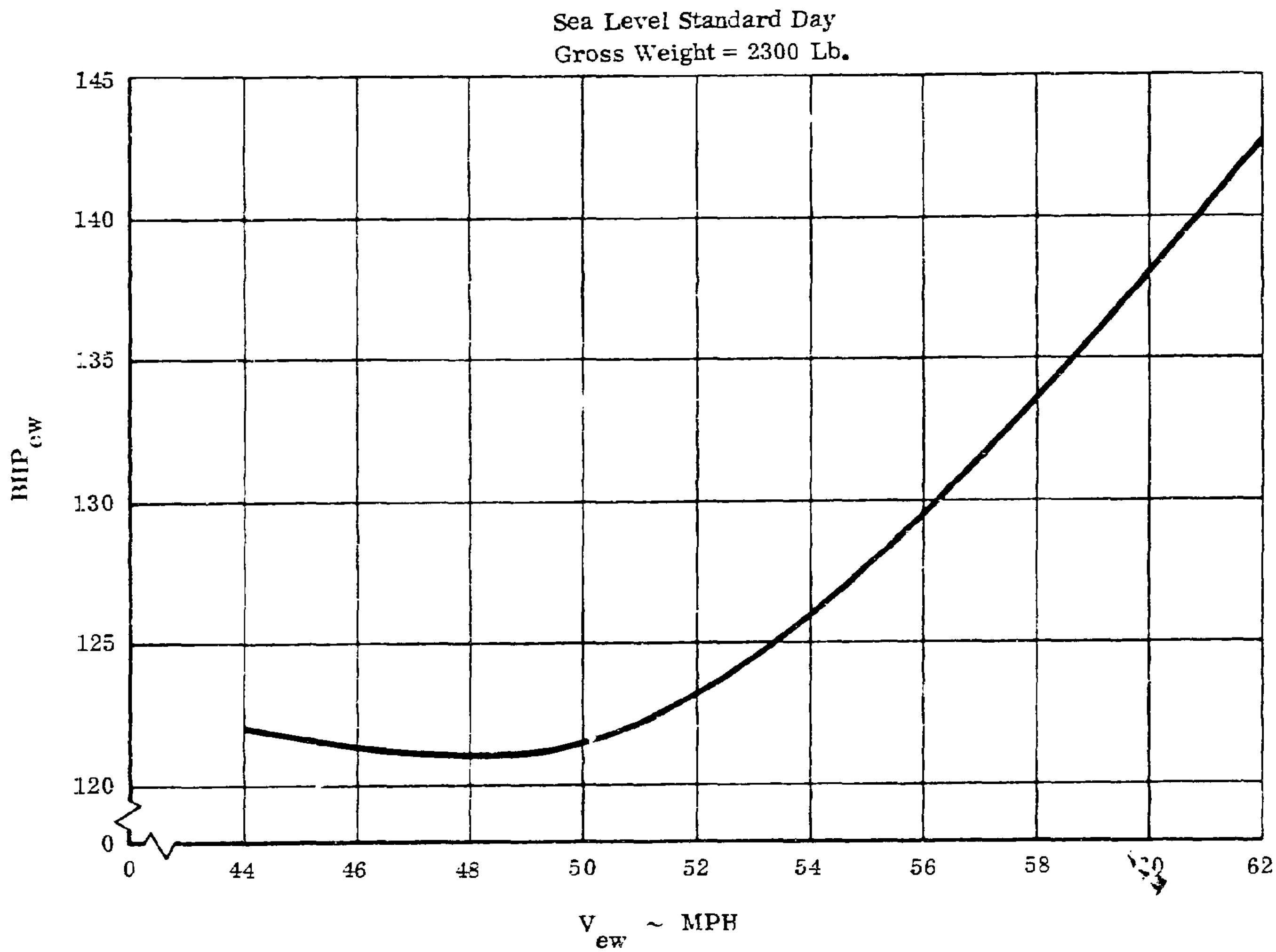


Figure 14. Generalized Power-Required Curve

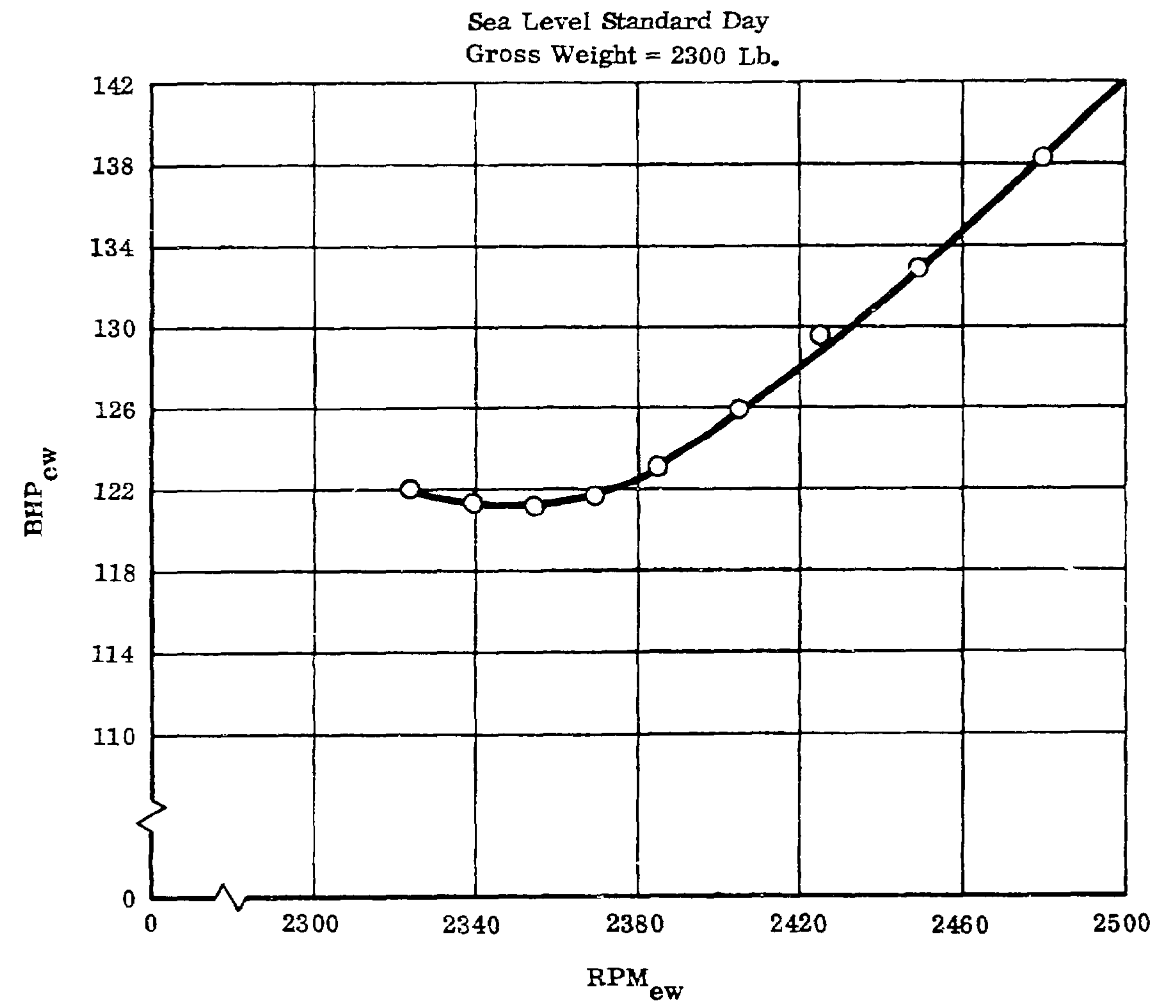


Figure 15. Generalized Engine RPM vs. Power Required

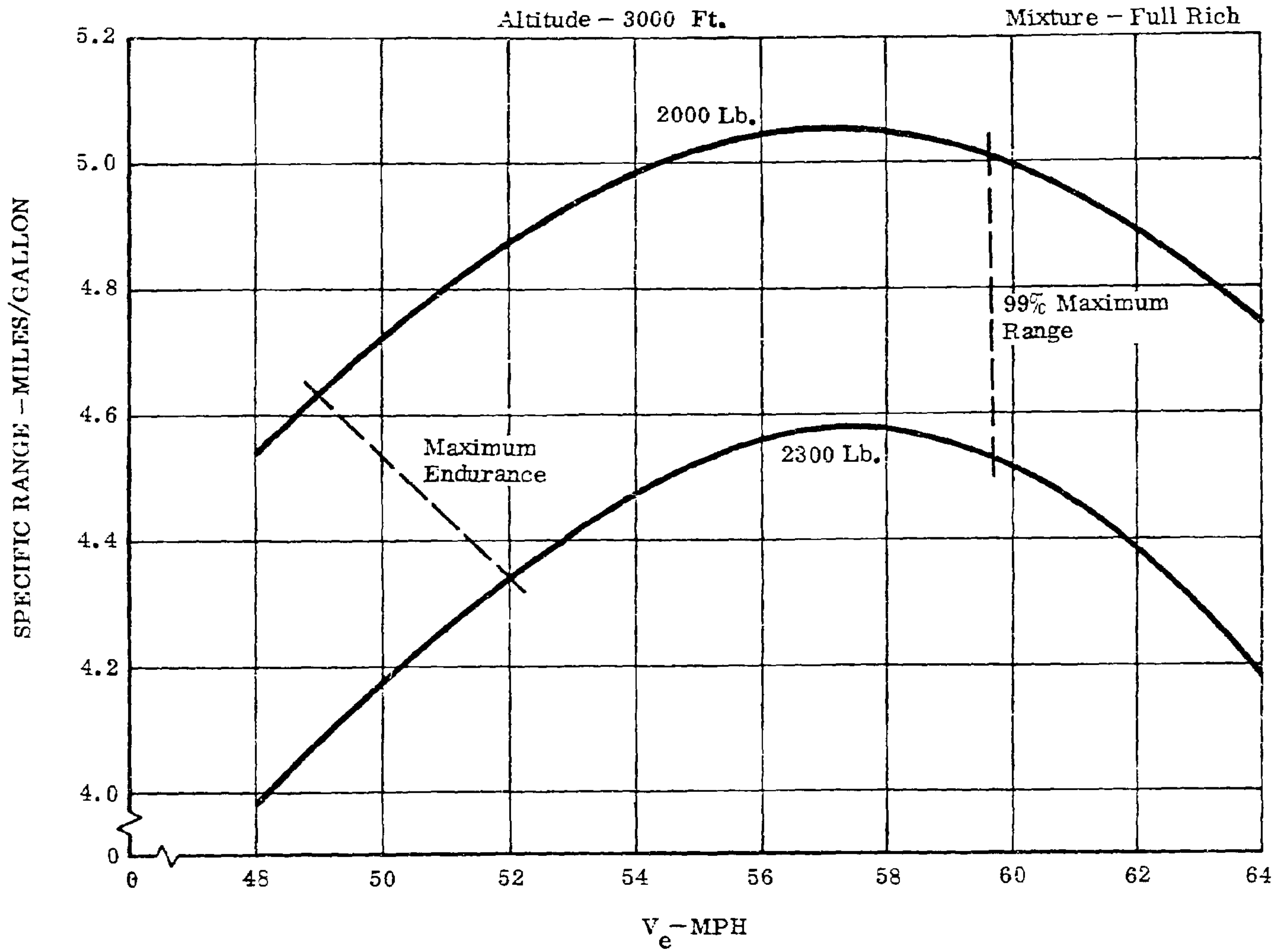


Figure 16. Specific Range vs. Airspeed - 3000-Foot Altitude

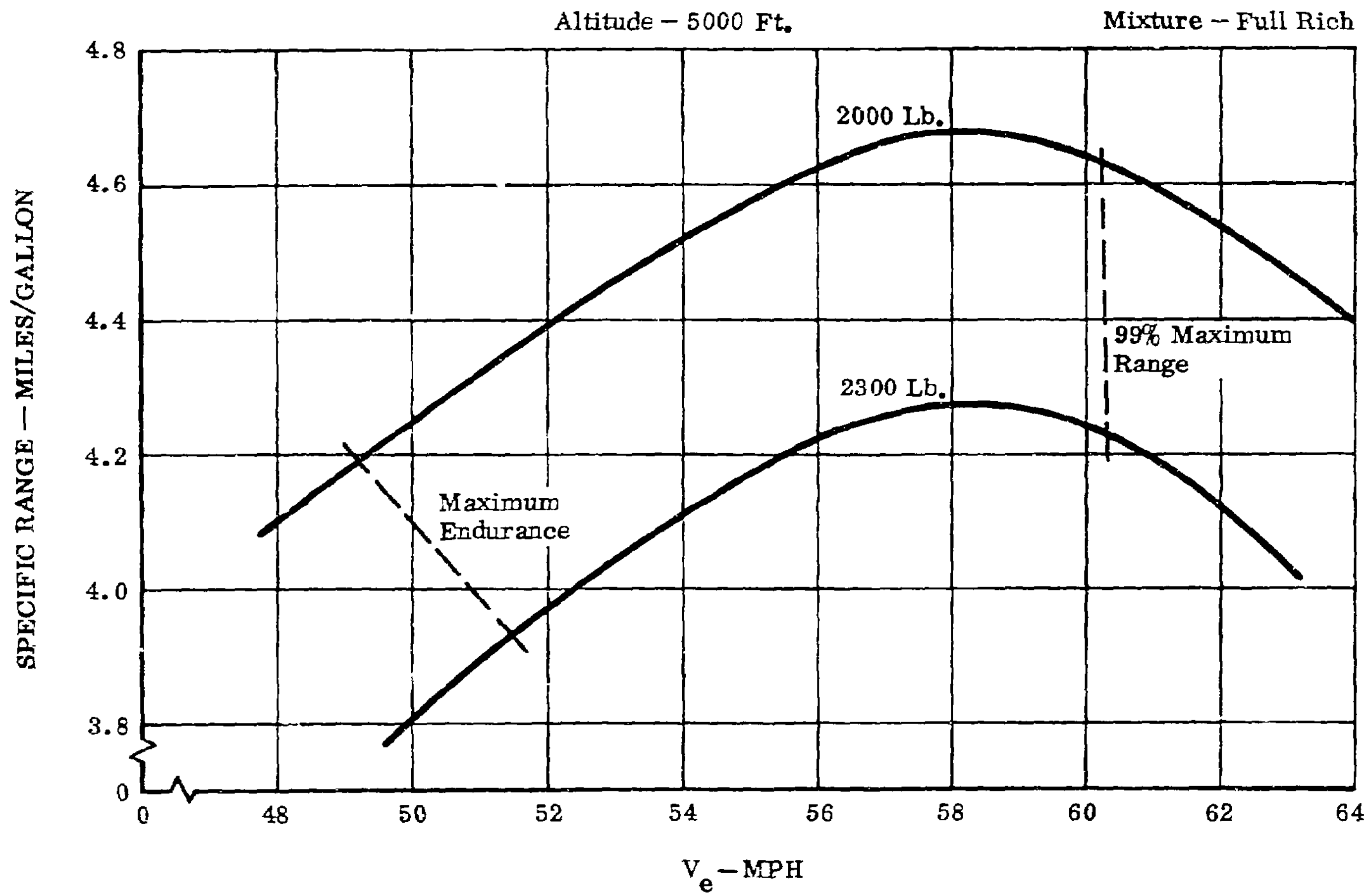


Figure 17. Specific Range vs. Airspeed - 5000-Foot Altitude

Lift And Drag Characteristics

The lift and drag characteristics of the airplane as determined from test are presented in Figures 18 through 20. The lift and drag coefficients (Figure 18) were obtained from level flight speed power data. The data show an improvement over estimated drag of the airplane. Figure 19 shows the associated L/D curve with the comparable improvement in maximum L/D. Lift-to-drag ratios obtained from two idle power glides recorded on the Fairchild Analyzer landing data show L/D's of 4.01 and 4.33. Figure 20 shows the lift coefficient as a function of wing angle of attack.

