AFFDL-TR-71-134

VALIDATION OF THE FLYING QUALITIES REQUIREMENTS OF MIL-F-8785B (ASG)

R. N. KANDALAFT

NORTHROP CORPORATION
AIRCRAFT DIVISION

FOREWORD

This report was prepared by the Northrop Corporation, Aircraft Division, Hawthorne, California, for the Air Force Flight Dynamics Laboratory, Air Force Systems Command, United States Air Force, Wright-Patterson Air Force Base, Ohio. The study was conducted under Contract F33615-71-C-1065, Project -8219, Task -05. Mr. Jerry L. Lockenour (FGC) was the Project Engineer.

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This report was submitted on 1 September 1971, and the internal Northrop Report is NOR 71-127. It represents the views of the author, which are not necessarily the same in all cases as the views of the Air Force. The purpose of this report has been best served, it is felt, by giving its author wide latitude for expressing his views without inhibition.

C. B. Westbrook
Chief, Control Criteria Branch
Flight Control Division
Air Force Flight Dynamics Laboratory

ABSTRACT

This study was conducted to validate Military Specification MIL-F-8785B(ASG), "Flying Qualities of Piloted Airplanes," dated 7 August 1969, by performing a detailed comparison of its requirements with the known characteristics of the Northrop F-5 fighter and pilot comments on them.

The comparison was based primarily on existing flight test data supplemented by analytical data as required for this evaluation process. Paragraph by paragraph, validations or discrepancies are noted, resolution attempted if necessary, and any recommendations given.

In addition, recommendations are made enumerating experimental and analytical investigations beyond the scope of this study which will provide data for further validation and updating of the requirements.

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SYMBOLS

Term	Definition
δa	Total aileron position ($\delta a_L + \delta a_R$)
δa_L	Left aileron position
δa_R	Right aileron position
$\bar{c}$, MAC	Mean aerodynamic chord
C_L	Lift coefficient
Fa	Lateral stick force
F_R	Rudder pedal force
F_S	Longitudinal stick force
δh	Horizontal stabilizer position
hp	Pressure altitude
I_x	Roll moment of inertia
I_y	Pitch moment of inertia
I_z	Yaw moment of inertia
I_{xy}, I_{xz}, I_{yz}	Products of inertia
k	Ratio of "commanded roll performance" to "applicable roll performance"
KCAS	Knots calibrated airspeed
KIAS	Knots indicated airspeed
Kp	Pilot model gain parameter
MN or M	Mach number

Term	Definition
MRP	Military rated power
n_L	limit normal load factor
n_z or n	normal load factor
p	Roll rate
p_{osc}/p_{av}	A measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a rudder-pedals-free step aileron control command: $\zeta_d \leq 0.2$: $p_{osc}/p_{av} = \frac{p_1 + p_3 - 2p_2}{p_1 + p_3 + 2p_2}$ $\zeta_d > 0.2$: $p_{osc}/p_{av} = \frac{p_1 - p_2}{p_1 + p_2}$ where p_1, p_2, and p_3 are roll rates at the first, second and third peaks, respectively.
PFLF	Power for level flight
q	pitch rate
r	Yaw rate
δ_R	Rudder position
δ_S	Longitudinal stick position
tn_β	Time for the Dutch roll oscillation in the sideslip response to reach the n^{th} local maximum for a right aileron control step or pulse command or the n^{th} local minimum for a left command
T_D	Damped period of the Dutch roll $Td = 2\pi / \left(\omega_{nd} \sqrt{1 - \zeta^2}\right)$
$T_{\phi = x}$	Time required to roll through x degrees
T_L	Pilot model lead parameter
Vc	Calibrated airspeed

Term	Definition
$V_{o_{min}}$	Minimum normal approach speed
$V_{o_{min}}-5$	Minimum normal approach speed minus 5 knots
V_S	Stall airspeed
V_T	True airspeed
W	Airplane weight
Z_α	Normal force due to angle of attack
α	Angle of attack
β	Sideslip angle
γ	Flight path angle $\gamma = \sin^{-1} \dfrac{\text{vertical speed}}{\text{true airspeed}}$
ζ_d	Dutch roll damping ratio
ζ_{SP}	Short period damping ratio
θ	Pitch angle
τ_R	Roll mode time constant
ϕ	Roll angle
ϕ_{osc}/ϕ_{av}	A measure of the ratio of the oscillatory component of roll angle to the average component of roll angle following a rudder-pedals-free impulse aileron control command: $\zeta \le 0.2$: $\phi_{osc}/\phi_{av} = \dfrac{\phi_1+\phi_3-2\phi_2}{\phi_1+\phi_3+2\phi_2}$; $\zeta > 0.2$: $\phi_{osc}/\phi_{av} = \dfrac{\phi_1-\phi_2}{\phi_1+\phi_2}$; where ϕ_1, ϕ_2, and ϕ_3 are roll angles at the first, second and third peaks, respectively.
ψ	Phase angle

Term	Definition
ψ_β	Phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip, where $$\psi_\beta = \frac{-360}{Td} t_{n_\beta} + (n - 1) \quad 360 \text{ (degrees)}$$ with n as in t_{n_β} above
ω_{n_d}	Dutch roll undamped natural frequency $$\omega_{n_d} = \frac{\omega_n \text{ (damped)}}{\sqrt{1 - \zeta^2}}$$
$\omega_{n_{SP}}$	Short period undamped natural frequency

SECTION I

INTRODUCTION

This report is prepared as part of a program initiated by the AFFDL, Wright Patterson Air Force Base, Ohio, in a continuing effort to update Military Specification MIL-F-8785B(ASG), "Flying Qualities of Piloted Airplanes." The specification contains requirements that are applied by the aircraft industry in design, development and flight test demonstration of new airplanes.

A detailed comparison of the F-5 airplane flying qualities with the requirements of MIL-F-8785B(ASG), (7 August 1969), is contained in this report. The comparison process is conducted to evaluate each paragraph of the specification, In some instances where complete validation was not possible within the scope of this program, suggested experimental work and supplementary studies were enumerated for the continued task to revise and update the requirements.

The F-5 was the primary candidate airplane for this study. The T-38 airplane which is similar to the F-5 was used as required to provide additional data. The T-38 is a Class IV trainer airplane. The F-5 is a Class IV, multipurpose tactical fighter airplane, capable of carrying external armament and fuel stores for a dual mission of air-to-air combat and ground attack. Consequently, the data comprised wing configurations with and without external stores. The results pertain largely to Class IV airplanes.

The primary source of the data is flight tests which were conducted in accordance with the requirements of MIL-F-8785 (ASG) amendment -2, the prevailing specification for the F-5 during its design and development time period. However, these flight test data were specifically reduced for this report for direct comparison with the present specification. In addition, analytical flying qualities data were generated to supplement the existing flight test data as required for the comparison study. Still, inevitably comparison of any airplane with the specification will be less than complete because of data limitations. In many cases the data given are typical rather than inclusive. Although more thorough coverage would be needed to show compliance of a new airplane, this depth of presentation seems adequate for a validation report.

Validation of the requirements for airplane failure states was given special effort. A thorough review and appraisal were carried out of all failures affecting flying qualities experienced by F-5 and T-38 airplanes.

Emphasis was placed on presenting quantitative data, both analytical and flight test. In addition, where the need was indicated, suggested clarifications to the wording of the specification are presented. As an aid in comparing flying qualities with the specification requirements, various detailed methods of data reduction, analysis and presentation are included.

SECTION II

AIRPLANE DESCRIPTION

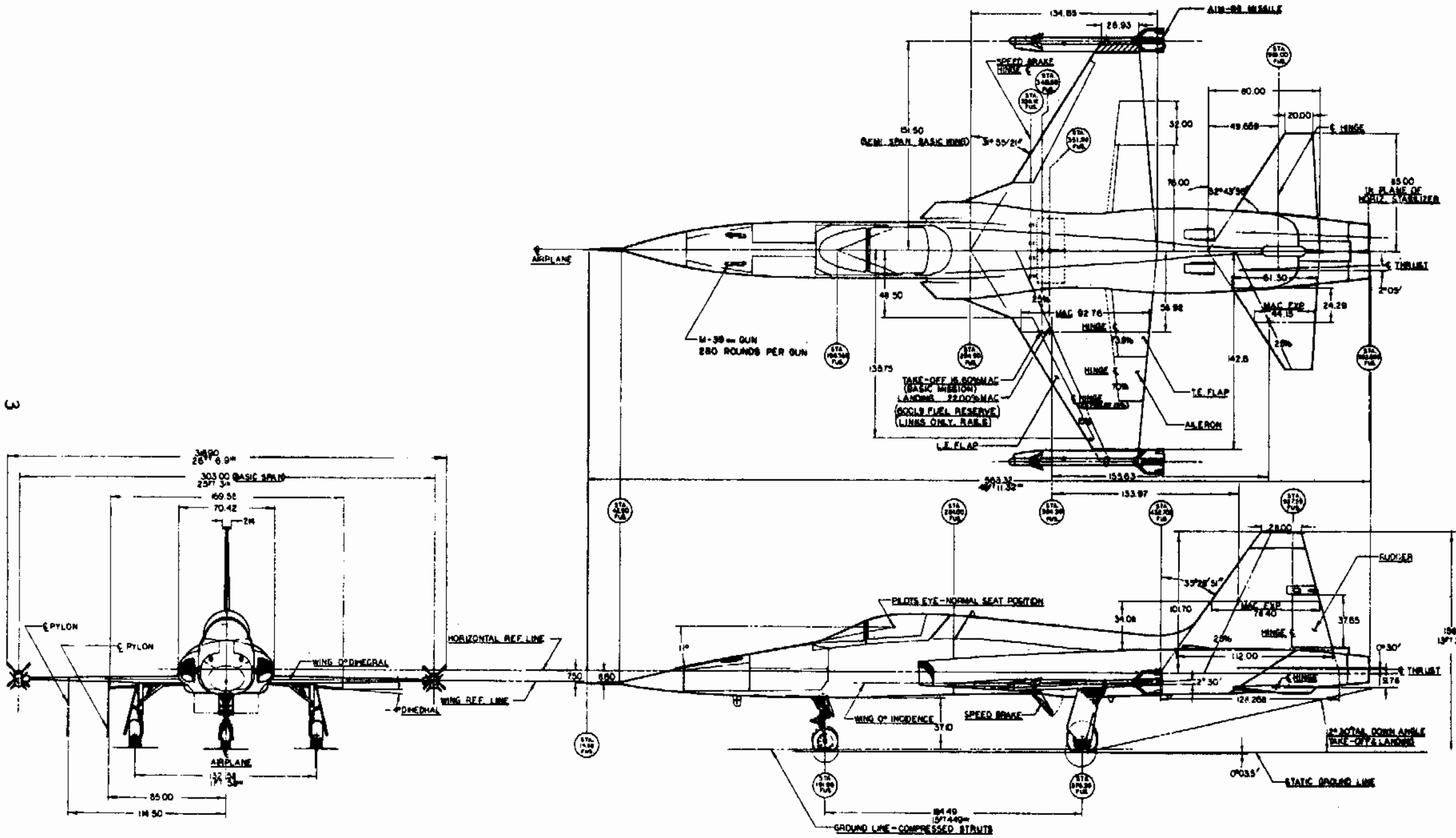
1. General Characteristics

The F-5 airplane flying qualities data provide the basis for this comparative validation of the new (7 August 1969) Flying Qualities of Piloted Airplanes Specification, MIL-F-8785B (ASG).

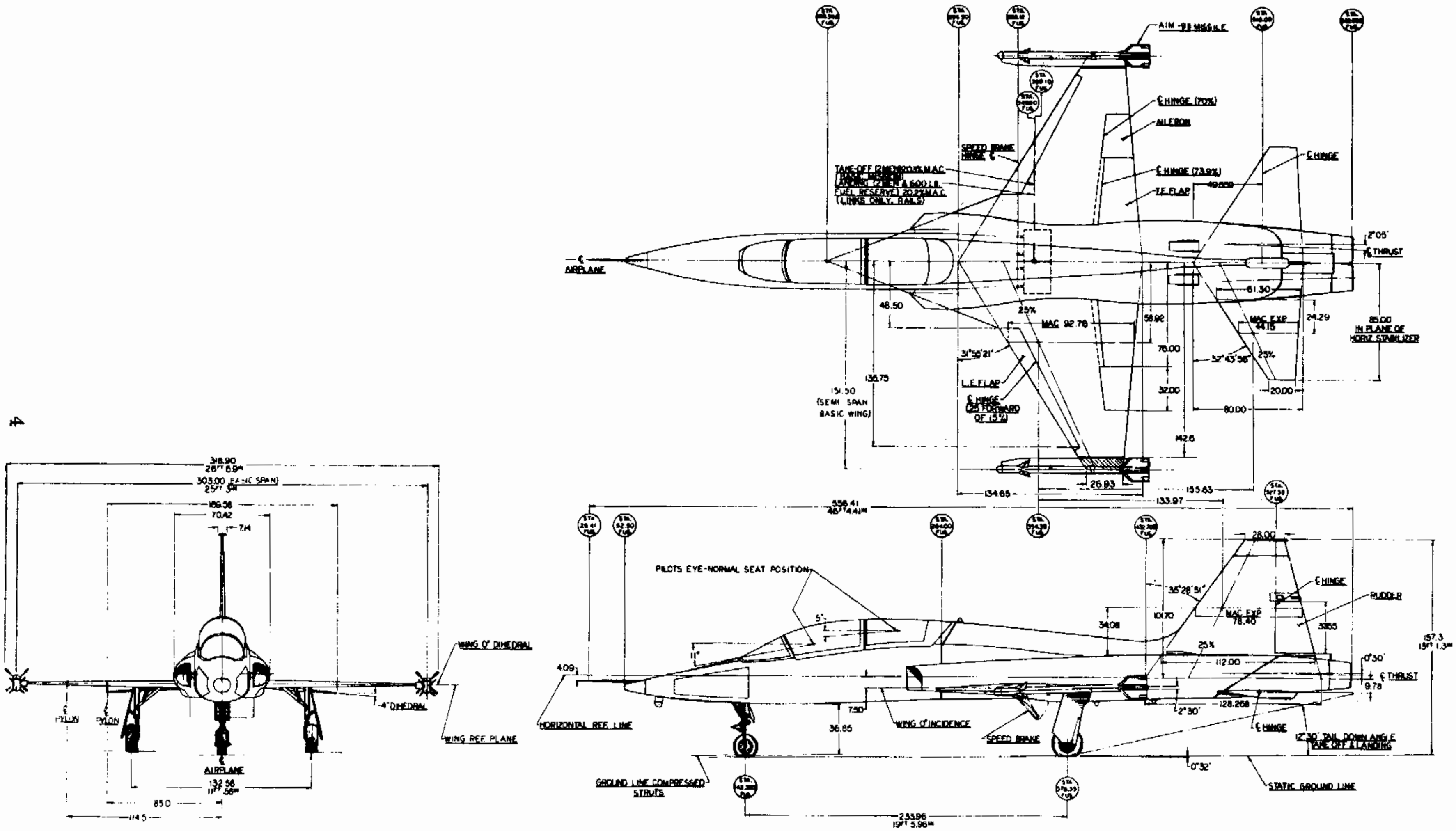
The F-5 series aircraft (single-place F-5A and two-place F-5B) was conceived and developed by Northrop Corporation to provide a high-performance, Class IV tactical fighter airplane. The F-5 reflects a trend-reversing concept in tactical fighter design, countering a trend toward increased complexity and cost that has been developing since the advent of the turbojet engine. Using two small, high-thrust-to-weight-ratio engines and advanced technologies, the F-5 airplanes combine high performance, low initial cost, low operating cost and minimum logistics requirements.

The flying qualities of the F-5 fighter and Northrop T-38 trainer are similar. For this comparative validation, and where pertinent data are not available from the F-5 but are available from the T-38, then these data are used and appropriately labeled. Over two thousand aircraft of the F-5 family are currently in service in the United States and fifteen foreign countries. Over three and one-half million flying hours have been logged to date, with one of the lowest accident rates and the lowest maintenance index of any supersonic aircraft in worldwide service.

Table 1.1 presents the general dimensional data and Figures 1.1 and 1.2 show general arrangement three-view drawings for the F-5A and F-5B, respectively. Basic performance data, based on the results of flight test programs conducted by Northrop and the U.S. Air Force are presented in Table 1.2.



F-5A THREE VIEW DRAWING
FIGURE 1.1



F-5B THREE VIEW DRAWING
FIGURE 1.2

SECTION	DETAIL	UNIT
WING	AREA TOTAL (INCLUDING AILERONS, FLAPS, 49.1 FT^2 OF FUSELAGE, AND EXPOSED LEADING EDGE EXTENSION)	173.82 FT^2
	AREA BASIC (INCLUDING AILERONS, FLAPS, 49.1 FT^2 OF FUSELAGE BUT EXCLUDING LEADING EDGE EXTENSION)	170.00 FT^2 (used for reference as in calculating C_L)
	TAPER RATIO – BASIC WING	0.20
	ASPECT RATIO – BASIC WING (SPAN 25 FT 3 IN. AREA 170 FT^2)	3.75
	SWEEPBACK AT 25% CHORD	24°
	AIRFOIL SECTION	NACA 65A-004.8 MODIFIED
	FLAP AREA – TRAILING EDGE (TOTAL)	19.00 FT^2
	FLAP AREA – LEADING EDGE (TOTAL)	12.30 FT^2
	FLAP MOVEMENT:	
	LEADING EDGE (ROOT)	23° DOWN
	TRAILING EDGE	20° DOWN
	AILERON AREA – AFT OF HINGE – PER AILERON	4.62 FT^2
	AILERON MOVEMENT:	
	GEAR DOWN	35° UP 25° DOWN
	GEAR UP	18.5° UP 14° DOWN
HORIZONTAL TAIL	AREA TOTAL	59.0 FT^2
	AREA EXPOSED	33.03 FT^2
	TAPER RATIO (EXPOSED)	0.33
	ASPECT RATIO (EXPOSED)	2.88
	SWEEPBACK AT 25% CHORD	25°
	AIRFOIL SECTION	NACA 65A-004
	SURFACE MOVEMENT TRAILING EDGE	17° UP 5.5° DOWN
VERTICAL TAIL	AREA EXPOSED	41.42 FT^2
	TAPER RATIO (EXPOSED)	0.25
	ASPECT RATIO (EXPOSED)	1.22
	SWEEPBACK AT 25% CHORD	25°
	AIRFOIL SECTION	NACA 65A-004 MODIFIED
RUDDER	AREA – AFT OF HINGE	6.10 FT^2
	MOVEMENT: (MAXIMUM)	30° RIGHT 30° LEFT
SPEED BRAKE	AREA TOTAL	6.42 FT^2
	SURFACE POSITION (MAX DOWN)	45° (RELATIVE TO H.R.L.)
USABLE FUEL	INTERNAL VOLUME	3790 LB (583 GAL)
POWER PLANT	(2) TURBOJETS WITH AFTERBURNERS	J85-GE-13
LANDING GEAR	MAIN GEAR TIRE SIZE	22 x 8.5
	NOSE GEAR TIRE SIZE	18 x 6.5
WEIGHTS	EMPTY	8085 LB (A) / 8361 LB (B)
	T.O. WEIGHT (LAUNCHER RAILS)	13,347 LB (A) / 12,731 LB (B)
	(AIM-9B CONFIGURATION)	13,677 LB (A) / 13,061 LB (B)
	(MAXIMUM GROSS WEIGHT)	20,677 LB (A) / 20,500 LB (B)

F-5A/B GENERAL DATA
TABLE 1.1

PERFORMANCE ITEM	F-5A	F-5B 1 MAN CREW
TAKEOFF GROSS WEIGHT – POUNDS	13,347	12,731
TAKEOFF DISTANCE – FEET	2,500	2,100
TAKEOFF SPEED – KNOTS	153	143
TIME TO CLIMB (BRAKE RELEASE TO 40,000 FEET) – MINUTES	4.3	4.0
RATE OF CLIMB (SEA LEVEL) – FEET PER MINUTE	28,700*	30,400⊛
COMBAT CEILING	50,000*	51,500⊛
MAXIMUM SPEED (36,000 FEET) – MACH NUMBER	1.40*	1.34⊛
MAXIMUM SPEED (SEA LEVEL) – MACH NUMBER	1.0*	0.99⊛
LANDING WEIGHT (LANDING FUEL RESERVES, AMMO EXPENDED) – POUNDS	9,931	9,619
LANDING DISTANCE – FEET	2,270	2,200
LANDING SPEED – KNOTS	131	129

* F-5A AT 11,452 POUNDS WITH 50% FUEL AND LAUNCHER RAILS, SERVICE CEILING 50,500 FT

⊛ F-5B AT 10,836 POUNDS WITH 50% FUEL AND LAUNCHER RAILS, SERVICE CEILING 52,000 FT

PERFORMANCE SUMMARY
(LAUNCHER RAIL CONFIGURATION)

TABLE 1.2

2. Flight Controls

Primary flight controls include ailerons, rudder and an all-movable horizontal stabilizer. The control system incorporates a system of springs and bobweights to provide the pilot with an "artificial feel". Conventional control stick and rudder pedals are used to operate the flight control system through cables and pushrods to the servo-valves which control the actuating cylinders. The control sticks and rudder pedals in the front and rear cockpits of the F-5B are mechanically interconnected. Two-axis (pitch and yaw) stability augmentation is provided, for both F-5A and F-5B aircraft.

The flight-control surfaces are operated by hydraulic actuators which are controlled by integral servovalves in response to manual commands by the pilot through the mechanical control system.