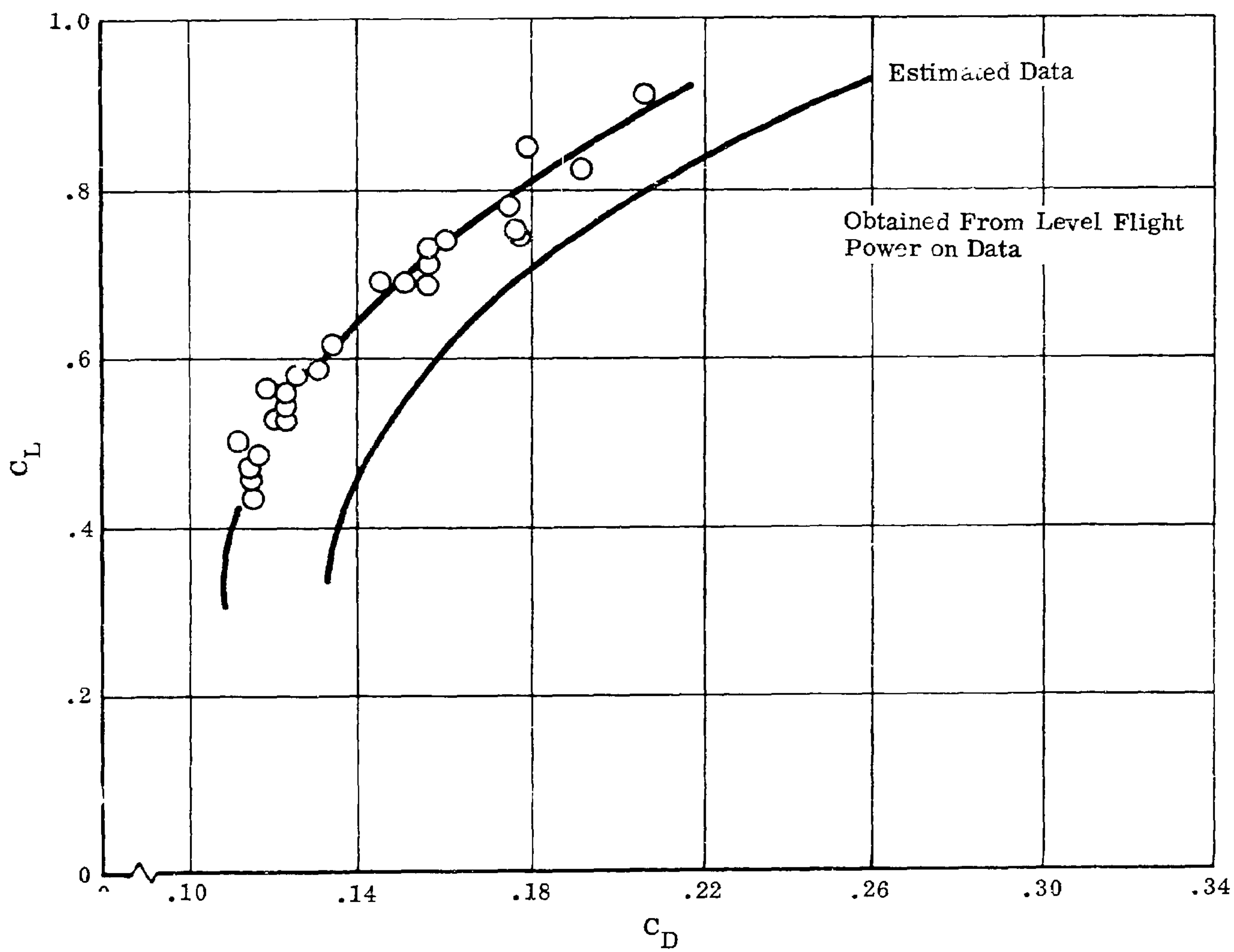


Figure 18. Lift Coefficient vs. Drag Coefficient

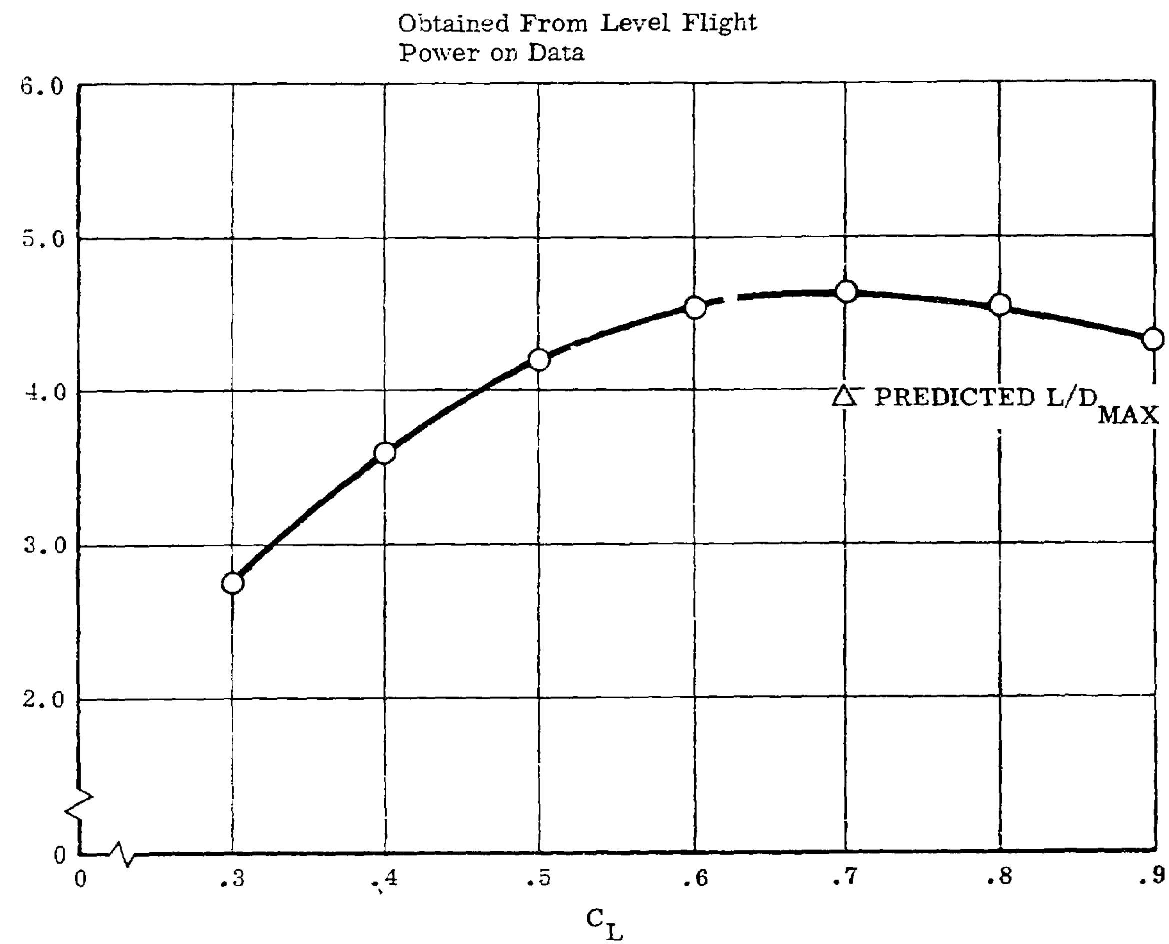


Figure 19. Lift to Drag Ratio vs. Lift Coefficient

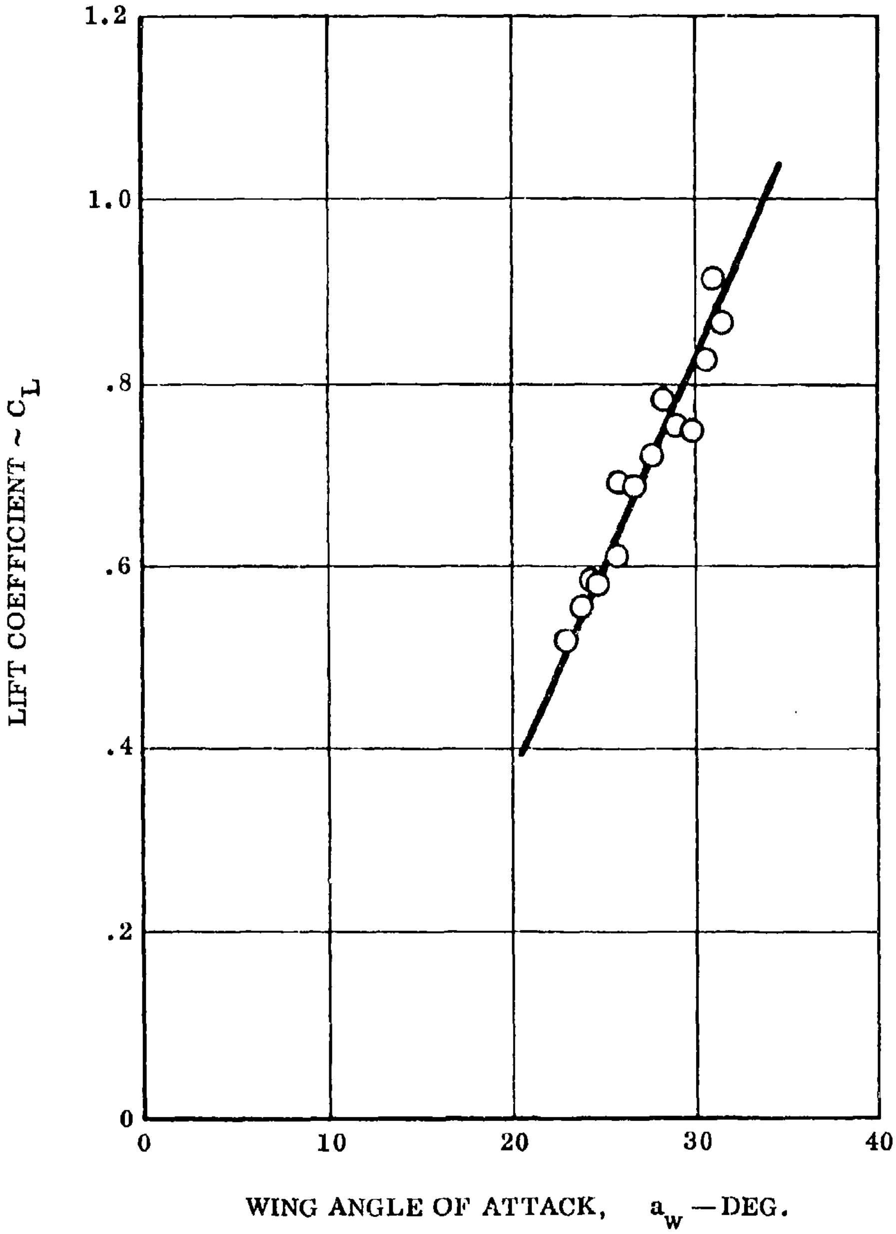


Figure 20. Lift Coefficient vs. Wing Angle of Attack

Operational Flight Envelope

The level flight speed envelopes for a 2300-pound gross weight at forward, nominal, and aft centers of gravity are shown on Figures 21 through 23. Figure 24 shows the envelope for 2000 pounds at nominal center of gravity. All data shown is for 3000 feet pressure altitude. The operational envelope is presented as a function of wing incidence setting, showing the maximum, trim, and minimum speeds attained at each setting.

The level flight V_{max} limit is defined by full-forward stick at speeds below 61 miles per hour. Between speeds of 61 to 62 miles per hour, a low-frequency aileron oscillation is experienced. This oscillation is induced by a travelling wave in the wing fabric. This wave originates near the wing spreader bar and moves aft. As each wave reaches the trailing edge of the wing, the flapping action is transmitted to the ailerons which in turn feed through the control system to the pilot's control wheel. This characteristic is present only at high speed when the wave frequency approaches two to three cycles per second. This phenomenon starts as a random pulse at the control wheel; as speed is increased, it builds up to a steady beat. In all cases, it has been readily discernible by the pilot. This characteristic does not present a serious operational limit to the aircraft. Trim speeds or normal operating speeds are well below V_{max}. Consequently, this oscillation will not be experienced unless a deliberate attempt is made to reach these speeds.

Handling qualities at low speeds (V_{stall} + 2 mph) are normal and are not much different from cruise performance except for the large aft stick displacements and forces. Stalls are difficult to obtain in level flight at nominal and impossible to obtain at forward C.G.; therefore, V_{min} under these conditions is defined by full-aft stick. Low wing loading prevents any significant g force buildup even in maximum pilot effort turns which minimizes the possibility of accelerated stalls. Level flight stalls at nominal to aft C.G. are difficult to obtain and are preceded by good stall warning indications. At higher power settings, torque effect causes right yaw followed by pitch-up resulting in a rolling turn to the right which is easily arrested by forward stick, opposite wheel, and increasing power as required. At reduced rpm, the stall warning consists of a rapid decay of high pull force with pitch-up which is easily arrested by nosing over and adding power as required. If aft stick position is held constant through a complete stall, the aircraft will assume a steep descent angle until air speed build-up increases elevator effectiveness for recovery.

Due to the narrow speed range available, it is practical to set the longitudinal trim prior to take-off for the entire flight. A curve of the optimum wing incidence setting versus C.G. locations is presented in Figure 25.

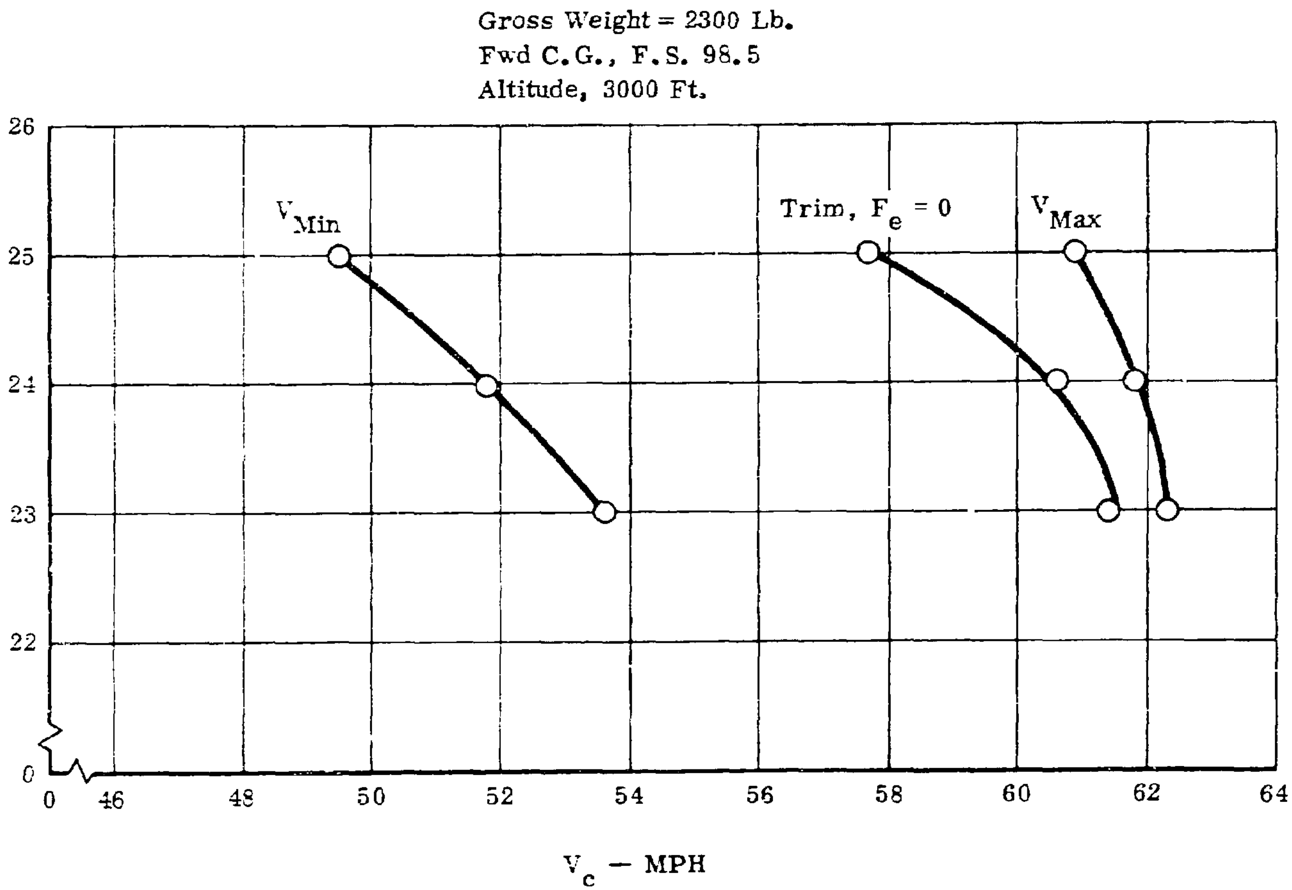


Figure 21. Level Flight Speed Envelope - Forward C.G., 2300 Pounds

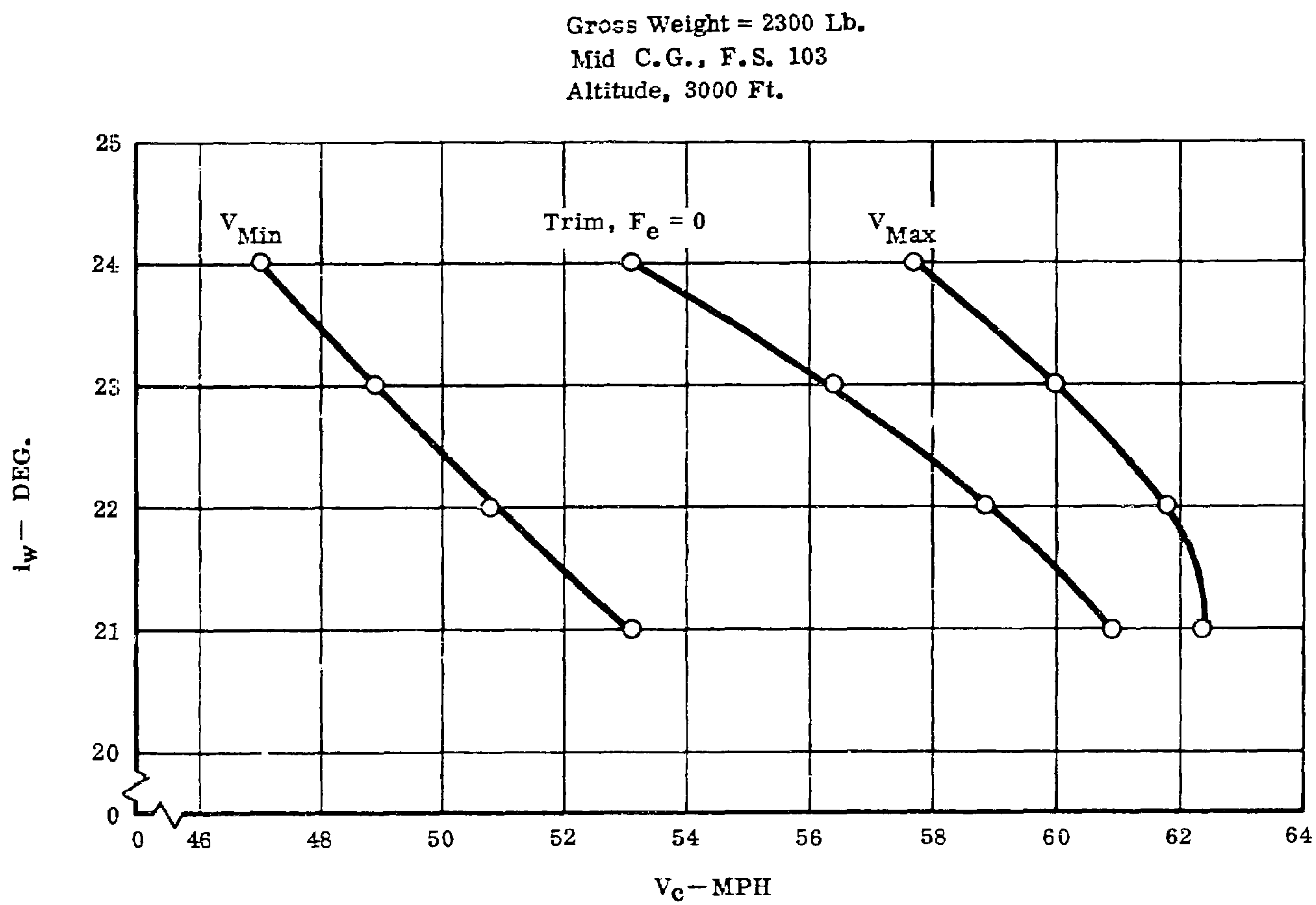


Figure 22. Level Flight Speed Envelope - Mid C.G., F.S. 103

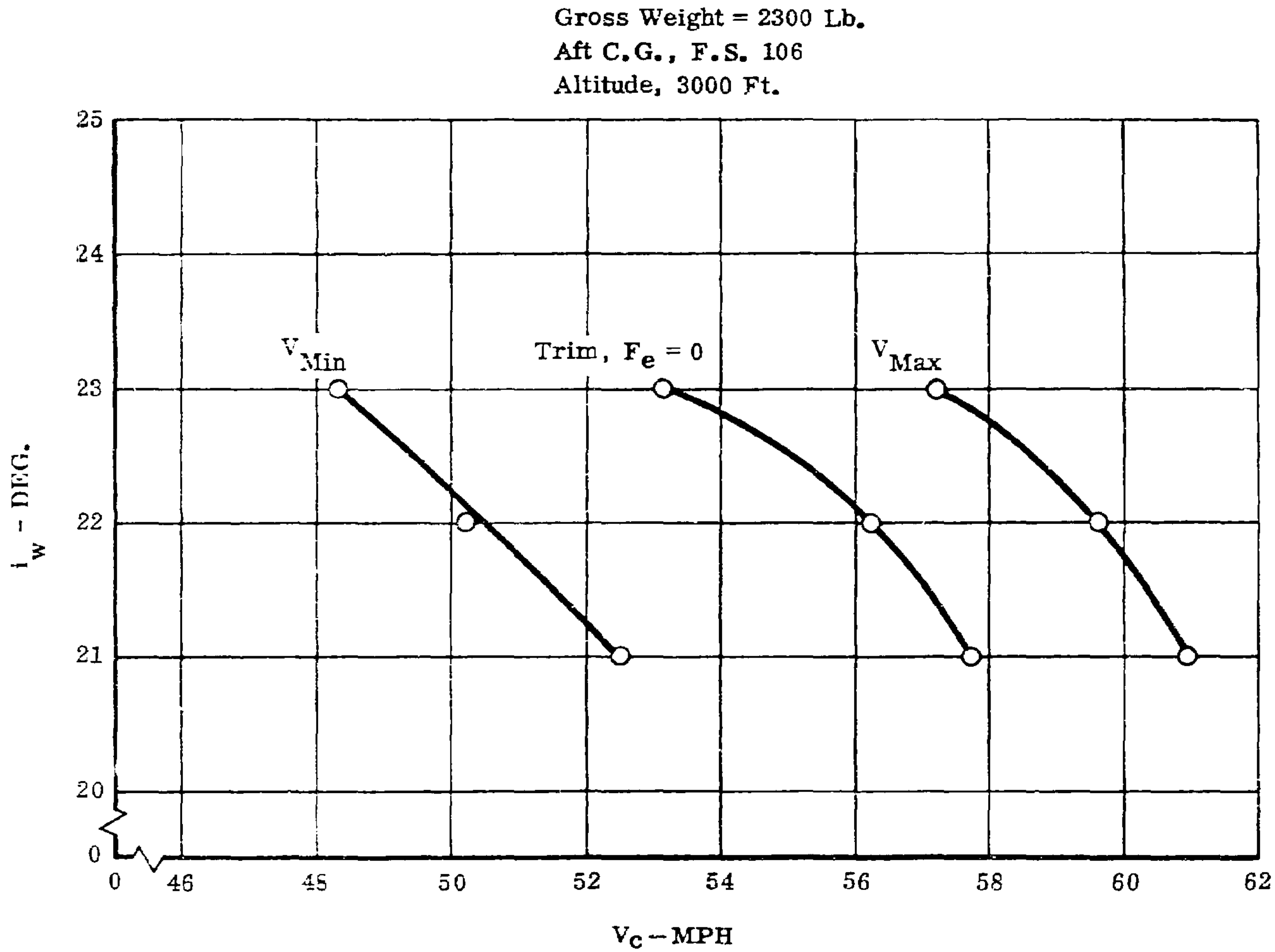


Figure 23. Level Flight Speed Envelope - Aft C.G., 2300 Pounds

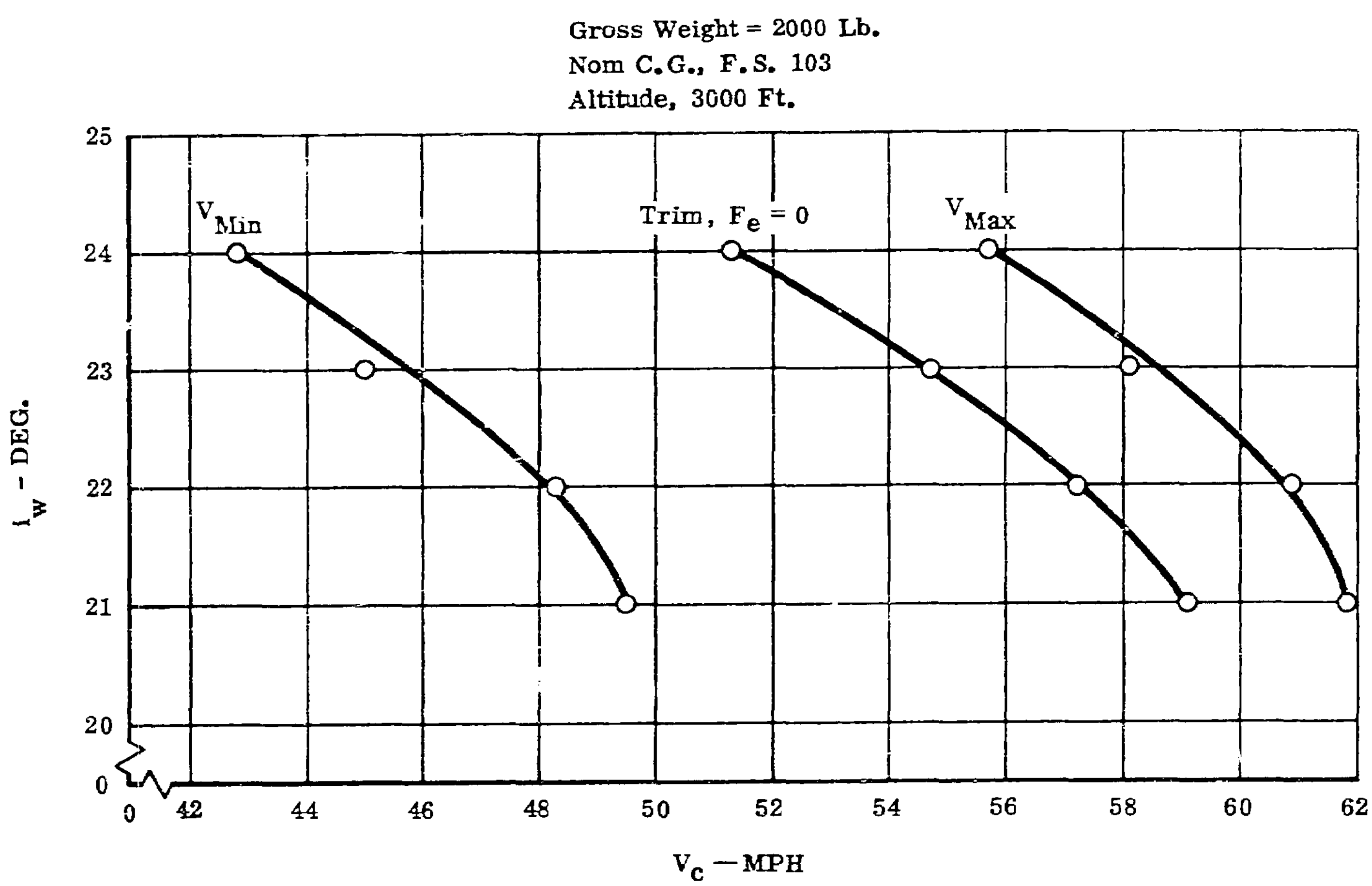


Figure 24. Level Flight Speed Envelope - Nom C.G., 2000 Pounds

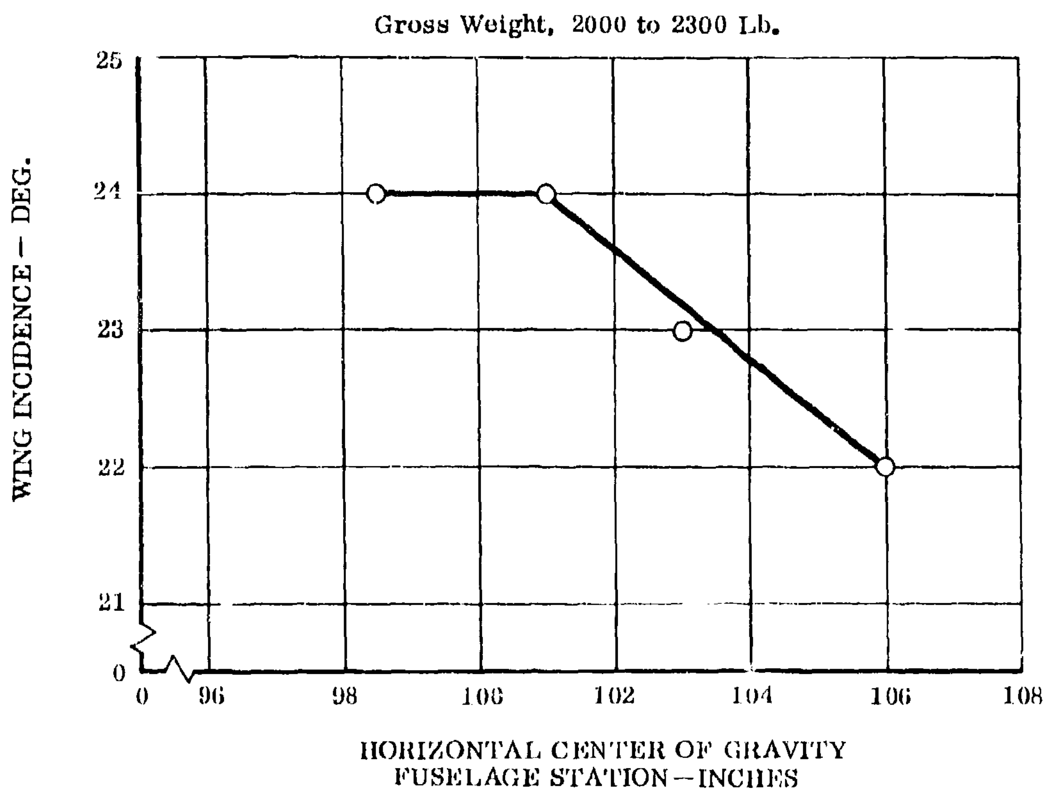


Figure 25. Operational Wing Incidence Setting vs. Horizontal C.G. Position

Center of Gravity Limits

A total of 7.5 inches of horizontal C.G. travel has been established as allowable limits for the airplane. The forward C.G. limit is at Fuselage Station 98.5, and the aft limit is at Fuselage Station 106.0. Nominal C.G. was considered to be at Fuselage Station 103.0. The maximum forward and aft C.G. limits were the maximum attainable with the aircraft configuration under test. These limits were dictated by the position of the instrumentation pallet located on the cargo platform. For this reason, the limits as defined here are not to be taken as absolute limits defined by marginal control or safety of flight.

No limits were established for vertical center of gravity travel. Throughout the test program, the vertical C.G. was maintained between water lines 35.0 and 36.0.