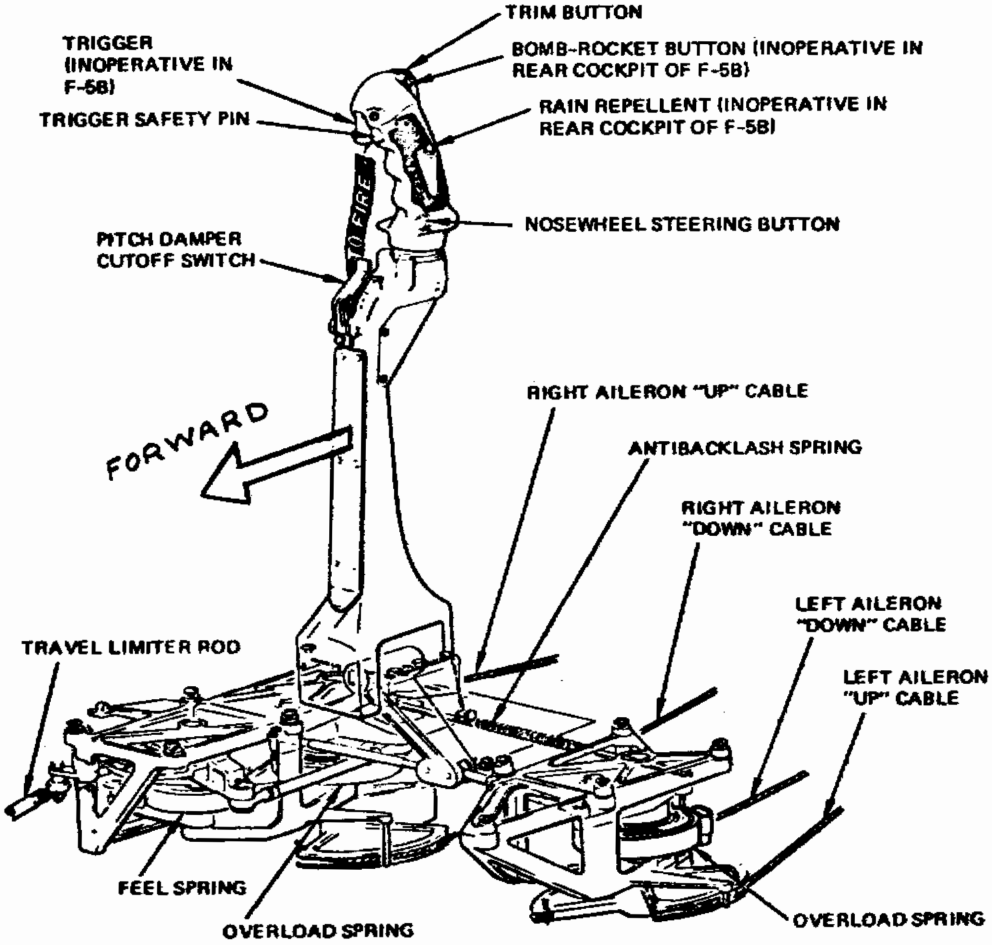
Hydraulic power for operation of the primary flight control actuators is provided through separate transmission lines by both the flight control and utility hydraulic systems. Two single actuators are provided at the rudder, one operated by each system. A dual actuator is provided at each aileron and two tandem actuators are provided for the horizontal stabilizer.

In the case of these dual or tandem actuators, each hydraulic system operates a separate piston in the actuator, with no fluid interflow between the two systems. The actuators incorporate integral filters and pressure-relief valves.

Secondary flight controls include the speed brakes and wing leading and trailing edge flaps.

Control Stick

The control stick shown in Figure 2.1 is equipped with a standard grip incorporating a trim button, bomb-rocket button, trigger, nosewheel steering button, a rain repellent system control button, and a pitch damper cutoff switch. The trim button provides aileron and horizontal stabilizer trim which allows the pilot to reduce control forces to a minimum.



CONTROL STICK
FIGURE 2.1

Aileron Control System

The aileron control system presented in Figure 2.2 is actuated by the control stick. Pushrods are connected through bellcranks and closed-cable systems to the valve operating-and-followup differential mechanism, mounted at the dual hydraulic actuator in each wing. Stick force is provided by an "artificial-feel" spring. Overload devices in each aileron system allow single aileron operation in case of binding in one aileron control. A centering mechanism, attached to the output quadrant in each wing, centers the aileron in case of mechanical failure in the system. Aileron trim is provided by pushing a button on the control stick grip which energizes a screwjack actuator mounted at the feel spring. An aileron limiter, mechanically positioned by retraction of the nose landing gear, provides a spring stop which limits the aileron travel to a safe value for high-speed flight. The spring stop may be overridden in an emergency. Table 2.1 lists the operational limits for the aileron control system.

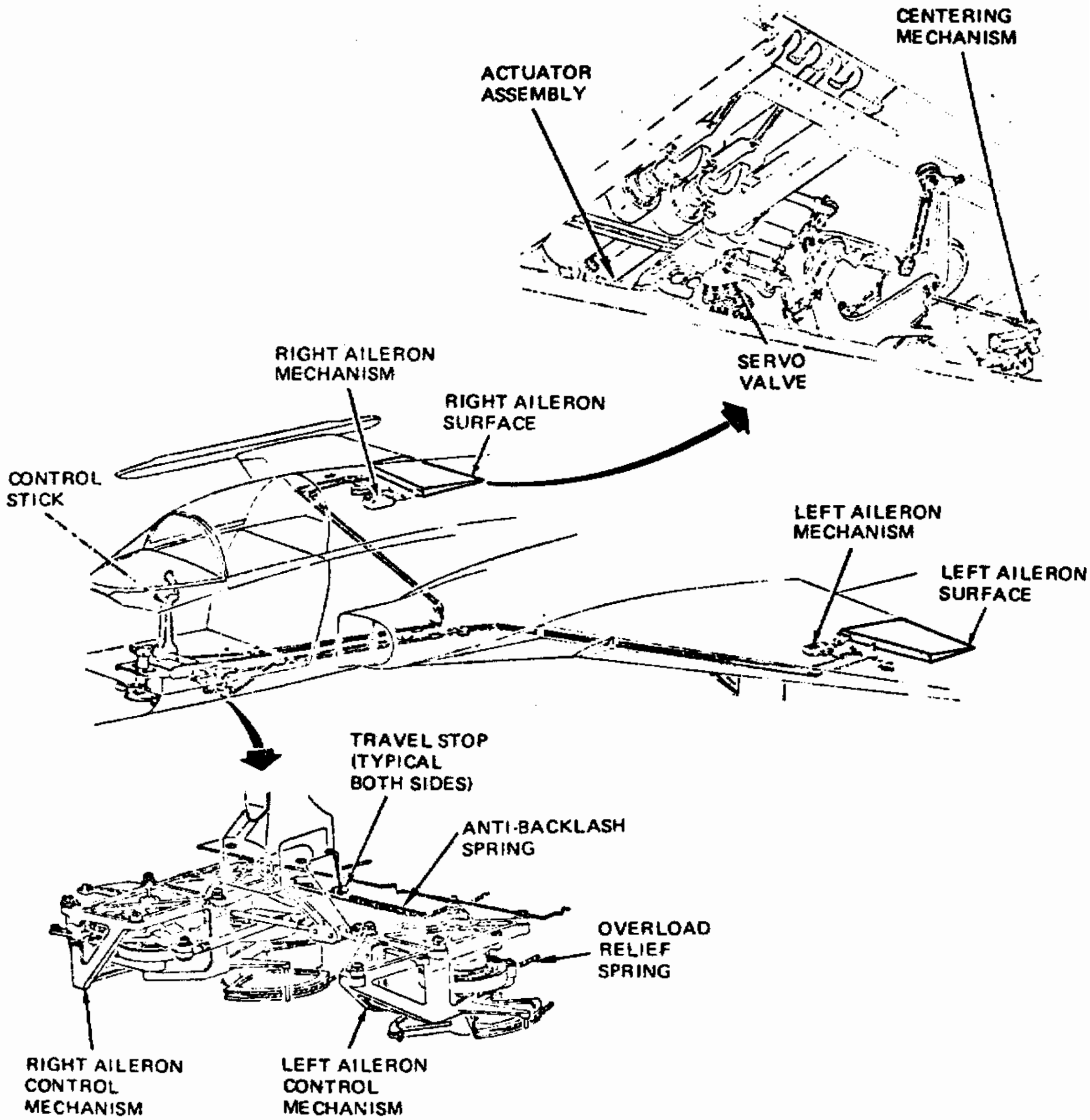
CONTROL	LIMITS
STICK TRAVEL	GEAR UP ± 3.2 INCHES EACH DIRECTION GEAR DOWN ± 4.0 INCHES EACH DIRECTION
AILERON TRAVEL	GEAR UP 18.5 DEGREES UP, 14 DEGREES DOWN GEAR DOWN 35 DEGREES UP, 25 DEGREES DOWN
AILERON TRIM	12-DEGREES DIFFERENTIAL

AILERON CONTROL LIMITS

TABLE 2.1

Stability Augmenter System

The F-5A/B aircraft are equipped with a two-axis stability augmenter system to improve damping of the longitudinal short period and the lateral-directional short period (Dutch roll) modes. The aircraft can be flown safely without augmentation if a system malfunction occurs.



AILERON CONTROL SYSTEM

FIGURE 2.2

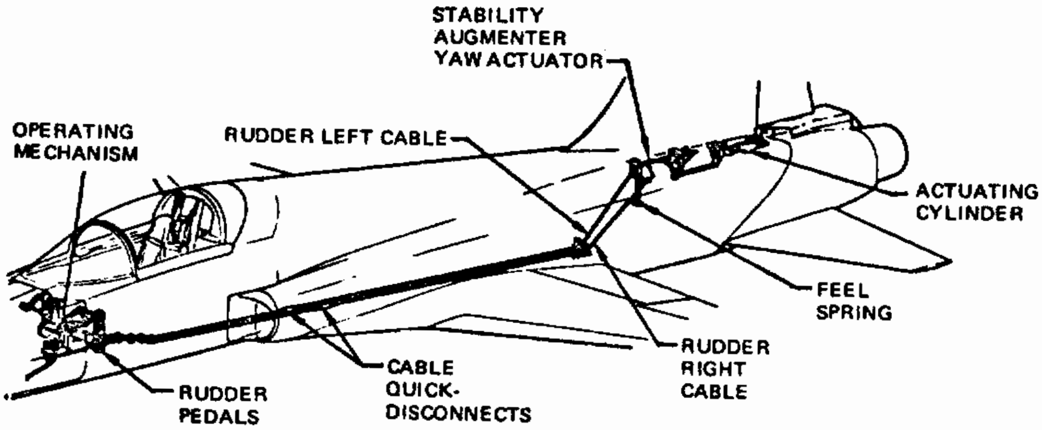
The augmenter system is engaged or disengaged by toggle switches (identified as PITCH and YAW) located on the left console in the cockpit (front cockpit only of the F-5B). The pitch axis of the augmenter can also be disengaged by an actuating lever mounted on the stick, forward of and below the grip. The yaw axis of the augmenter also provides rudder trim capability by using the control knob located adjacent to the toggle switches on the control panel.

Two identical rate gyros are used to sense aircraft body axis angular rates, one about the vertical axis for yaw and another about the transverse axis for pitch. Signals from the rate gyros are conditioned through their respective shaping networks which have the $\frac{\tau S}{\tau S + 1}$ transfer functions. These signals are then summed with the servoactuator position feedback signals which in turn are amplified to drive the servoactuators. Gain scheduling with compressible dynamic pressure is accomplished through an airspeed compensator connected to the pitot-static system.

The augmenter uses limited-authority electrohydraulic servoactuators that assume a neutral center position on disengagement and can be overridden by pilot action. Each of the two servoactuators incorporates an electrohydraulic servovalve and a feedback position transducer. One actuator is in series with each mechanical control system (rudder and horizontal stabilizer) to provide stability augmentation by modifying the pilot's mechanical control inputs. When not in operation, the servoactuators act as fixed links in the mechanical controls. Hydraulic power to each servoactuator is supplied through solenoid-operated valves.