Propeller Blockage

Two flights were made with simulated cargo loads set at various heights above the cargo platform to determine any possible effects on performance and handling qualities. The first simulated cargo load tested, Figure 26, was a box measuring 57 x 65 x 21.5 inches set toward the rear end of the platform. The width of the load was equal to the width of the platform and the height represented a distance equal to half the distance from the platform to the engine thrust line. The second configuration, Figure 27, consisted of an additional box measuring 57 x 32.5 x 17 inches set sideways on the lower box. This brought the cargo height to within 5 inches of the thrust line. Both flights were made at maximum gross weight and nominal C.G.

General handling qualities on both flights were favorable throughout all flight regimes. No noticeable changes in forces, control response, and maneuverability could be detected.

A change in airplane performance was observed as a result of propeller blockage. Figure 28 shows the change in the power-required curve for the two configurations tested. The same trim, maximum, and minimum speeds were attained; however, more power was required in each case.

A rate of climb and rate of descent performance check was also made for each configuration. Figures 29 and 30 show the effect of climb and descent performance respectively. A decrease in rate-of-climb performance of 150 feet per minute is experienced with maximum propeller blockage. A corresponding 150-foot-per-minute increase in rate of descent is obtained for the same configuration. This degradation in climb and descent performance is commensurate with the increased power required to obtain the same speed at cruise, thus indicating increased drag and/or decreased propeller efficiencies.

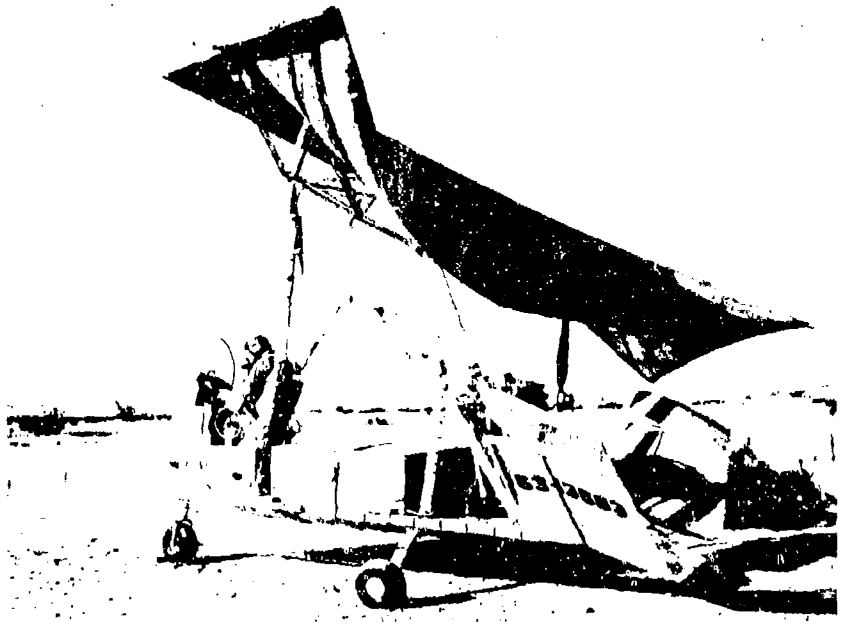


Figure 26. Medium Height Propeller Blockage Configuration

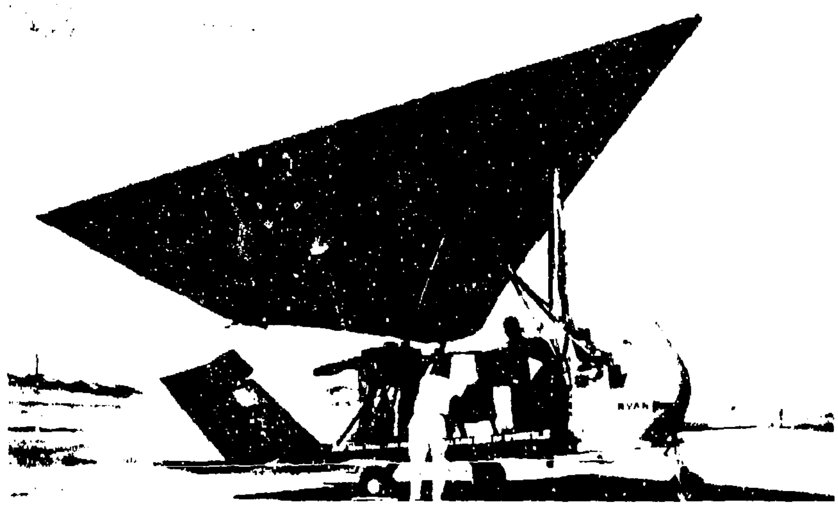


Figure 27. Maximum Height Propeller Blockage Configuration

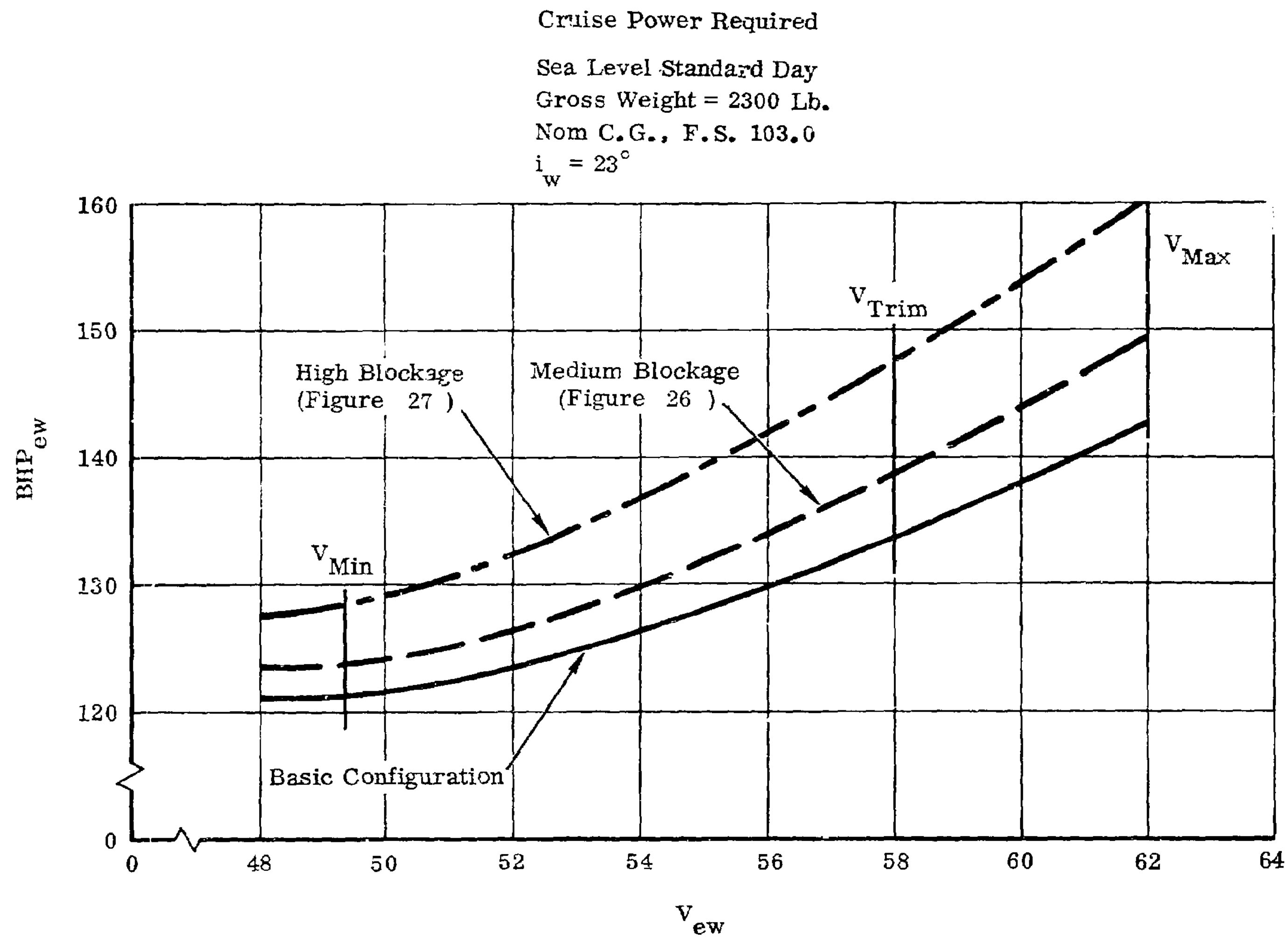


Figure 28. Effect of Propeller Blockage on Cruise Power Required

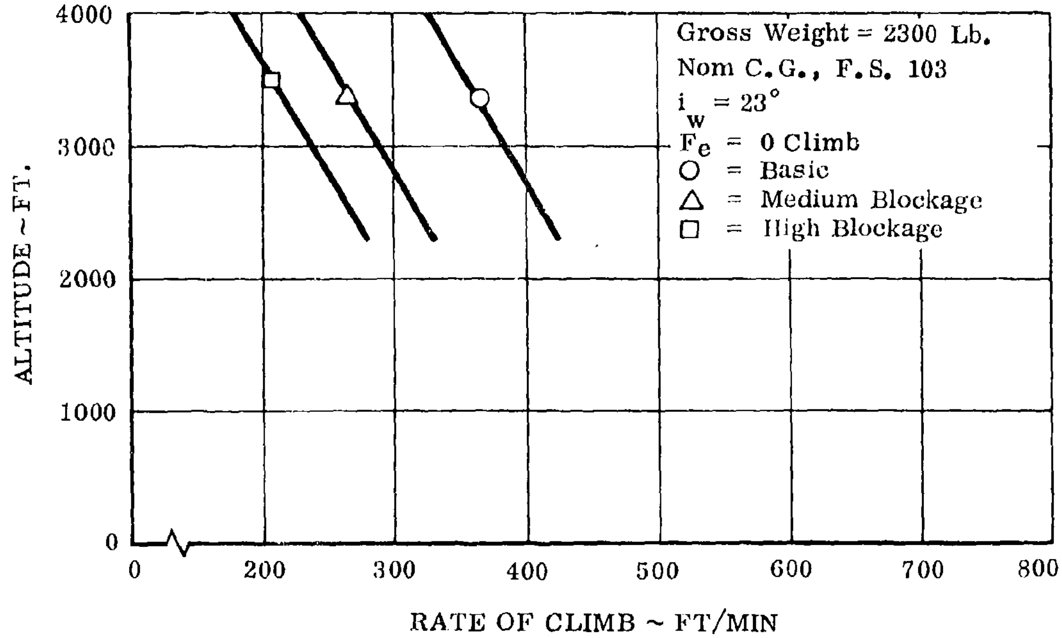


Figure 29. Propeller Blockage Effect on Rate of Climb

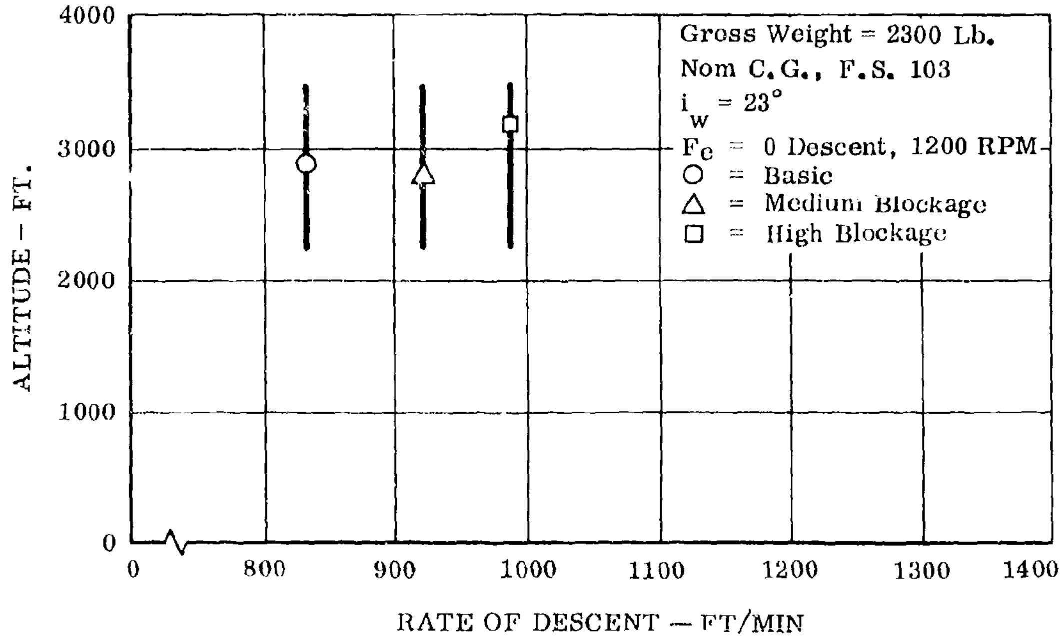


Figure 30. Propeller Blockage Effect on Rate of Descent

Crosswind Capability

Experience to date shows that the aircraft can be landed easily in ground distances of 75 to 150 feet, which reduced the crosswind operational question primarily to taxi and take-off. Because of the low approach speeds, rugged landing gear, and gear geometry, it is felt that a suitable area can always be found for an approach generally into the wind (e.g., taxiways, across a normal runway, helicopter pads, reasonably flat open field or desert, etc.). Using the two-control system, landings in winds up to 5 knots/90 degrees across and up to 10-15 knots/20-30 degrees across are feasible. The best crosswind landing technique is to accept the crab angle and to drift with resulting side loads on touchdown, using immediate directional correction by nose-wheel steering, which is very effective. Corrections can be made for drift, using roll control down to the flare point; but such corrections cannot be held without a separate directional control system.

Taxi operations are feasible in winds of 20 knots and possibly higher at reasonably slow speeds. During crosswind taxiing, the upwind wing will tilt up, full deflection, and the pilot will have no lateral control authority until a ground speed of about 25-30 miles per hour is reached (when q forces provide enough lateral control power for wing control).

Since take-off ground rolls average 350 to 550 feet, depending on gross weight, a safe useable technique was evolved for crosswind take-offs varying from 15 knots from 35-40 degrees, to 10 knots from 90 degrees. Maximum power is applied with brakes, followed by roll with wing tilted until about 25 miles per hour, when the wing can be rolled down into the wind. The aircraft is lifted off the ground at 40 miles per hour, and roll control is used to correct heading if necessary for climb.

Rough Terrain Operation

Three test operations were conducted from an unprepared desert surface. Several take-offs and landings were made with observed ground roll distances very similar to those attained on hard-surfaced runways. No operational difficulties were encountered during any of these test operations.

These operations proved the structural integrity of the main gear Fiberglas strut system. Sufficient flexibility is in the strut system to absorb high landing impact loads and bump loads encountered on unprepared surfaces.

One explicit advantage realized from these operations is that the take-off and landing runway becomes omnidirectional, thereby eliminating any concern for crosswind.

Loads

A complete static structural test program was completed on the aircraft prior to initiating the flight test program. At this time, all structural members were tested to the design load limit and in all cases were found to be satisfactory.

During the flight test program, key structural members on the aircraft were fitted with strain gages to permit monitoring of the loads received in flight. At no time during the test program were any of the loads observed to be beyond limits. Table 5 includes the observed and allowable loads for the structural members monitored.

TABLE 5
SUMMARY LOADS DATA

	Function	Allowable	Actual
T*	Keel @ Apex	33000 psi	1800
T	Center Strut	42000	900
C/T	Spreader Bar (top left)	5573/40000	5000/700
C/T	Spreader Bar (diag. left)	5573/40000	1400/15000
C/T	Fwd "V" (left)	14050/20200	3700/2000
C/T	Fwd "V" (right)	14050/20200	2200/1500
C/T	Aft "V" (left)	10500/10500	1500/1700
C/T	Aft "V" (right)	10500/10500	1000/3400
Load Pitch Cable		3700 Lb.	150 Lb.

*T = tension

c = compression

Aircraft Maintenance and Serviceability

Virtually maintenance-free operation was experienced throughout the entire testing period. A 100 percent in-commission rate was achieved for a program time of 46 engine hours, 36 of which were flight hours. Due to the simplicity of the entire system, routine maintenance consisted of brief preflight and post-flight checks, which were easily accomplished in a short period of time between operations. Airplane turnaround times depended solely on the time required for refueling.

No engine discrepancies were logged during the program, thus establishing the reliability of the installed power plant. Inasmuch as there are no generator and starter installed, the engine is started by hand-spinning the propeller. The engine never failed to start within two tries, even after the engine had been idle for two months. The only mechanical discrepancy encountered was a flat oleo caused by a leaking O-ring seal. The wing proved to be trouble free and required no special treatment or techniques. Tire wear was commensurate with conventional lightweight aircraft.

This program has demonstrated the ability of the XV-8A to be operated and maintained in austere environments with minimum crew and logistic support.

STABILITY AND CONTROL

Longitudinal Characteristics

Longitudinal trim is accomplished by decreasing the wing incidence for increased speed and increasing incidence for decreasing trim speed. The available incidence range was more than adequate to trim for any flight condition; however, practical limits do exist. At lower incidence angles and high speed, the fabric begins to flap and produces a mild aileron oscillation of 1 to 3 cycles per second. The buildup is gradual and serves as an excellent speed warning. A minimum wing incidence of 21 degrees for aft C.G. and 23 degrees for forward C.G. locations was selected to minimize the oscillation. A limit is also required for the higher incidence angles to avoid pitch-up, which occurs at high angles of attack. Lateral control also decays as high angles of attack are approached, and the aircraft rolls off as stall speed is reached. The maximum trim wing incidence selected was 23 degrees for the aft C.G. and 25 degrees for the forward C.G. to provide adequate margins from roll off and pitch-up and to give elevator maneuvering capability below the trim speed. With these wing incidence settings, satisfactory limit speeds are obtained with maximum elevator throws. Since the speed range is small, it is practical to set the trim for the entire flight based on the horizontal C.G. location. Flight path control is obtained in the conventional manner with elevator and power variations. The level flight trim speed versus wing incidence angles for forward, mid, and aft C.G. locations are shown in Figures 21 through 24. Elevator angles versus airspeed for several wing incidence angles and C.G. locations are plotted in Figures 31 through 34. The maximum and minimum speed limits are also shown.

The pilot's comments (on static longitudinal stability) indicate light elevator forces with positive stability throughout take-off, climb, cruise, approach, and landing maneuvers. Figures 35 and 36 present stick force versus airspeed for a constant trim setting during climb, cruise, and approach. The slope of the force curve is similar for the climb and cruise conditions at 2 pounds/mile-per-hour speed change. At idle power, the force gradient becomes more positive wi with 4 - 5 pounds/mile-per-hour speed variation. Stick force/g data was not obtained due to the low g-maneuvering capability of the aircraft.

The maximum load factor recorded during any maneuver was 1.1 to 1.2 g's. This in no way limits the maneuvering capability of the airplane, since the turning characteristics of the flexible wing are excellent.

The long-period dynamic longitudinal stability characteristics are shown in Figures 37 and 38. The times for the pitch oscillations to damp to one-half amplitude for the aft C.G. conditions during climb and cruise average 8.5 seconds, compared to a predicted value of 10 seconds. The cycles required to damp to one-half amplitude for the same points average .7 cycle, comparing closely with the predicted value of .66 cycle (Table 6).

The short-period dynamic longitudinal stability was reported to be excellent by the pilot. A typical plot is shown in Figure 39. The recovery of the elevator from the up-elevator pulse appears to be dead beat, and the response of the aircraft does not show any short-period oscillation characteristics. It is concluded that the amplitude of the short-period oscillation is too small to be significant and is not shown by the instrumentation.

The effect of power reduction on trim is quite significant. Due to the high thrust line, an engine power decrease will produce a nose-up pitching moment. The power chop data presented in Figure 40 indicates approximately 8 degrees of down elevator are required to maintain trim speed for a rapid power reduction from take-off to idle setting. This compares favorably with a predicted value of 10 degrees down elevator for a complete power failure. The rapid application of take-off power produces a corresponding nose-down pitch, and the data indicates that approximately 7 degrees of up elevator from trim would be required to maintain trim speed.