Rudder Control System

The rudder control system uses conventional rudder pedals (both cockpits of the F-5B) as shown in Figure 2.3. Pushrods are connected through bellcranks and a closed-cable system, to the dual hydraulic actuators mounted at the control surface. Pedal forces are provided by an artificial-feel type spring. A rudder pedal adjustment T-handle is located on the cockpit pedestal for repositioning the pedals. Rudder trim is obtained by electrically biasing the yaw axis stability augmenter actuator mounted in series with the control system. The trim control knob is mounted on the left console in the cockpit. Rudder authority is available up to ±30 degrees as a function of dynamic pressure. Hydraulic power to the rudder actuators is regulated at 1500 psi in both the utility and flight control hydraulic systems. Table 2.2 specifies the control limits for the rudder control system.



RUDDER CONTROL SYSTEM

FIGURE 2.3

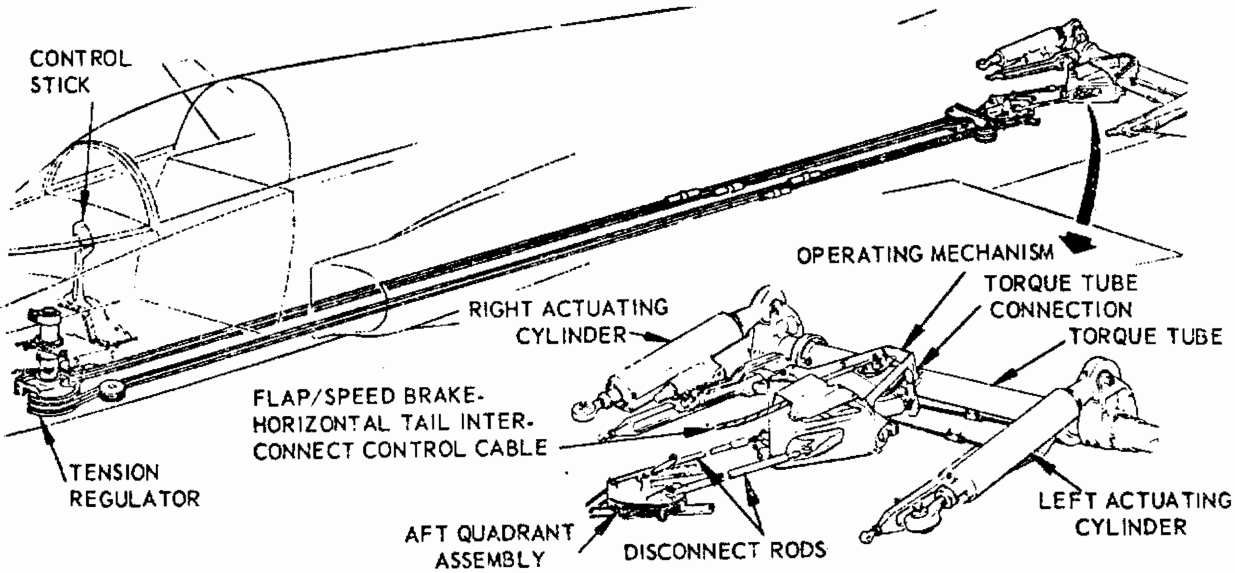
CONTROL	LIMIT
PEDAL TRAVEL (MAXIMUM)	±3.52 INCHES
RUDDER TRAVEL (MAXIMUM)	±30 DEGREES
YAW-AUGMENTER RUDDER TRAVEL	±6 DEGREES
RUDDER TRIM RANGE	±4 DEGREES

RUDDER CONTROL LIMITS

TABLE 2.2

Horizontal Stabilizer Control System

The horizontal stabilizer control system presented in Figure 2.4 is actuated by the conventional control stick (both cockpits of the F-5B) geared through pushrods, linkage, and a dual tension-regulated closed-cable system which actuates a differential mechanism mounted in the aft section of the airplane. The differential mechanism is connected through an overload bungee to a walking beam, which drives a closed-cable system interconnecting dual hydraulic tandem surface actuators. Stick forces are provided by a stick-mounted bob-weight and an artificial-feel type spring mounted in the aft section. Trim is obtained through a screwjack-type actuator. Trim is selected by use of the trim button on the control stick grip. The selected trim position is displayed on the trim indicator located on the control panel. Table 2.3 specifies the control limits for the horizontal stabilizer control system.



HORIZONTAL STABILIZER CONTROL SYSTEM

FIGURE 2.4

CONTROL	LIMIT
STICK TRAVEL	8.1 INCHES AFT, 3.0 INCHES FORWARD
HORIZONTAL TAIL TRAVEL (T.E.)	17 DEGREES UP, 5.5 DEGREES DOWN
HORIZONTAL TAIL TRIM RANGE (T.E.)	9 DEGREES UP, 0 DEGREES DOWN
PITCH AUGMENTER HORIZONTAL TAIL TRAVEL (T.E.)	1.5 DEGREES UP, 1 DEGREE DOWN

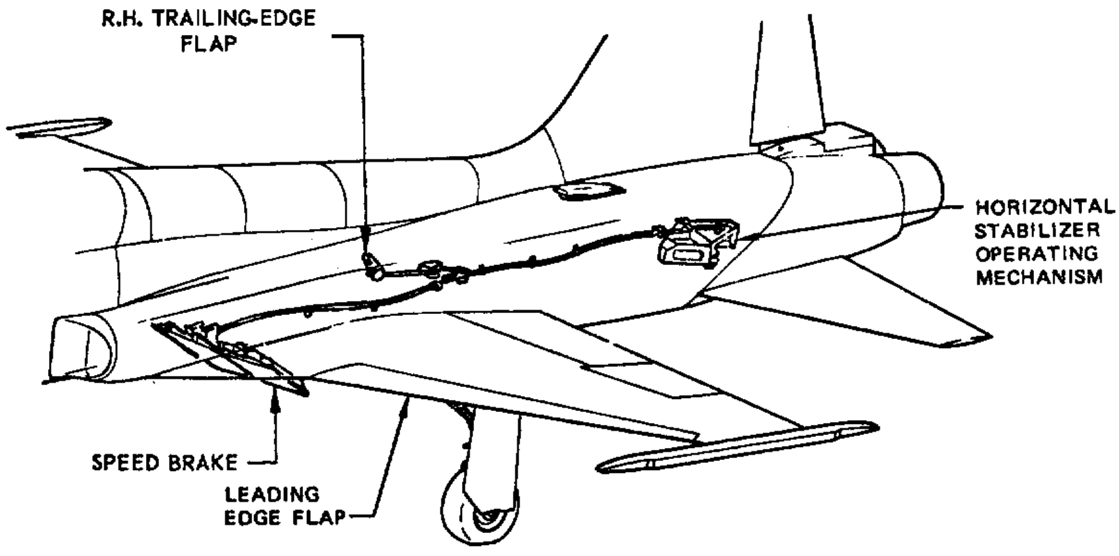
HORIZONTAL STABILIZER CONTROL SYSTEM LIMITS

TABLE 2.3

Speed Brake System

The speed brake control system consists of hydraulic-directional, relief, and rate-control valves, together with two actuating cylinders. The cockpit control consists of a three-position detented switch incorporating OPEN, CLOSE, and OFF positions and is mounted on the right-hand power control lever in the F-5A cockpit (both cockpits in the F-5B). Rear cockpit override of front cockpit position selection is provided on the F-5B. By short intermittent actuation, any degree of speed brake deflection between closed and fully-open can be selected. Maximum speed brake deflection is 45 degrees from the airplane horizontal reference line. A ground adjustment reduces the speed brake travel limit approximately 10 degrees when carrying a 2000-pound-class store on the centerline pylon. The speed brake is a variable-position type which requires approximately 4 seconds to open and 3 seconds to close. At high airspeeds the speed brake may not fully extend, but as airspeed decreases it moves out to a fully deflected position.

A mechanical system interconnects the trailing edge flaps and speed brake to the horizontal stabilizer mechanism in the aft fuselage as shown in Figure 2.5. This arrangement produces automatic pitch trim compensation. The interconnection is a series input to the horizontal stabilizer system which does not change the control stick position.



FLAP/SPEED BRAKE -- HORIZONTAL STABILIZER INTERCONNECT

FIGURE 2.5

Flap System

Leading-edge and trailing-edge flaps provide additional lift for takeoff and landing. The flap control system is shown in Figure 2.6.

Each leading-edge and trailing-edge flap is driven by an electrical actuator located at the inboard end of each flap. Left and right actuators are interconnected by rotary flex shafts to prevent asymmetry and to provide single actuator operation capability. Each actuator is driven by a 115 vac, 3-phase, 320 to 480 Hz motor. Circuits incorporated in each actuator allow normal flap operation in the event of a single actuator or circuit failure.

Mechanical stops and limit switches are provided in each leading-edge and trailing-edge actuator. The right trailing-edge flap is mechanically interconnected to the horizontal stabilizer to provide automatic trim compensation when flaps are used.

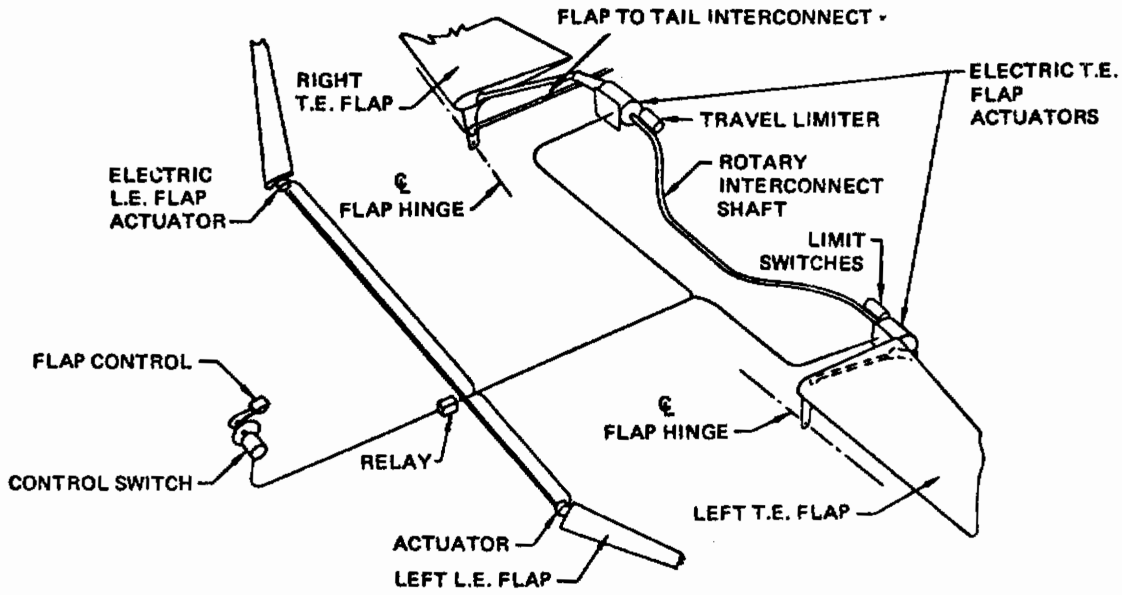
Structural twist is manufactured into each leading-edge flap; in the faired position, the outboard edge makes contact with the structure before the inboard edge. A "droop" circuit in the leading-edge flap control system limits retraction of leading edge flaps to approximately five degrees when the airplane is on the landing gear. This reduces fatigue effects in the flap structure due to the structural twist. The "droop" circuit can be bypassed for ground maintenance purposes.

A flap control lever mounted in the throttle quadrant is used to select any of three positions: leading and trailing-edge flaps up; leading-edge flaps down; leading and trailing-edge flaps down. A three-position indicator is mounted in the cockpit (both cockpits on the F-5B). The control circuit operates relays to provide power to the actuators. Table 2.4 lists the control limits for the flap-control system.

CONTROL	LIMIT
LEADING-EDGE FLAP TRAVEL	23 DEGREES
TRAILING-EDGE FLAP TRAVEL	20 DEGREES

FLAP CONTROL SYSTEM LIMITS

TABLE 2.4



FLAP CONTROL SYSTEM

FIGURE 2.6

SECTION III

REQUIREMENTS VALIDATION

This section contains the comparison of the F-5 flying qualities with the requirements of MIL-F-8785B (ASG). The paragraph numbers of the specification are used directly in this report for ease of comparison, and each paragraph will be validated individually.

Comparison Format

The comparison format will comprise four specific parts. The listing and description of possible contents of these parts are as follows:

1. Requirement

   In this part, the requirement paragraph is written exactly as it appears in the specification.

2. Comparison

   In this part, the data, qualitative and/or quantitative, are presented to compare the characteristics of the F-5 airplane with the requirements of the specification. The comparison is analyzed and a discussion presented to exhibit: (a) compliance with the specification, (b) non-compliance with the specification, or (c) disagreements (i.e., partial compliance or non-compliance may exist, or quantitatively non-compliance was exhibited but pilot qualitative comments indicate acceptable flying qualities). These conditions, if exhibited, define disagreements which need to be resolved. Other disagreements may be the result of engineering judgment regarding the feasibility, wording or purpose of the requirements. Resolution of these disagreements is to be covered in the third part of the comparison format.

3. Resolution

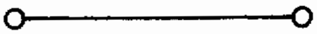
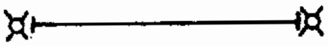
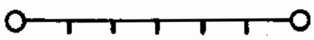
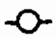
   In this part, the disagreements presented in the comparison part are resolved. Data, background information, substantiating arguments, and discussion are given in the process of resolving the disagreements. The basis for the recommendation is presented in this part.

4. Recommendation

The recommendations, if any, are given in this part. These recommendations are a result of Parts 2 and 3. If a complete rewrite of the specification paragraph is recommended, then it is so written in this part. If a partial rewrite of the specification paragraph is recommended then either the specification paragraph is rewritten with the partial changes included or changes are just indicated. If the recommendations consist of other relevances such as additional work necessary to obtain resolution, then this work is defined.

Configuration Definition

The configuration symbols used throughout this report are defined below.

CONFIGURATION SYMBOLS	DEFINITION
	Wing-tip tanks (Filled symbols mean tanks full)
	Wing-tip missile launcher rails
	Wing-tip missile launcher rails with missiles
	Wing-tip tanks (filled symbols mean tanks full) with pylons at all four wing stations and on the fuselage centerline
	Wing-tip tanks with stores at all external pylon stations
	Unfinned external pylon store
	Finned bomb
	Underwing or centerline fuel tank (filled symbol means tank full)

Data Symbol Definition

For data plots including data with stability augmenters both on and off, "augmenters on" will be identified by clear symbols and "augmenters off" will be identified by filled symbols.

Axis System

All flight test and analytical data presented in this report are in the body axis system, referred to the horizontal reference line and airplane centerlines of Figures 1.1 and 1.2.

List of Conclusions

The results of comparing the F-5 airplane flying qualities with the requirements of this specification yielded the following general conclusions:

1. The specification represents an outstanding improvement over past specifications with regards to requirements definition, paragraph organization and overall clarity.

2. The two most pertinent new requirements need to be expanded for more comprehension. First, the "Airplane Failure States" should include guidelines and sample approaches to provide evaluation methods for contractor guidance when comparing or designing airplanes to this specification. Second, the "Atmospheric Disturbances" requirements should be defined and should include quantitative values for compliance levels.

3. Most of the requirements have been basically validated, although additional work is necessary (see index of recommendations) to acquire data, preferably through experimentation or flight testing for completeness of specification validation which is somewhat lacking due to obvious limitations of previously acquired data.

INDEX OF RECOMMENDATIONS

The following lists the paragraphs of the specification indicating by page number where recommendations have been made. The most significant recommendations are marked by an asterisk.

Requirement

Paragraph 1. SCOPE AND CLASSIFICATIONS

Paragraph 1.1 Scope. This specification contains the requirements for the flying qualities of U.S. military piloted airplanes.

Comparison

The F-5 was designed to MIL-F-8785 (ASG), Amendment -2, 17 October 1955. Therefore, complete compliance with MIL-F-8785B is not possible.

Resolution

None

Recommendation

None

Requirement

Paragraph 1.2 Application. The requirements of this specification shall be applied to assure that no limitations on flight safety or on the capability to perform intended missions will result from deficiencies in flying qualities. The flying qualities for all airplanes proposed or contracted for shall be in accordance with the provisions of this specification unless specific deviations are authorized by the procuring activity. Additional or alternate special requirements may be specified by the procuring activity.

Comparison

None

Resolution

Last part of the sentence next to last and the last sentence indirectly, although without intention, suggest deviations to the specification. This is considered to be not relevant to the specification and therefore does not contribute to it and, in fact, might mislead. If deviations are necessary, they will be spontaneously initiated.

Recommendation

It is suggested that the paragraph be revised to read as follows:

"The requirements of this specification shall be applied to assure that no limitations on flight safety or on the capability to perform intended missions will result from deficiencies in flying qualities. The flying qualities for all airplanes proposed or contracted for shall be in accordance with the provisions of this specification. Additional or alternate special requirements may be specified by the procuring activity."

Requirement

Paragraph 1.3 Classification of airplanes. For the purpose of this specification, an airplane shall be placed in one of the following Classes:

Class I — Small, light airplanes such as
- Light utility
- Primary trainer
- Light observation

Class II — Medium weight, low-to-medium maneuverability airplanes such as
- Heavy utility/search and rescue
- Light or medium transport/cargo/tanker
- Early warning/electronic countermeasures/airborne command, control, or communications relay
- Antisubmarine
- Assault transport
- Reconnaissance
- Tactical bomber
- Heavy attack
- Trainer for Class II

Class III — Large, heavy, low-to-medium maneuverability airplanes such as
- Heavy transport/cargo/tanker
- Heavy bomber
- Patrol/early warning/electronic countermeasures/airborne command, control, or communications relay
- Trainer for Class III

Class IV — High-maneuverability airplanes such as
- Fighter/interceptor
- Attack
- Tactical reconnaissance
- Observation
- Trainer for Class IV

The procuring activity will assign an airplane to one of these Classes, and the requirements for that Class shall apply. When no Class is specified in a requirement, the requirement shall apply to all Classes. When operational missions so dictate, an airplane of one Class may be required by the procuring activity to meet selected requirements ordinarily specified for airplanes of another Class.

Paragraph 1.3.1 Land- or carrier-based designation. The letter -L following a Class designation identifies an airplane as land-based; carrier-based airplanes are similarly identified by -C. When no such differentiation is made in a requirement, the requirement shall apply to both land-based and carrier-based airplanes.

Comparison

The Northrop F-5, a Class IV fighter airplane, is to be used in comparing its flying qualities characteristics with the requirements of MIL-F-8785B (ASG). Description and geometric drawing of the F-5 appear in Section II of this report.

Resolution

None

Recommendation

None

Requirement

1.4 Flight Phase Categories. The Flight Phases have been combined into three Categories which are referred to in the requirement statements. These Flight Phases shall be considered in the context of total missions so that there will be no gap between successive Phases of any flight and so that transition will be smooth. When no Flight Phase or Category is stated in a requirement, that requirement shall apply to all three Categories. In certain cases, requirements are directed at specific Flight Phases identified in the requirement. Flight Phases descriptive of most military airplane missions are:

Nonterminal Flight Phases:

Category A - Those nonterminal Flight Phases that require rapid maneuvering, precision tracking, or precise flight-path control. Included in this Category are:

a. Air-to-air combat (CO)

b. Ground attack (GA)

c. Weapon delivery/launch (WD)

d. Aerial recovery (AR)

e. Reconnaissance (RC)

f. In-flight refueling (receiver) (RR)

g. Terrain following (TF)

h. Antisubmarine search (AS)

i. Close formation flying (FF)

Category B - Those nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this Category are:

a. Climb (CL)

b. Cruise (CR)

c. Loiter (LO)

d. In-flight refueling (tanker) (RT)

e. Descent (D)

f. Emergency descent (ED)

g. Emergency deceleration (DE)

h. Aerial delivery (AD)

Terminal Flight Phases:

Category C - Terminal Flight Phases are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this Category are:

a. Takeoff (TO)

b. Catapult takeoff (CT)

c. Approach (PA)

d. Wave-off/go-around (WO)

e. Landing (L)

When necessary, recategorization or addition of Flight Phases or delineation of requirements for special situations, e.g., zoom climbs, will be accomplished by the procuring activity.

Comparison

Flight test data and information on the F-5 airplane are predominant for Category A and Category C. Consequently, comparison of the F-5 airplane flying qualities with the requirements of this specification will primarily involve the following Flight Phases:

1. Air-to-air combat (CO)

2. Ground attack (GA)

3. Reconnaissance (RC)

4. Takeoff (TO)

5. Approach (PA)

6. Landing (L)

Some of the flight conditions (Mach number and altitude) that represent a Flight Phase (CO) may also apply for a Flight Phase Cruise (CR) inasmuch as the Flight Phase CO boundary encompasses Flight Phase CR. Consequently, certain flight test results, depending on the maneuvers performed, may be directly applicable to both Categories.

Resolution

None

Recommendation

None

Requirement

Paragraph 1.5 Levels of flying qualities. Where possible, the requirements of section 3 have been stated in terms of three values of the stability or control parameter being specified. Each value is a minimum condition to meet one of three Levels of acceptability related to the ability to complete the operational missions for which the airplane is designed. The Levels are:

Level 1	Flying qualities clearly adequate for the mission Flight Phase
Level 2	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists
Level 3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 2. APPLICABLE DOCUMENTS

Paragraph 2.1 The following documents, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein:

SPECIFICATIONS

Military

MIL-D-8708	Demonstration Requirements for Airplanes
MIL-F-9490	Flight Control Systems - Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems, Design, Installation and Test of, Aircraft (General Specification for)
MIL-S-25015	Spinning Requirements for Airplanes
MIL-W-25140	Weight and Balance Control Data (for Airplanes and Rotorcraft)

STANDARDS

MIL-STD-756 Reliability Prediction

(Copies of documents required by suppliers in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

Comparison

The F-5A airplane design is defined in Northrop Specification, NS 1010C, revised 2 January 1968, General Assembly, F-5A Airplane.