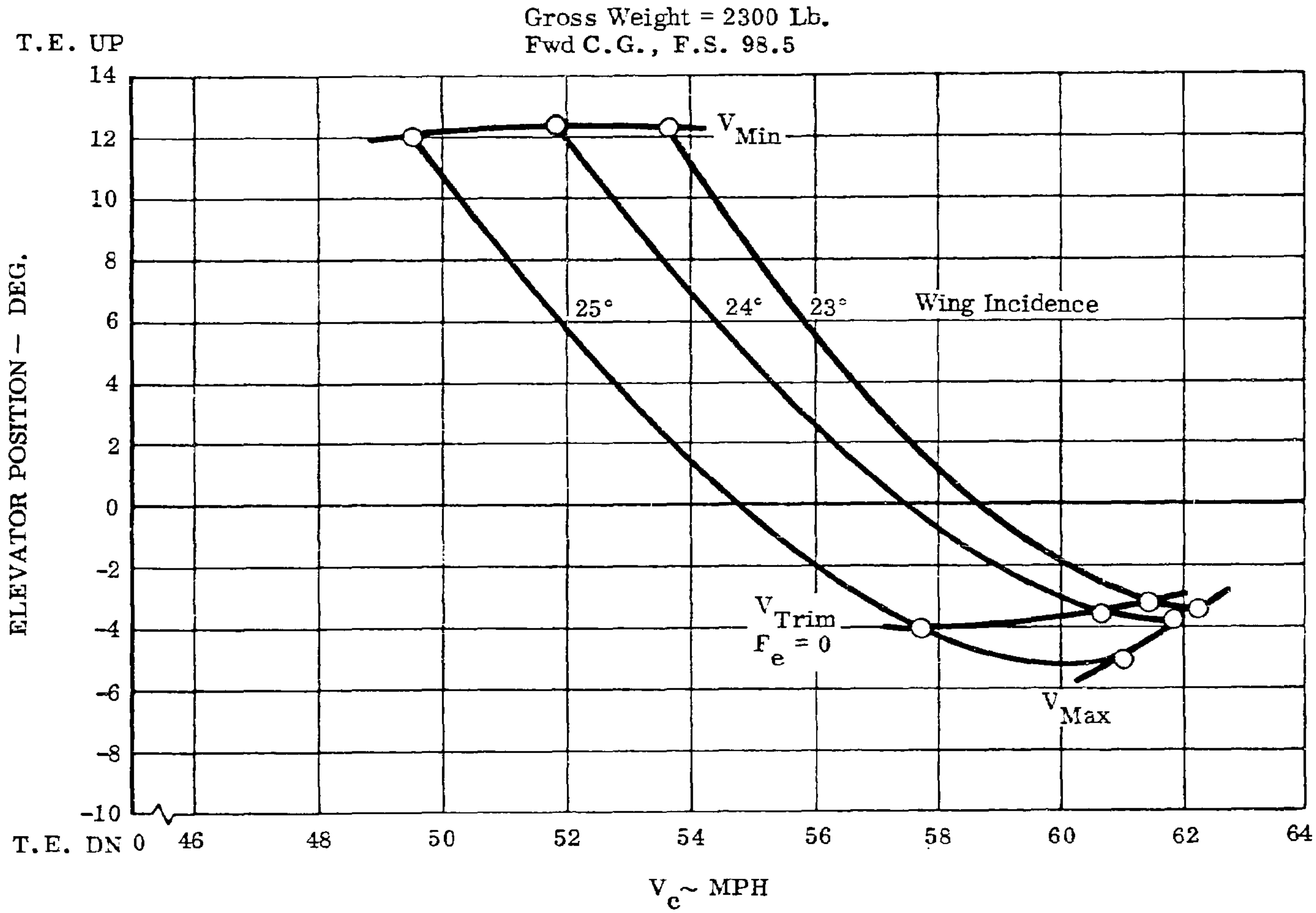


Figure 31. Elevator Angles vs. Velocity - Forward C.G., 2300 Pounds

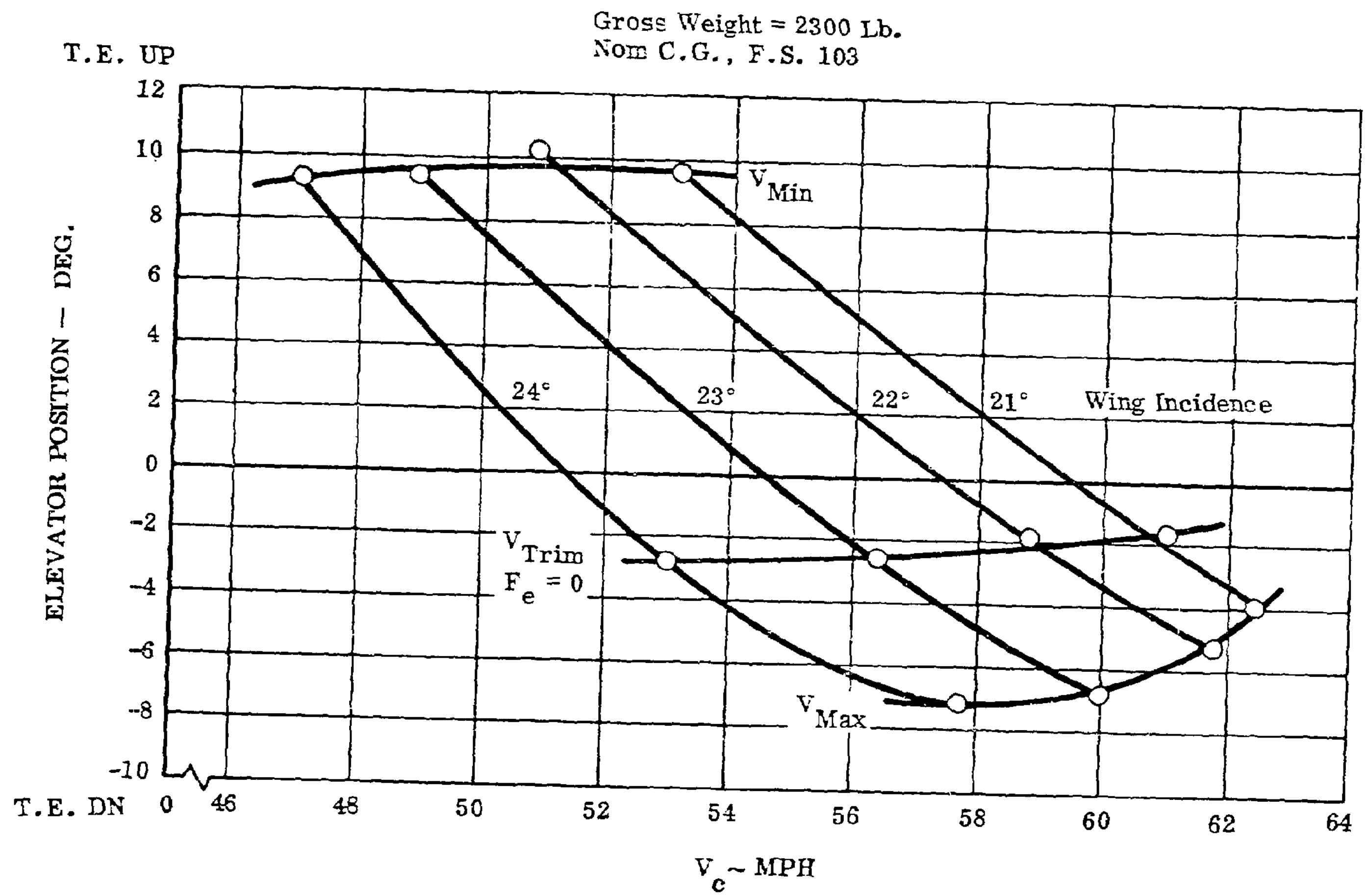


Figure 22. Elevator Angles vs. Velocity - Nominal C.G., 2300 Pounds

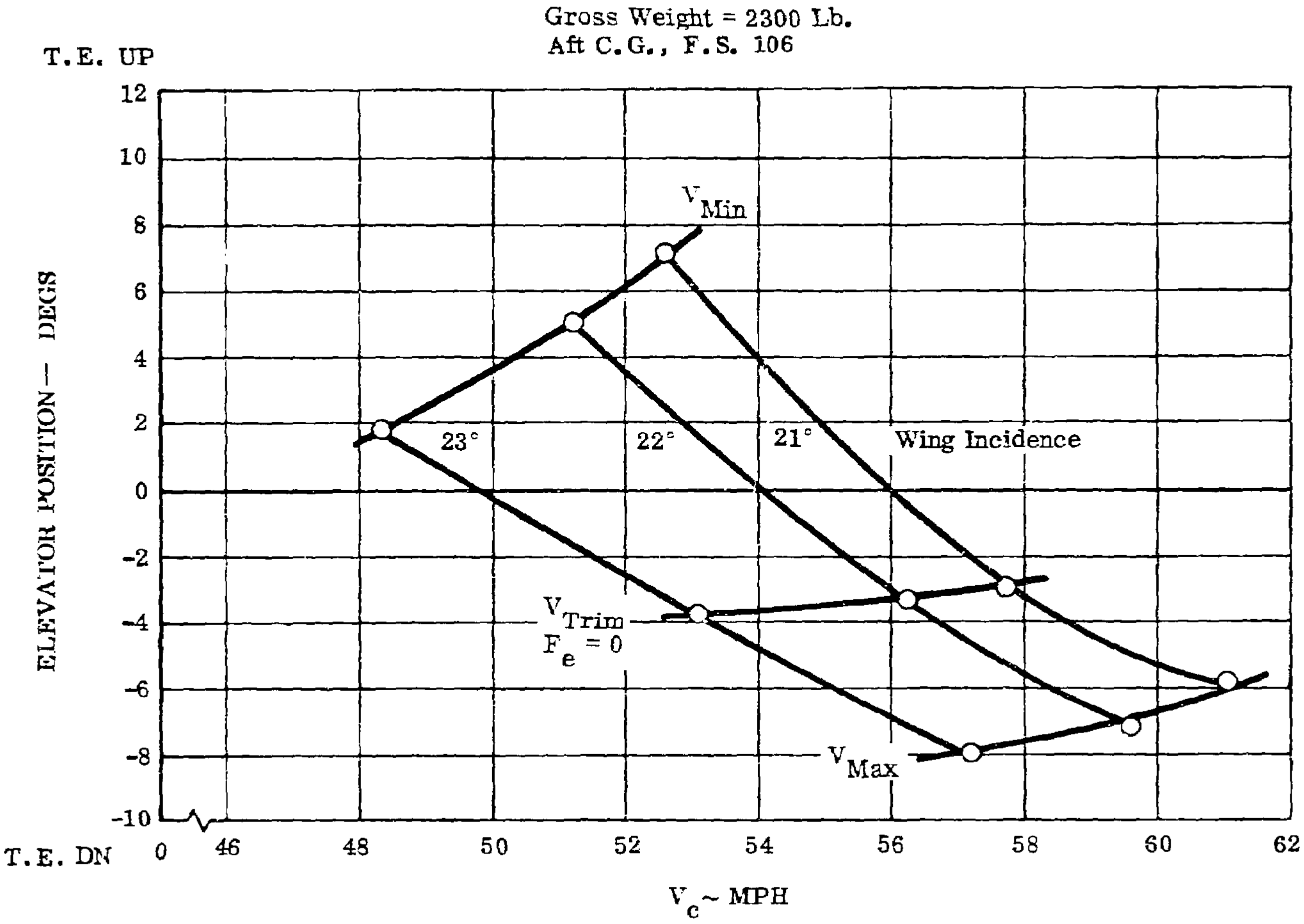


Figure 33. Elevator Angles vs. Velocity - Aft C.G., 2300 Pounds

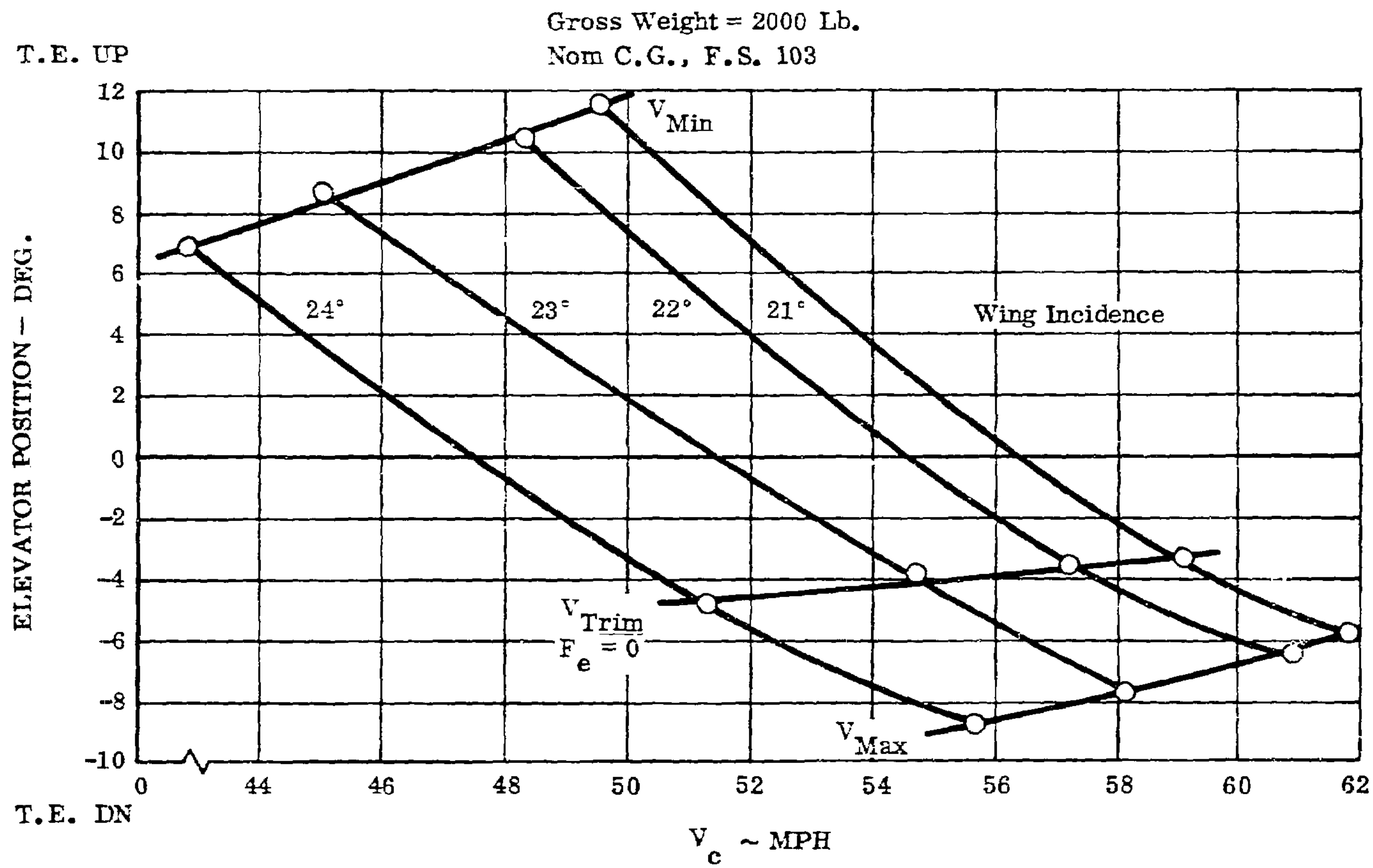


Figure 34. Elevator Angles vs. Velocity - Nominal C.G., 2000 Pounds

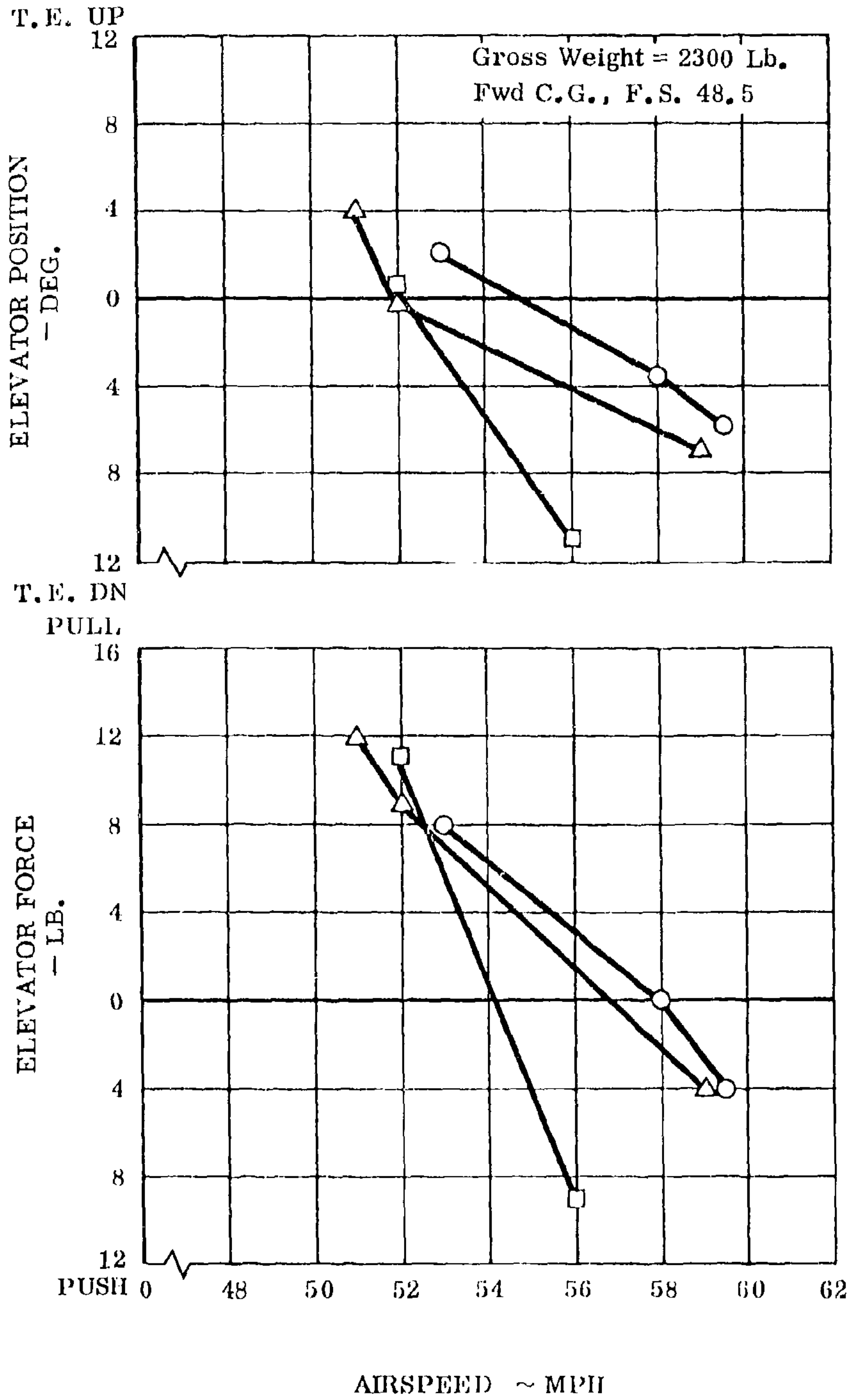


Figure 35. Static Longitudinal Stability - Forward C.G.

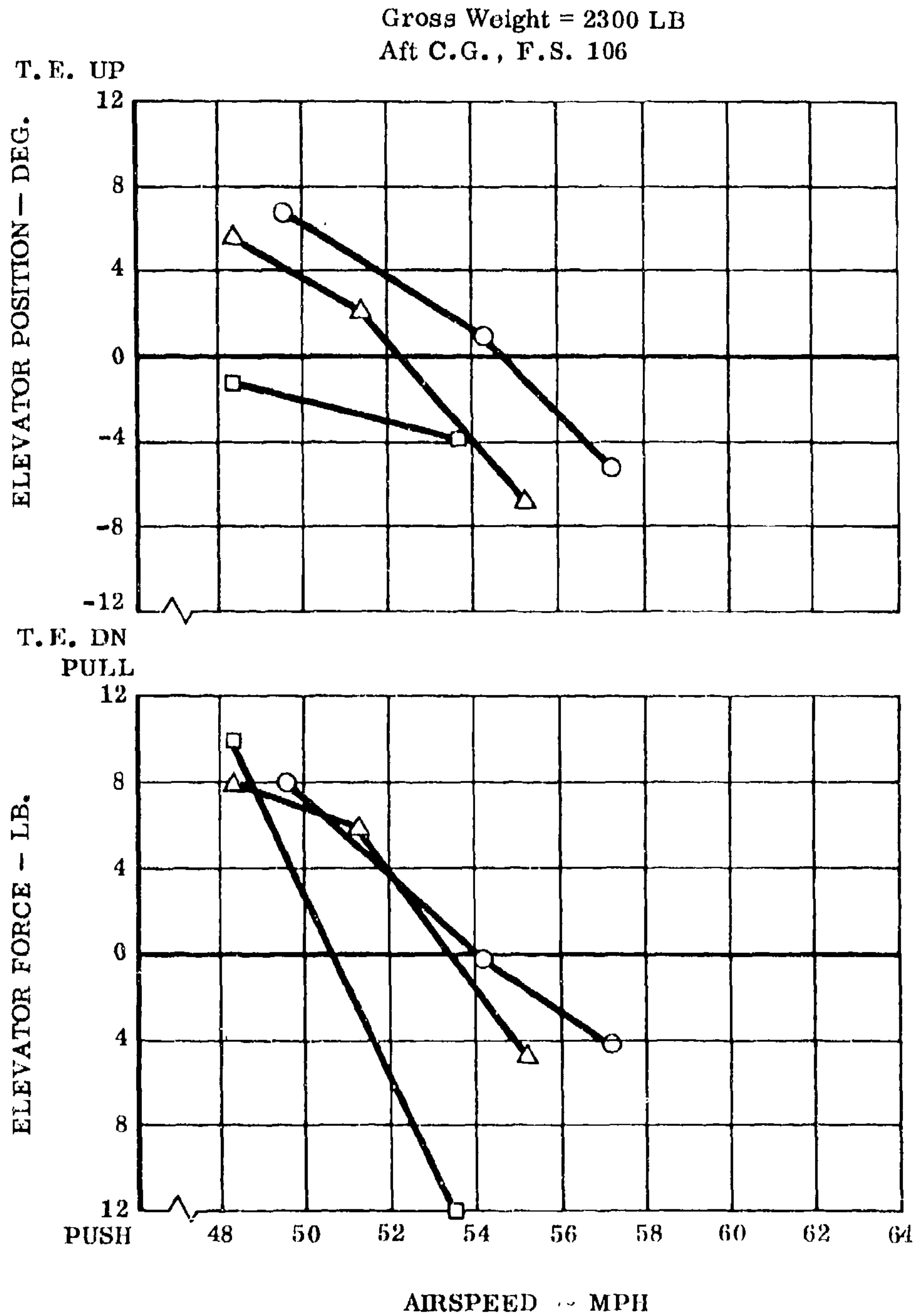


Figure 36. Static Longitudinal Stability - Aft C.G.

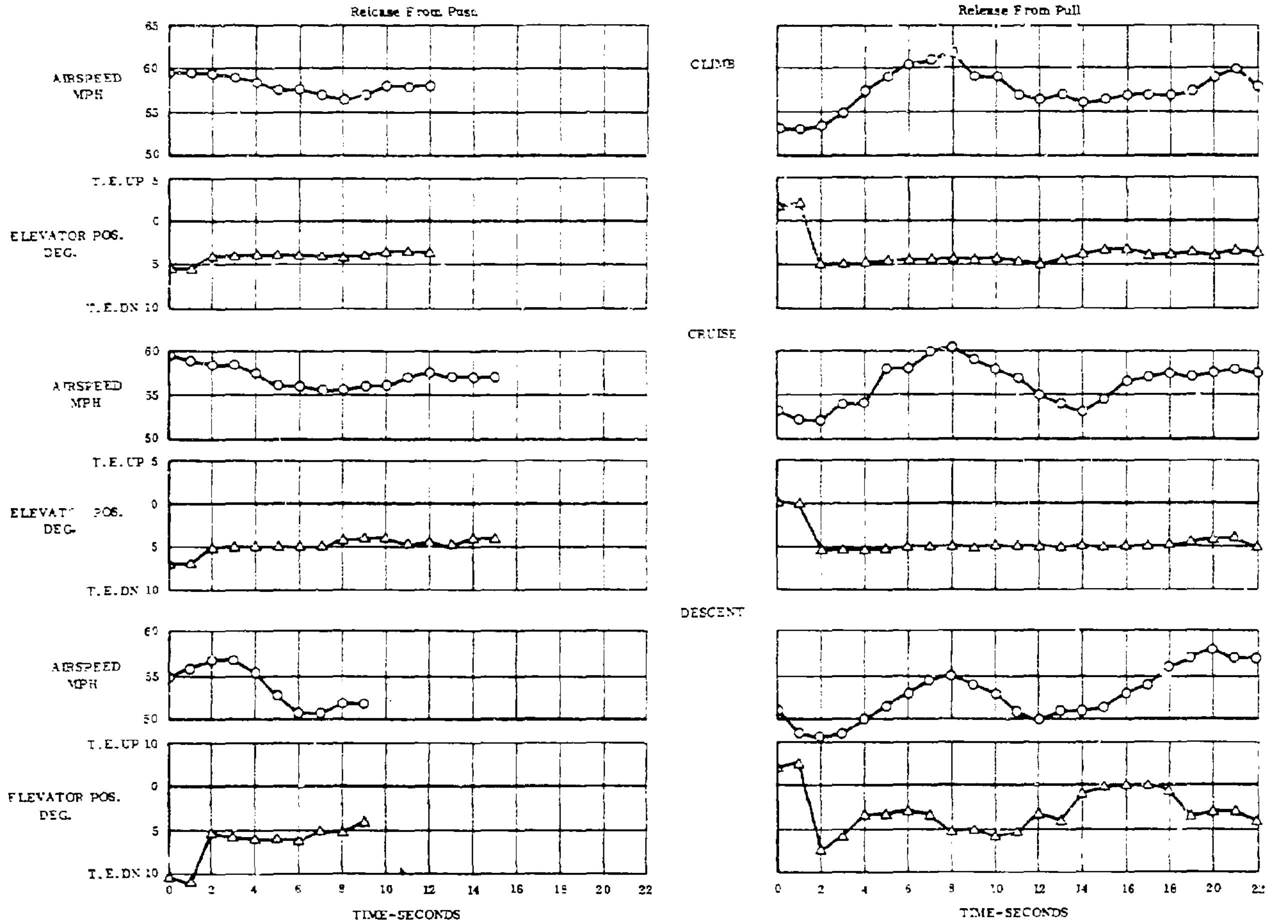


Figure 37. Long Period Dynamic Longitudinal Stability (Phugoid) Forward C.G.