Resolution

None

Recommendation

None

Requirement

Paragraph 3. REQUIREMENTS

Paragraph 3.1 General requirements

Paragraph 3.1.1 Operational missions. The procuring activity will specify the operational missions to be considered by the contractor in designing the airplane to meet the flying qualities requirements of this specification. These missions will include the entire spectrum of intended operational usage.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.2 Loadings. The contractor shall define the envelopes of center of gravity and corresponding weights that will exist for each Flight Phase. These envelopes shall include the most forward and aft center-of-gravity positions as defined in MIL-W-25140. In addition, the contractor shall determine the maximum center-of-gravity excursions attainable through failures in systems or components, such as fuel sequencing, hung stores, etc., for each Flight Phase to be considered in the Failure States of 3.1.6.2. Within these envelopes, plus a growth margin to be specified by the procuring activity, and for the excursions cited above, this specification shall apply.

Paragraph 3.1.3 Moments of inertia. The contractor shall define the moments of inertia associated with all loadings of 3.1.2. The requirements of this specification shall apply for all moments of inertia so defined.

Paragraph 3.1.4 External stores. The requirements of this specification shall apply for all combinations of external stores required by the operational missions. The effects of external stores on the weight, moments of inertia, center-of-gravity position, and aerodynamic characteristics of the airplane shall be considered for each mission Flight Phase. When the stores contain expendable loads, the requirements of this specification apply throughout the range of store loadings. The external stores and store combinations to be considered for flying qualities design will be specified by the procuring activity. In establishing external store combinations to be investigated, consideration shall be given to asymmetric as well as to symmetric combinations.

Comparison

Paragraphs 3.1.2, 3.1.3 and 3.1.4 are to be considered simultaneously because of their related requirements. The data presented apply to all three paragraphs.

In order to cover the extremes in terms of c.g. travel, magnitudes of moments of inertia and weight variations, Figures 1 through 6 (3.1.3) are presented. Six configurations in terms of external store loading consisting of clean underwing through external loading of maximum underwing stores have been selected as representative loadings for the Flight Phases being considered as mentioned in the comparison part of Paragraph 1.4.

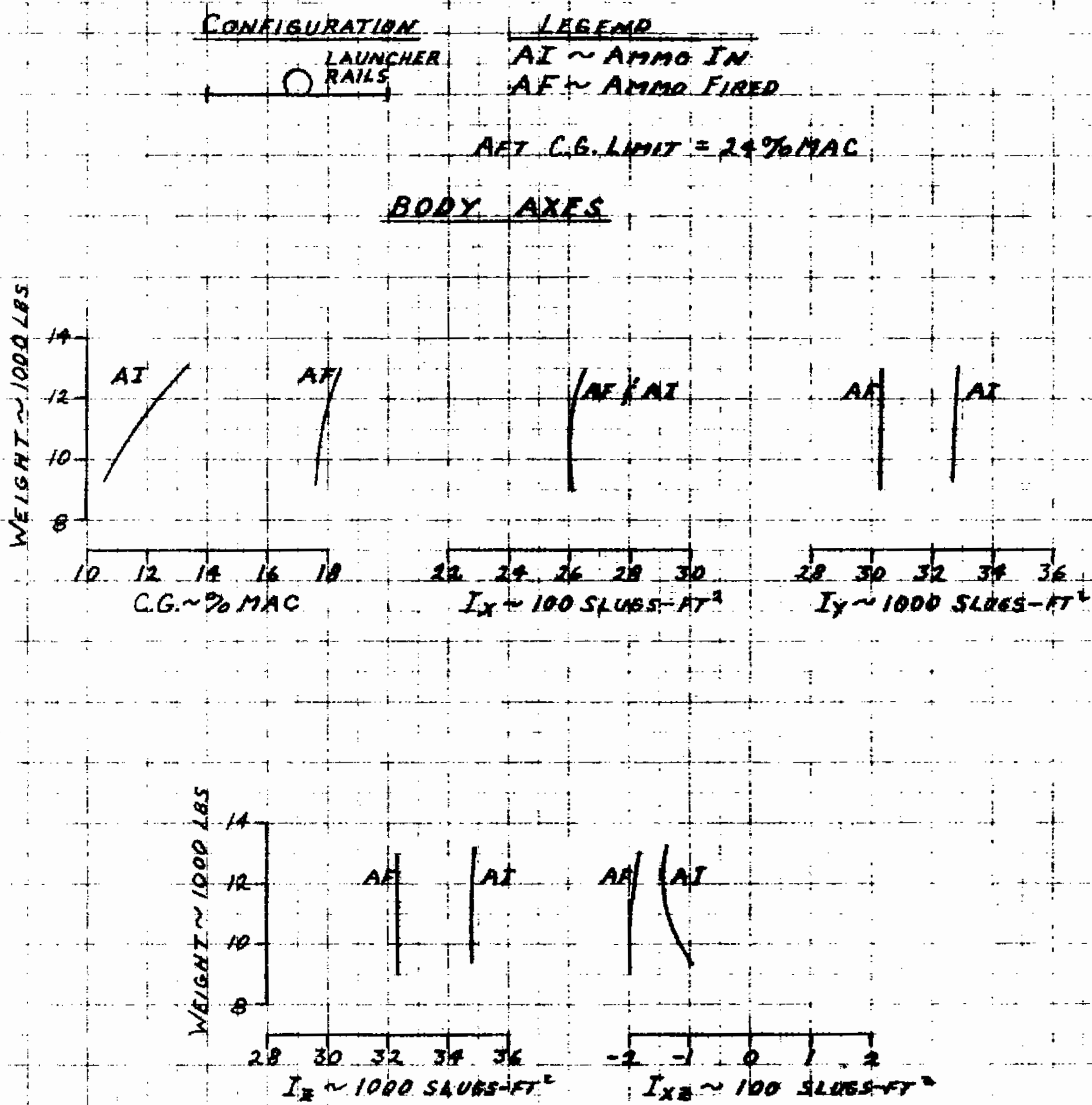
The asymmetric loadings of external stores, Figures 7 (3.1.3) through 11 (3.1.3), represent hung stores; i.e., stores that failed to jettison. Data presented in Figures 1 through 6 (3.1.3) reflect the extreme c.g. travel due to ammo in, which constitutes the full complement of ammunition for the two 20 mm cannons located in the nose of the aircraft. Ammo fired means that all the ammunition has been expended. The moments of inertia are in the body axes system.

Resolution

None

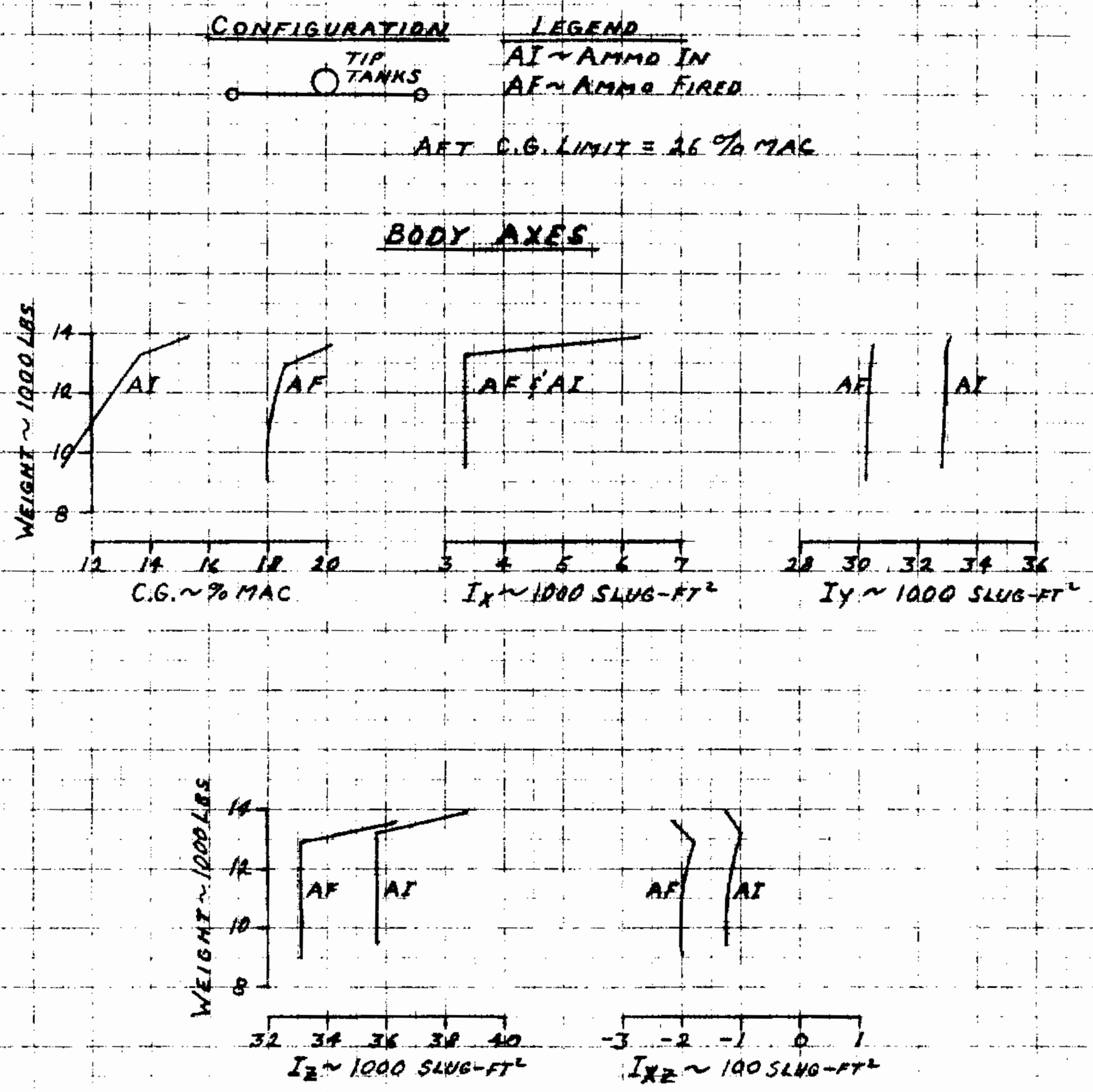
Recommendation

None



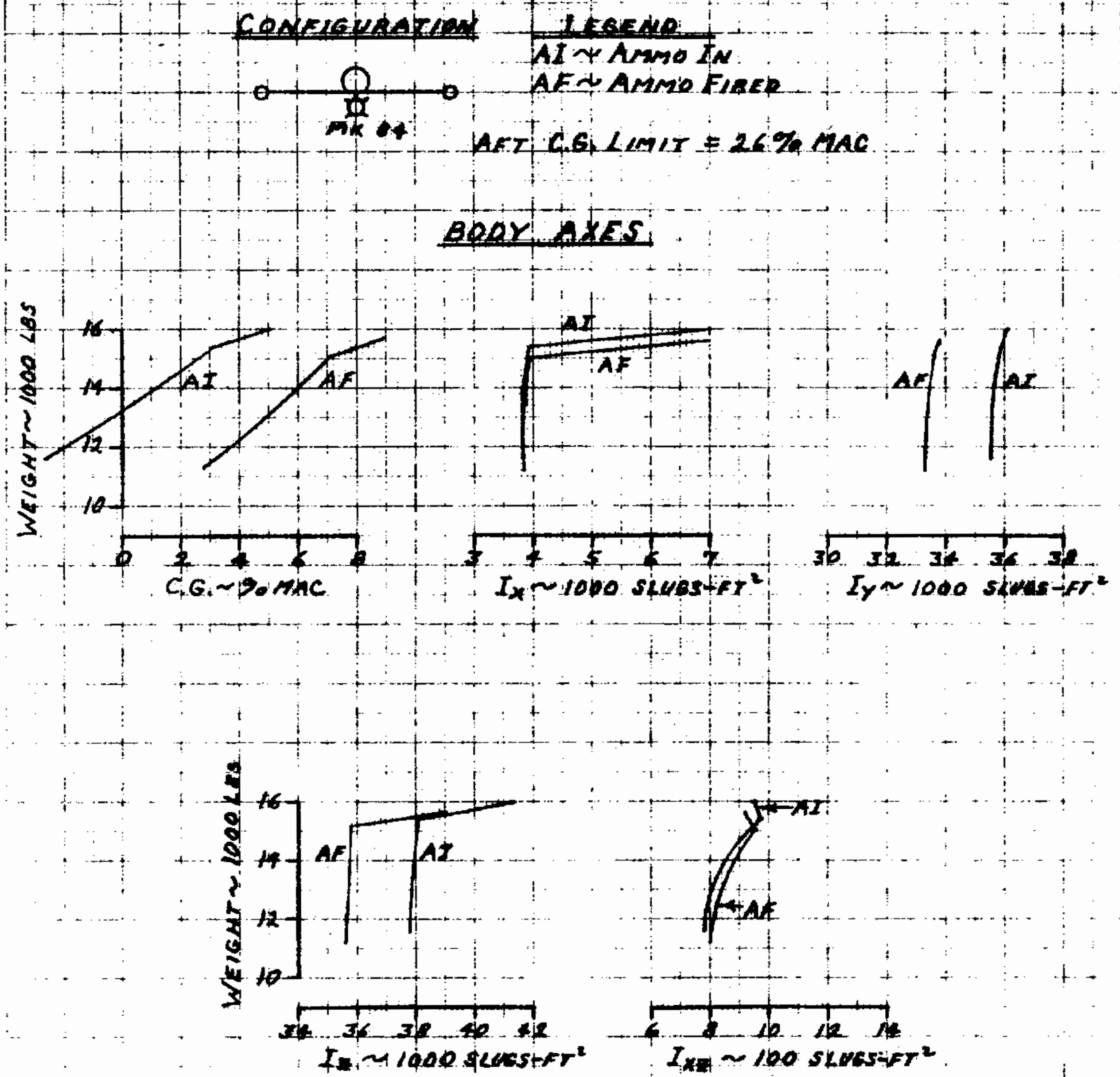
F-5A WEIGHT, C.G., & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 1 (3.1.3)



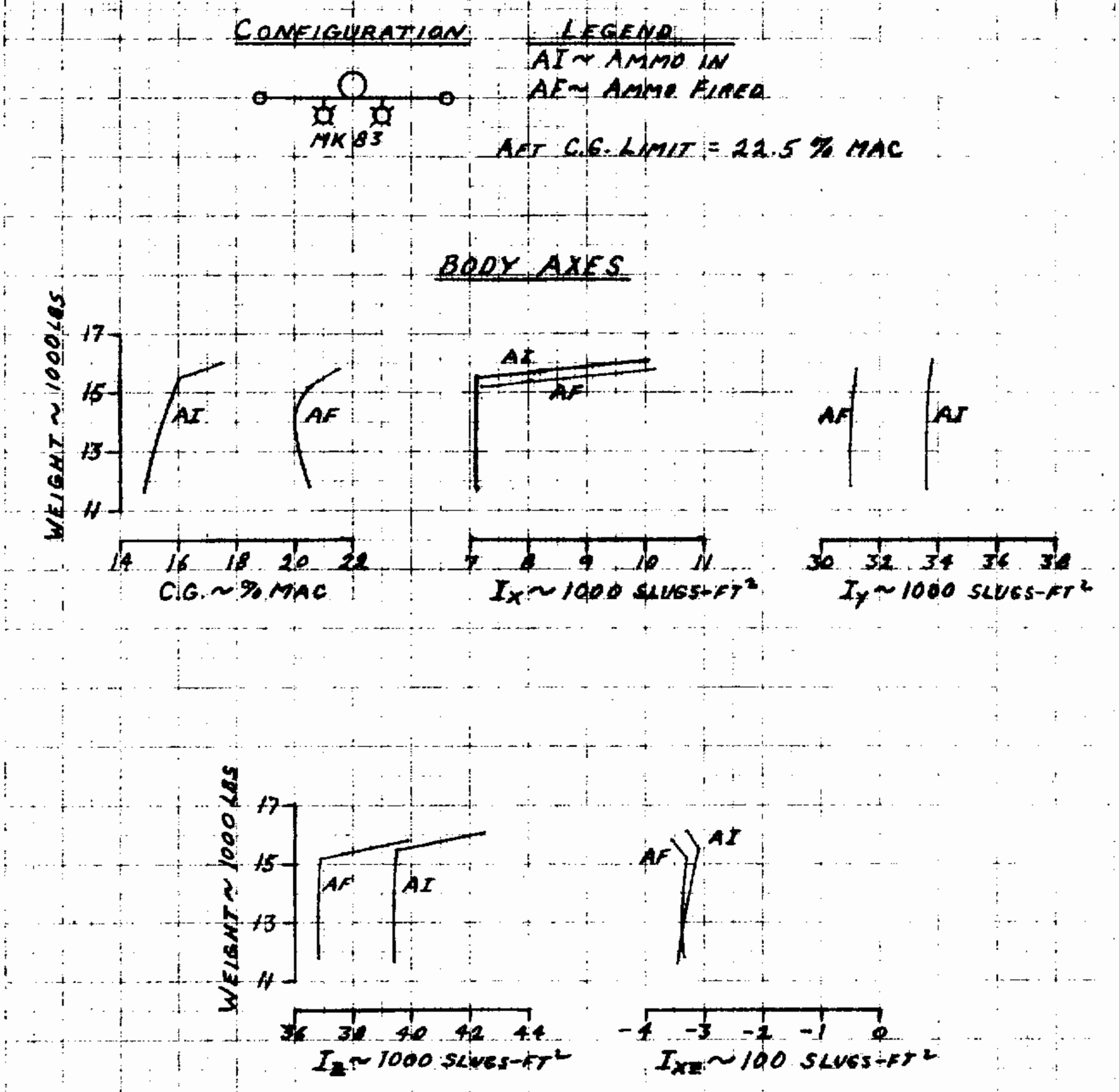
F-5A WEIGHT, C.G., & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 2 (3.1.3)



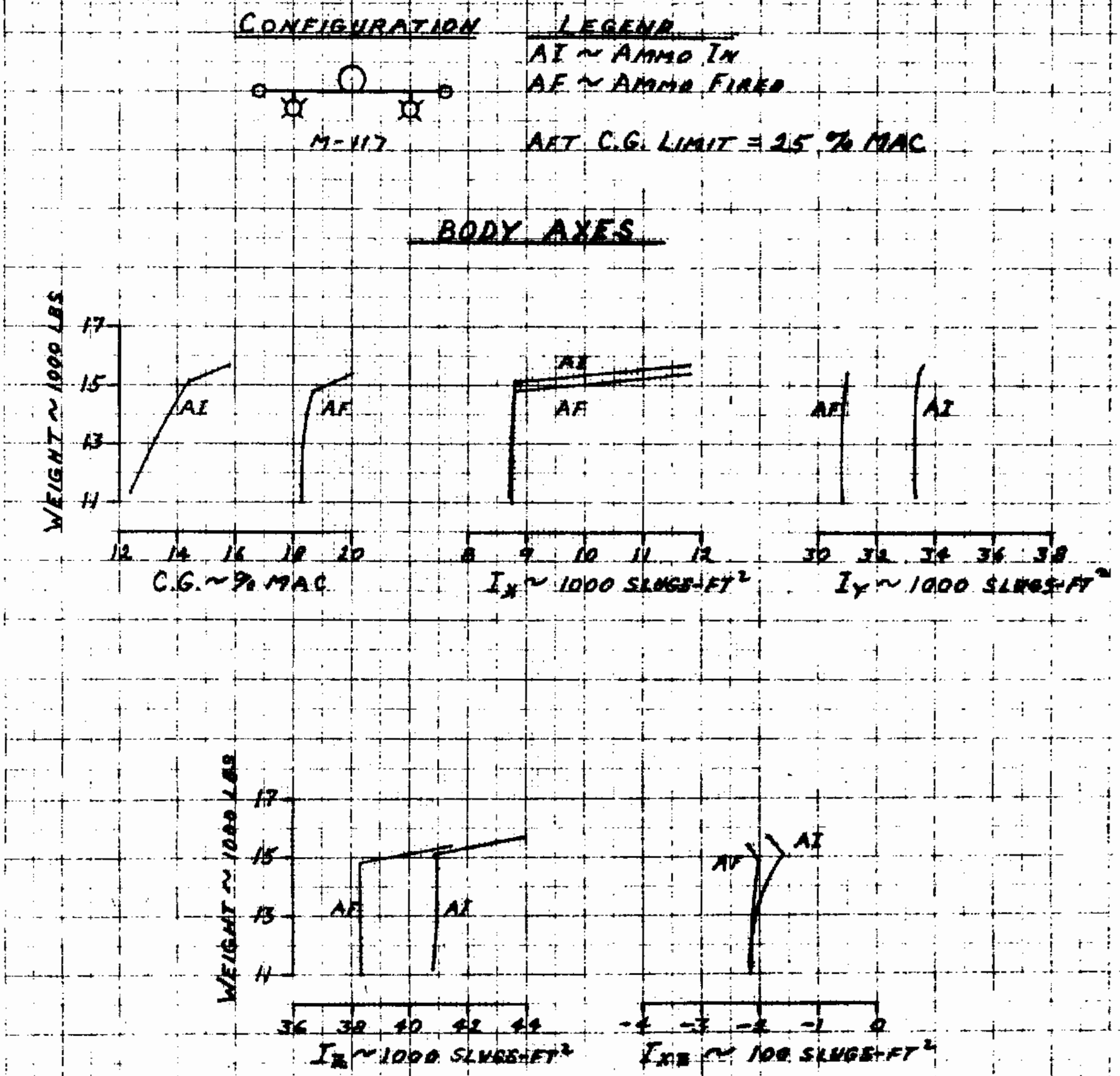
F-5A WEIGHT, C.G., & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 3 (3.1.3)