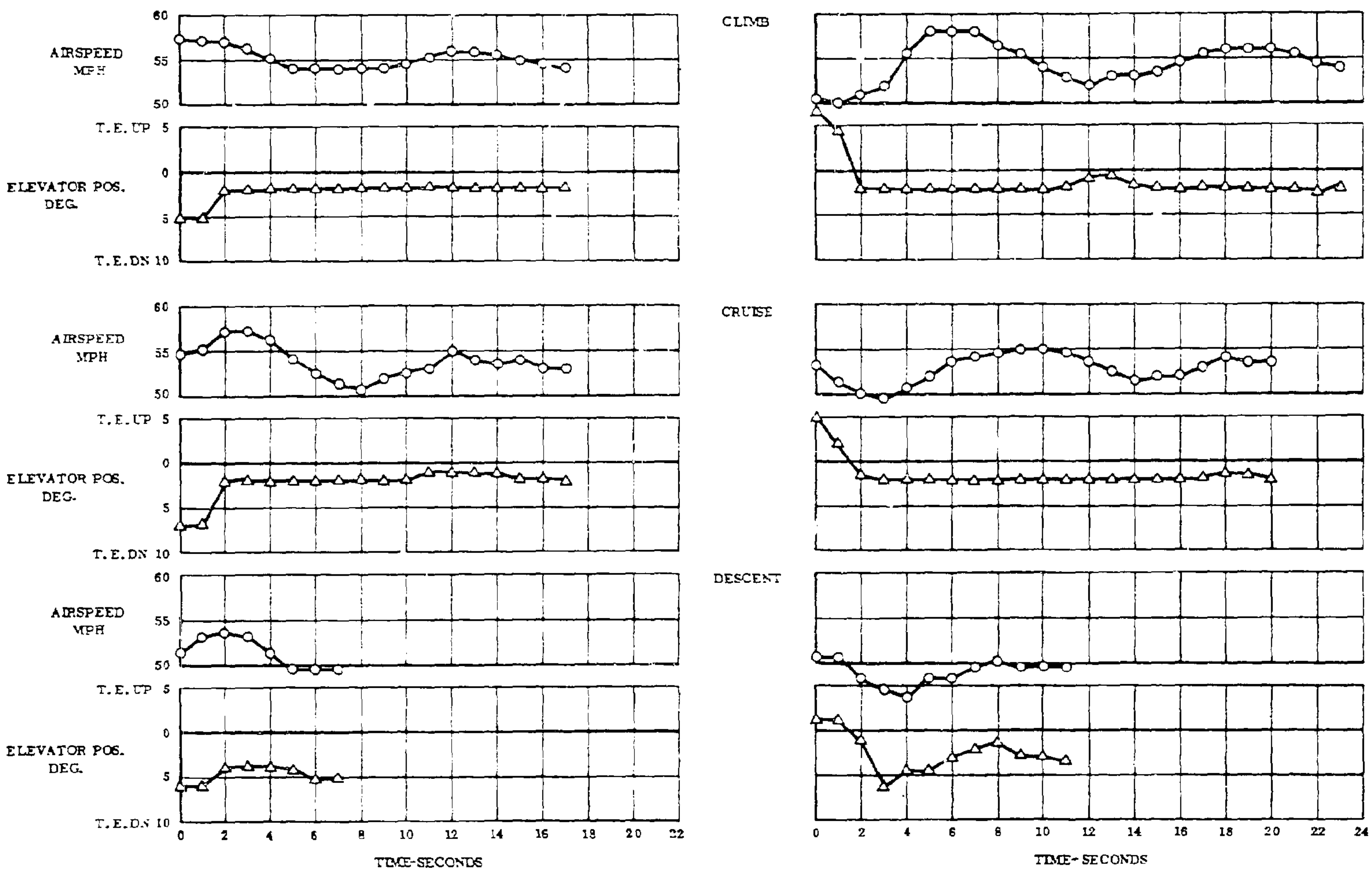


Figure 38. Long Period Dynamic Longitudinal Stability (Phugoid) Aft C.G.

TABLE 6
LONG-PERIOD LONGITUDINAL DYNAMIC STABILITY
(PHUGOID)

	Time to Damp to 1/2 Amp	Cycles to Damp to 1/2 Amp	Predicted Time	Predicted Cycles
Climb Release From Push	10	.77	9.8	.65
Climb Release From Pull	6	.45	9.7	.63
Cruise Release From Push	7	.64	10.1	.68
Cruise Release From Pull	11	.92	10.3	.70
Average Values	8.5	.7	10	.66

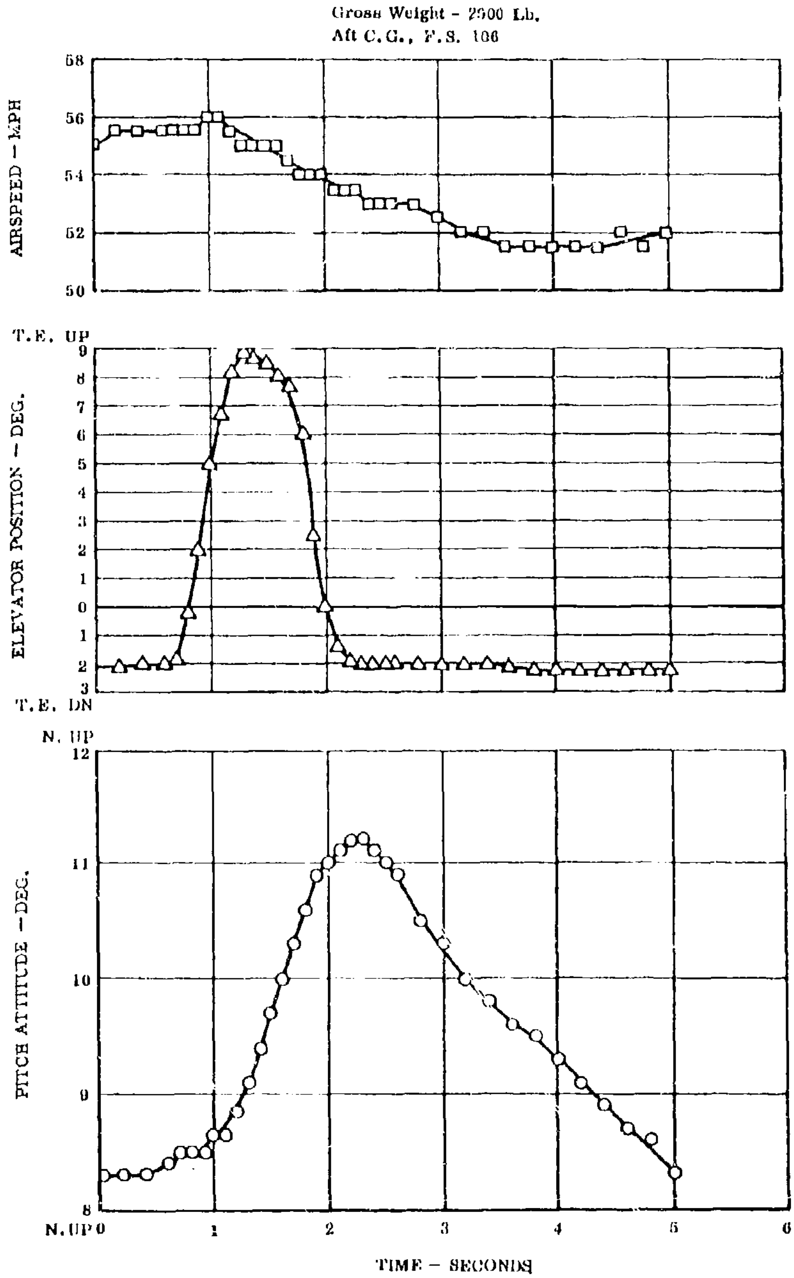


Figure 39. Short Period Dynamic Longitudinal Stability

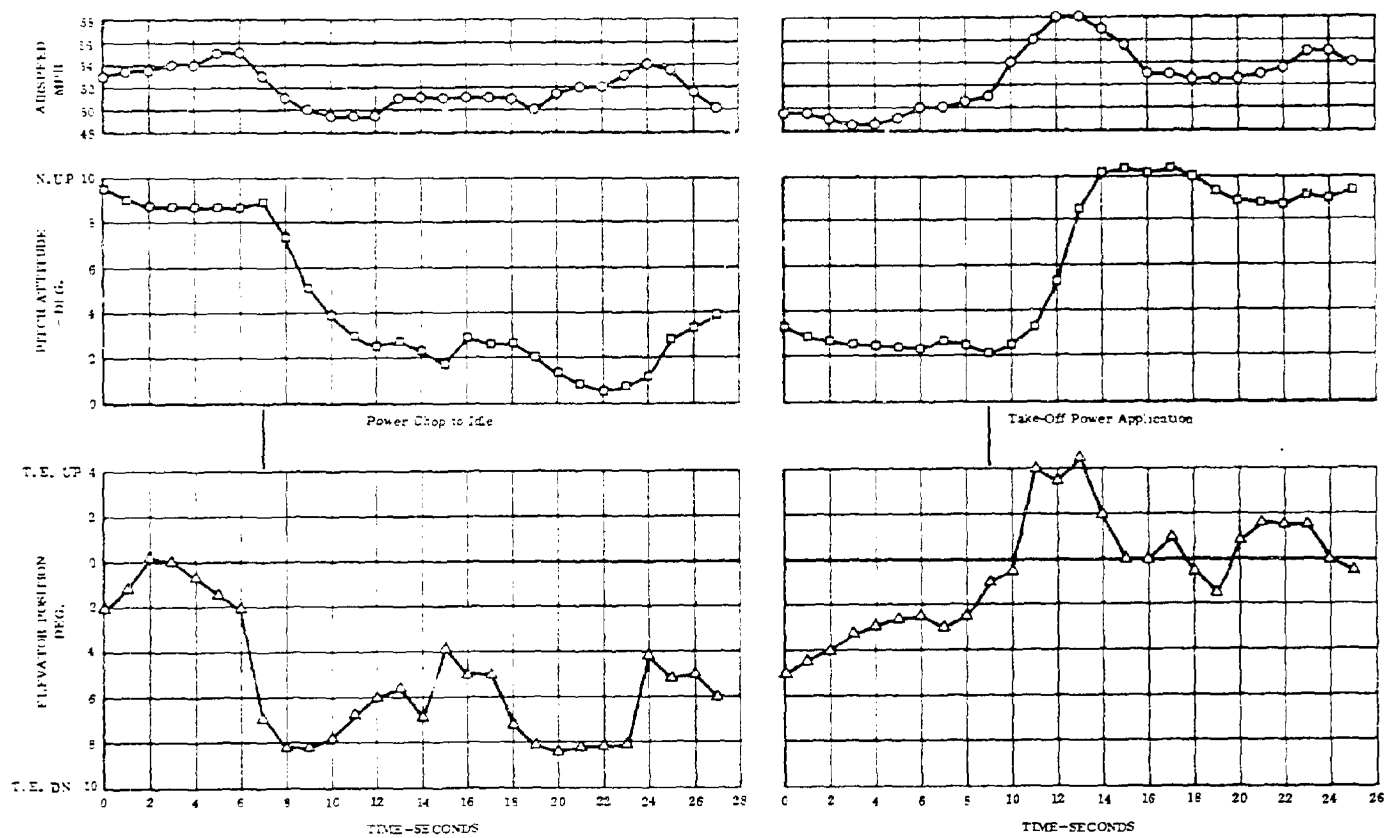


Figure 40. Longitudinal Controllability During Power Changes

LATERAL-DIRECTIONAL CHARACTERISTICS

The first one-half control wheel throw produces lateral control through conventional aileron motion. Continued wheel motion produces additional rolling control by moving the wing itself. During rapid roll maneuvers using full wheel, an abrupt increase in force gradient associated with moving the wing is apparent. The wheel force required to move through this artificial stop increases from about 12 to 25 pounds. Data from bank-to-bank rolls using one-half and full wheel deflections are presented in Figures 41 through 44. The average roll rate is 4 to 5 degrees/second for one-half wheel displacement and 6 to 8 degrees/second for full wheel. The pilot's report, that very little adverse yaw is apparent, is corroborated by the data indication of less than 4 degree sideslip angle for full control rolls. Bank angles of 20 to 30 degrees are readily obtained using ailerons alone. During the course of the test program, the aileron control system alone appeared adequate for roll control. Manual movement of the wing occurred only during tests specifically for full roll tests. The turn radius obtained with 20- to 30-degree bank is small enough for any normal purpose.

The aircraft has good positive spiral stability with no tendency to wrap-up in steep turns. The low wing loading prevents any significant build-up of acceleration loads, no matter how tight the turn. Also, lateral and longitudinal control forces become excessive for sustained pilot comfort when lateral directional maneuvers are attempted beyond normal performance requirements. This normal performance envelope, with acceptable control forces and good control harmony, provides a very tight turning radius and speed control, which should be more than satisfactory for any flight conditions or requirements.

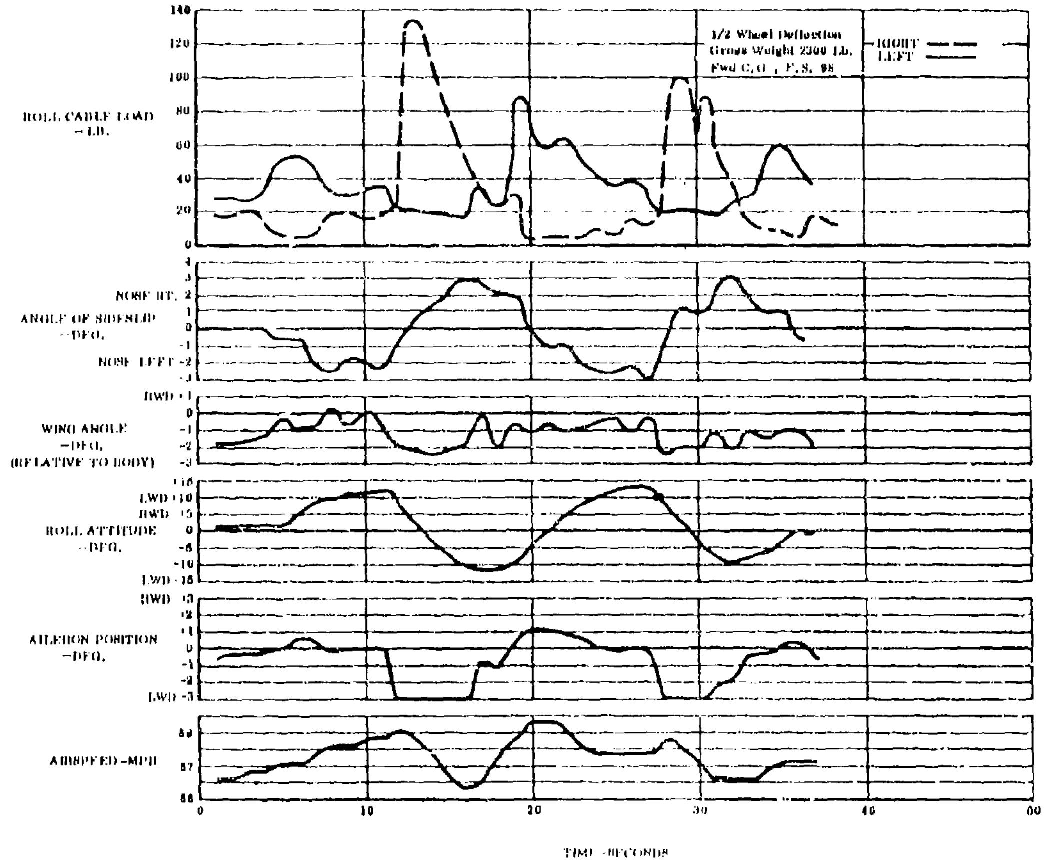


Figure 41. Bank to Bank Pull - 1/2 Wheel Deflection - Forward C. G.

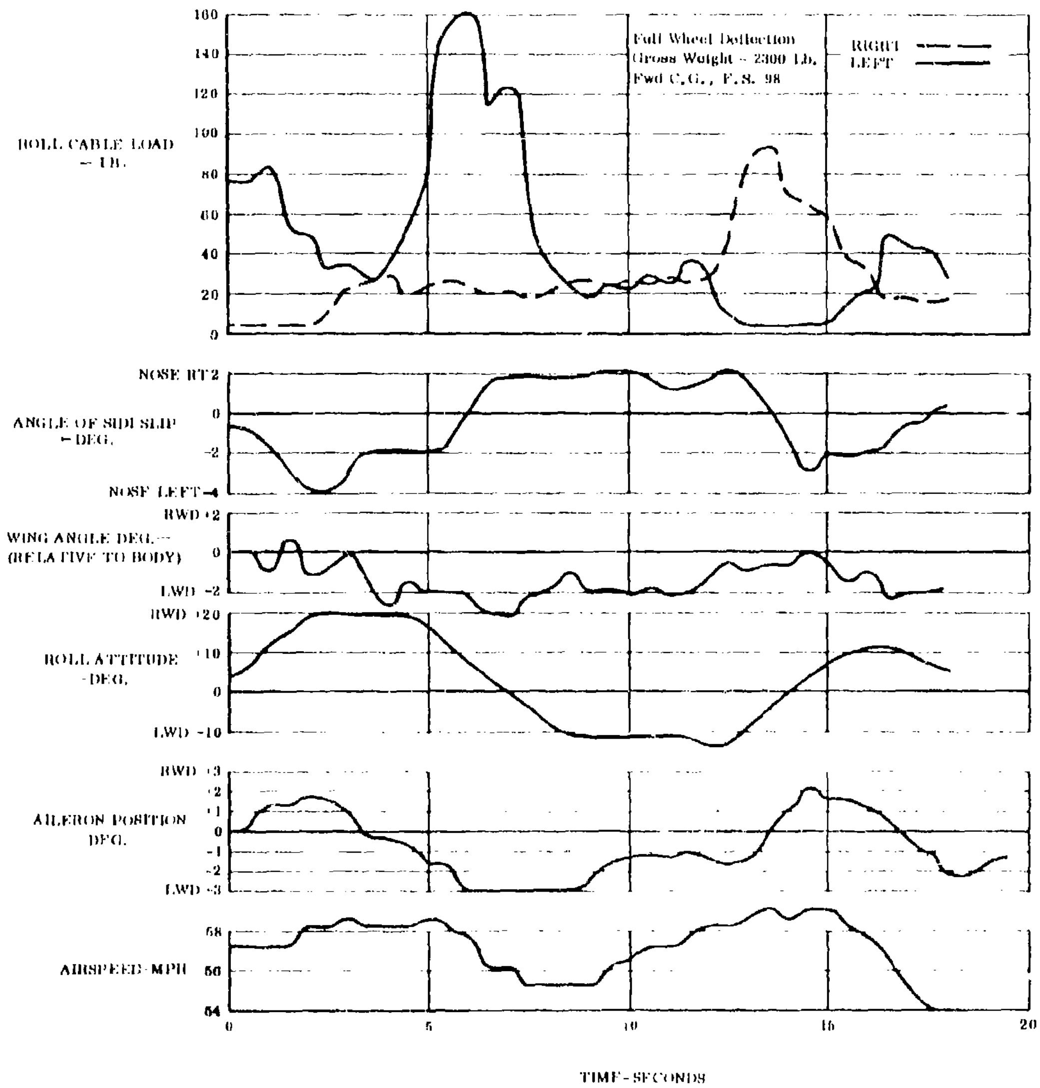


Figure 42. Bank to Bank Roll - Full Wheel Deflection - Fwd. C. G.

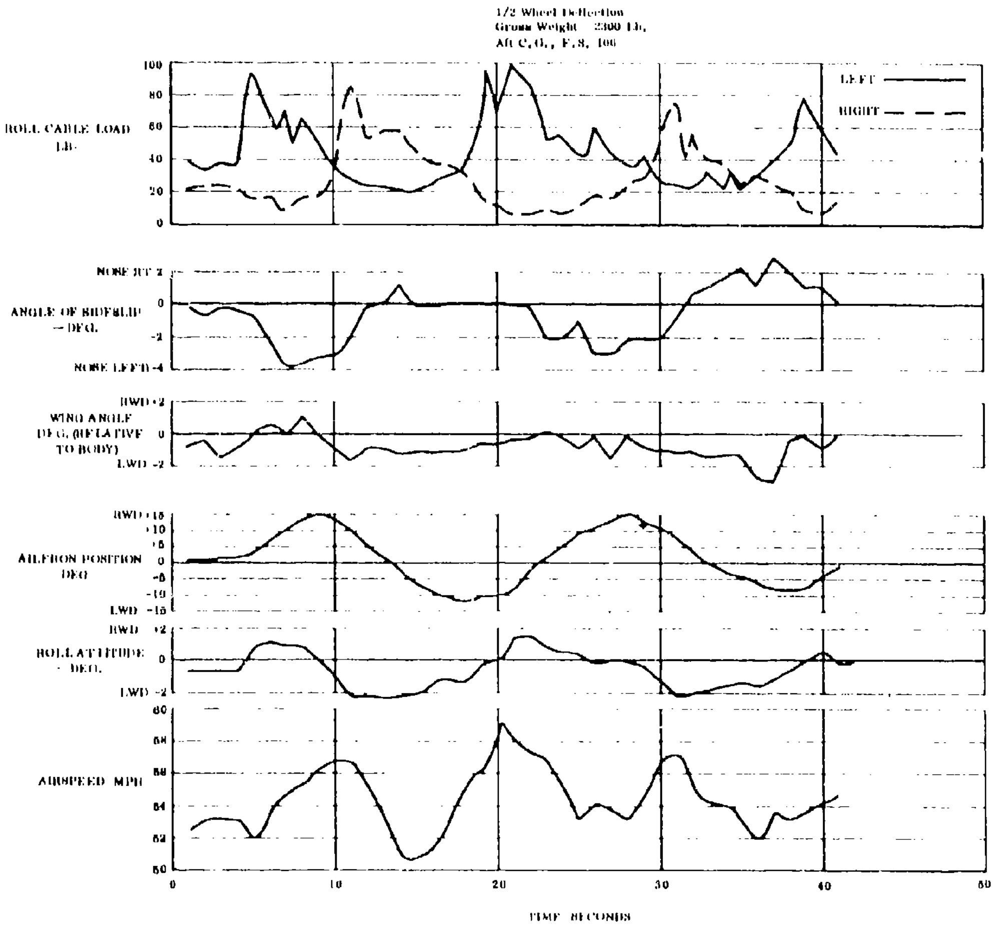


Figure 43. Bank to Bank Roll - 1/2 Wheel Deflection - Aft C.G.

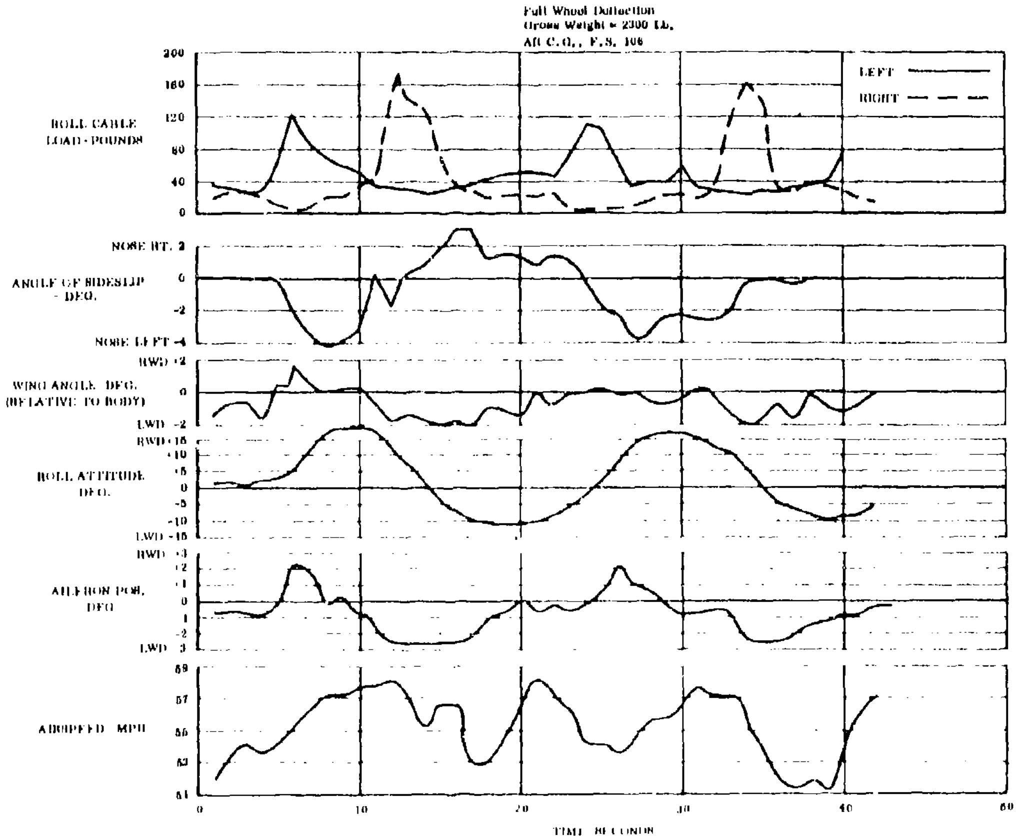


Figure 44. Bank to Bank Roll - Full Wheel Deflection - Aft. C. G.

FIXED WING OPERATION

One taxi operation was conducted to observe preliminary roll system effectiveness with the wing fixed relative to the fuselage, thereby utilizing only aileron action for lateral control. Initial high speed taxi run and low-level flights down the runway were made with gradually increasing bank angles and S-turns. An aileron-only lateral control system appeared feasible, but the control power in this configuration was marginal and would be acceptable only under calm wind conditions. Larger ailerons and/or more aileron deflection is necessary for additional control power to maintain a crosswind handling capability. This became obvious on the last taxi run, when a sharp gust from the right resulted in a complete loss of directional control. Available lateral control power was insufficient to maintain or regain control. Recovery was made by kicking the nose wheel left with drift and then right to regain control.

More testing with a modified aileron control system is required before a definite conclusion as to the feasibility and/or practicability of an aileron-only lateral control system can be made.

ADDENDUM

XV-8A SUPPLEMENTAL FLIGHT TEST REPORT

SUMMARY

The purpose of the additional test program was to qualitatively determine the control and handling characteristics of the three-control system, the effect of heavy transverse battens, the effects of cross-wind, and the idle-power landing capability with high (height) cargo, and to permit the conducting of additional fixed-wing tests.

The three-control system does improve the flying qualities of the aircraft but is not required for normal flight operation. Its greatest asset is that it doubles the cross-wind handling capability during take-off and landing. Rudder forces are light (< 5 pounds) and control harmony is good.

The heavy transverse battens had little effect on the flight speed at which aileron oscillation occurred. The frequency and amplitude of the ensuing oscillation were both increased over that encountered with the original battens.

Take-off and landings were made in 90-degree cross-winds up to 10 knots and 60-degree winds up to 15-17 knots.

Sufficient elevator control power is available to make idle-power landings with high cargo loadings.

The aircraft affords ample stall warning at both nominal and aft C.G. limits. Standard stall recovery techniques are applicable. At aft C.G., if the aircraft is held in a stall condition, the possibility exists that a spin could be entered.

The fixed-wing tests were not accomplished. The additional aileron deflection obtained by rerigging, and necessary for increased lateral control, was insufficient to warrant additional testing.

CONCLUSIONS

1. The three-control system improves the over-all flying qualities of the airplane.

2. The cross-wind limits for take-off and landing are doubled by the use of the three-control system.

3. Heavier tranverse trailing-edge battens do not alleviate the aileron oscillation encountered near maximum speed.

4. Sufficient elevator control power is available with high (height) cargo loadings to make idle-power landings.

5. The aircraft affords ample stall warning of both nominal and aft C.G. limits. Standard stall recovery techniques are applicable. At aft C.G., if the aircraft is held into a stalled condition by applying full aft stick, the possibility exists that the aircraft might enter into a spin.

TEST RESULTS

Three-Control System

The incorporation of a directional control system in the XV-8A aircraft was accomplished by providing differential deflection of the elevator (ruddervator) surfaces. Control was actuated by the use of conventional rudder pedals.

Standard directional control testing techniques were used to evaluate the system.

The maximum steady-state, wings-level, sideslip angle achieved with full rudder deflection was approximately 4 to 5 degrees. Rudder forces were estimated at less than 5 pounds.

With the lateral control system held fixed in the neutral position, balanced turns and bank-to-bank maneuvers can be made with rudder only. With full rudder deflection, a steady 30-degree banked turn can be maintained with no tendency to diverge. Consequently, the directional control system increases the roll rate and reduces the turning radius of the airplane.

A rudder-lock condition develops when the rudder is deflected beyond one-half of the total travel available. Since the rudder forces are light and the lateral control power is strong enough to override the directional control system, no adverse or dangerous tendencies were encountered due to the rudder-lock condition. The incorporation of a simple rudder neutralizing spring would eliminate the rudder-lock.

Dynamic rudder pulses showed that the aircraft was stable and highly damped. Spiral stability checks were also positive with no tendency for divergence. Adverse yaw during bank-to-bank roll maneuvers was hardly noticeable and qualitatively comparable to that experienced with the two-control system. The three-control system proved to be an asset in cross-wind take-off and landing characteristics. Conventional cross control techniques can be used to correct for wind on the landing approach. The 90-degree cross-wind limit for take-off and landing was increased from 4 to 9 knots.

Heavy Batten Investigation

Heavier transverse trailing-edge battens with a rigidity of approximately twice that of the original battens were tested in an effort to determine their effect on the aileron oscillation phenomenon at high speed.

Speed checks made with this configuration still resulted in aileron oscillation. The most significant observation was that the heavy battens masked the random pulses and slow oscillation buildup experienced with the original configuration at speeds of 62 to 63 miles per hour.

Consequently, when the oscillation does become noticeable at a speed of 64 to 65 miles per hour, it is much more sudden and severe with a frequency of 6 to 8 cycles per second as compared to 2 to 3 cycles previously reported. Considerably more airframe buffet was also experienced.

Subsequent tests with the original battens installation showed that the same oscillation characteristics are present if the airplane is flown at the same speeds, 64-65 miles per hour.

It was, therefore, concluded that the heavier transverse battens merely masked the onset of the aileron oscillation and did not eliminate the oscillation phenomenon.

Stall Investigation

Stall investigations were made at nominal and aft C.G. and at maximum gross weight.

The stalls at nominal C.G. are conventional with no adverse characteristics and easy recovery. Pitch control is very positive for recovery, as is power application. Rudder control is marginal at stall but does help maintain directional control during initial stall entry. Lateral control is effective throughout the stall maneuver and can accelerate the stall when abrupt inputs are made during the maneuver. At the stall, the aircraft tends to fall off to the right in a nose-down attitude. Recovery can be accomplished with only a 50- to 100-foot altitude loss.

The aft C.G. stall investigations showed basically the same characteristics; however, the stall entry is much easier and more abrupt. The effectiveness of the roll and yaw control is in essence the same as at nominal C.G.; however, the pitch control is reduced due to the normal forward stick position at aft C.G. By holding the aircraft in the stall maneuver, well past the initial stall onset, the aircraft acts as though it may be attempting to enter into a spin. This condition was not carried to the point of determining if the aircraft would actually spin.

In all cases, the aircraft gives sufficient warning when approaching a stall, and stall recovery can be made with either control movements, power application, or both.

High-Cargo, Idle-Power Landings

A series of landings were made at reduced power settings from 1600 to idle rpm to determine the feasibility of landing with idle power and a high (height) cargo loading. The simulated cargo load was identical to the highest cargo load used for the propeller blockage tests conducted during the basic test program. The flight was made at forward C.G., F.S. 98.5. No adverse characteristics were observed during these tests, and sufficient elevator control power is available to execute the landing flare maneuver safely and properly. Forward C.G. requires earlier initiation of the flare maneuver. At forward C.G., the stick position at trim is more aft and, consequently, less incremental stick travel or elevator deflection is available.

Cross-Wind Operational Evaluation

The cross-wind take-off and landing limits were determined at 2300 pounds gross weight and at nominal C.G. (F.S. 103).

Take-offs were made in 30-degree cross-winds at 12 knots, 60-degree cross-winds at 15 knots, and 90-degree cross-winds at 10 knots. Landings were made under similar conditions except that the maximum 90-degree cross-wind experienced was 9 knots.

The addition of the directional control system more than doubled the cross-wind capability of the airplane. With the winds encountered during test, conventional flying techniques could be employed to correct for drift and to hold a runway heading. Consequently, side landing loads were considerably less than those experienced with the two-control system.

Wing Fixed

No tests were conducted with the wing locked in the fixed position. The additional aileron deflection required for increased lateral control power was not available by rerigging; consequently, further testing was not warranted.

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13 ABSTRACT

The results of a test program of a flexible wing manned utility vehicle are presented. Discussed are performance characteristics, handling qualities and operational flight envelopes. Included is a supplemental flight test report in the Addendum which reflects configuration changes.

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  Report Date: February 1, 1965

- Document Number: AD 460405
  Unclassified Title: XV-8A Flexible Wing Aerial Utility Vehicle
  Report Date: February 1, 1965

- Document Number: AD 431128
  Unclassified Title: Operational Demonstration and Evaluation of the Flexible Wing Precision Drop Glider in Thailand
  Report Date: March-July 1963

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