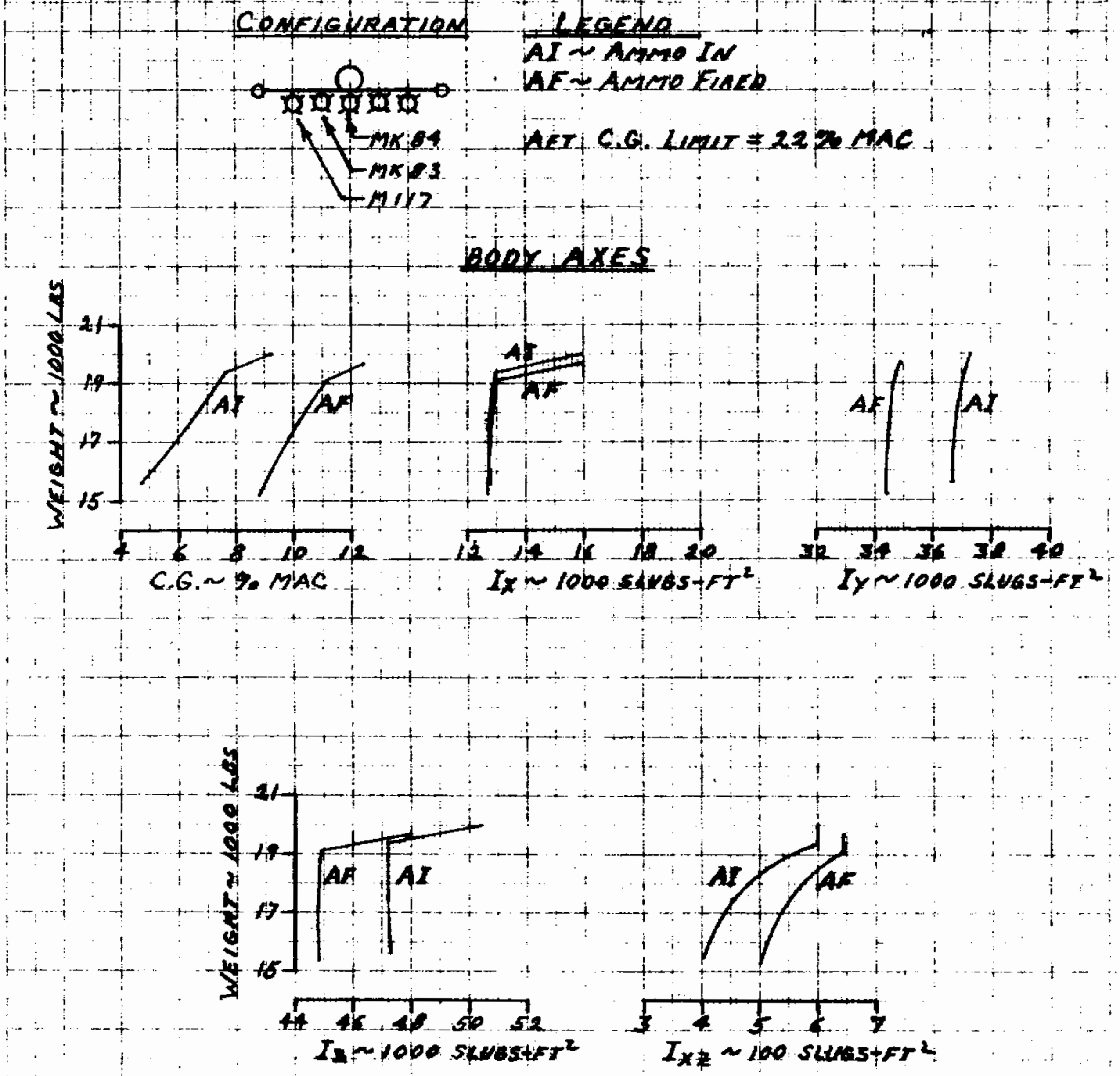
F-5A WEIGHT, C.G., & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 4 (3.1.3)



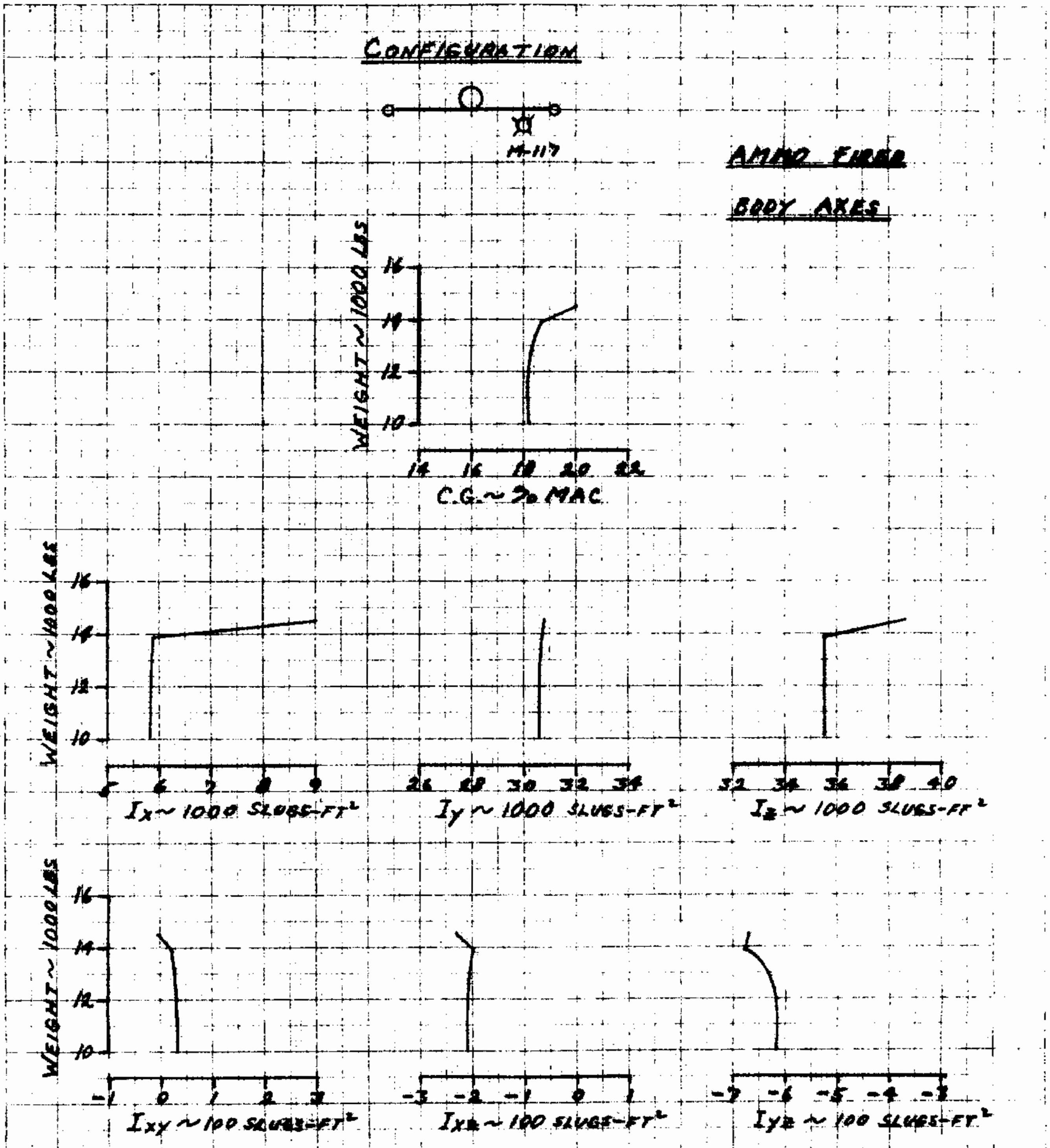
F-5A WEIGHT, C.G. & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 5 (3.1.3)



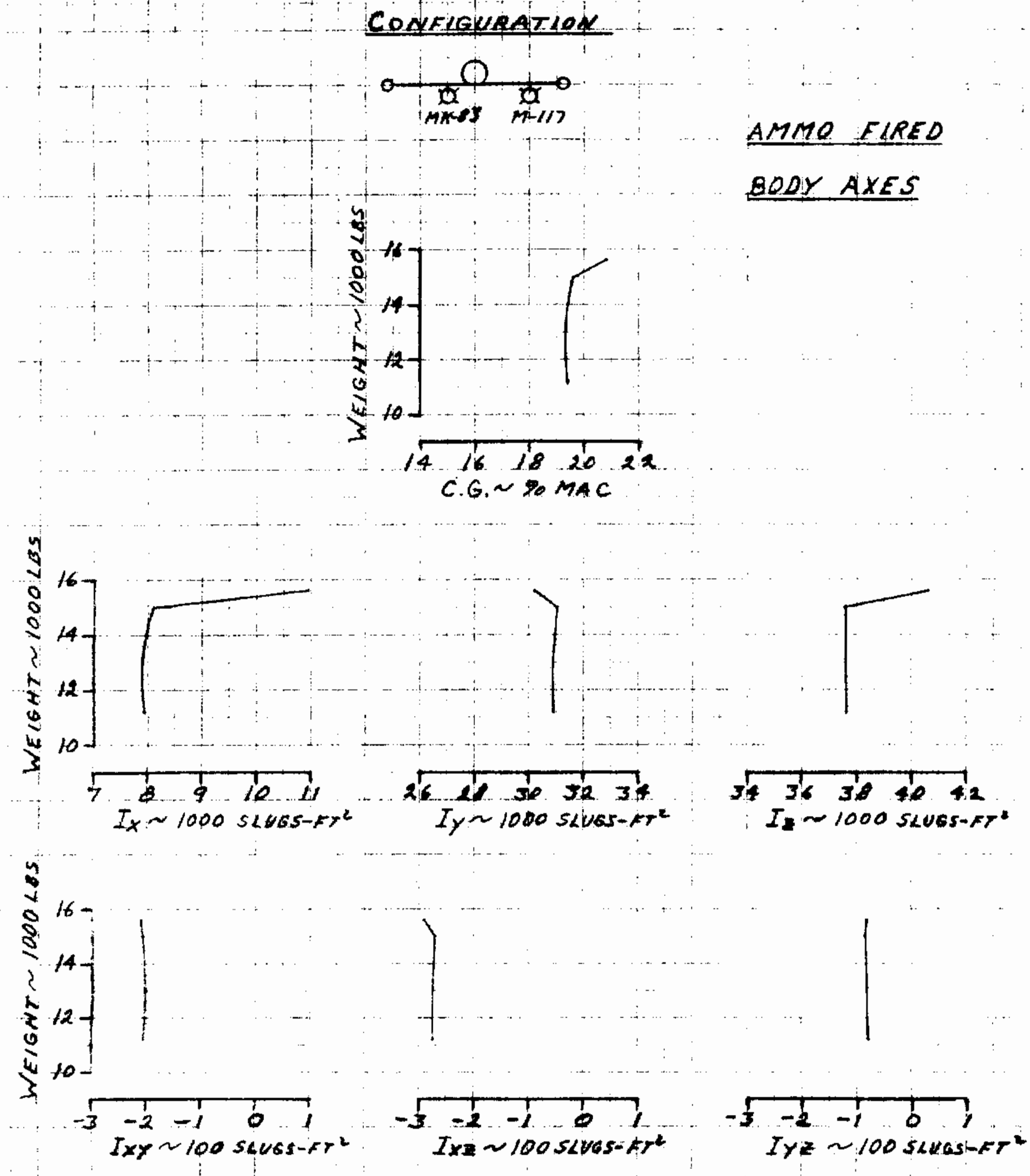
F-5A WEIGHT, C.G., & MOMENTS OF INERTIA
SYMMETRICAL LOADING

FIGURE 6 (3.1.3)



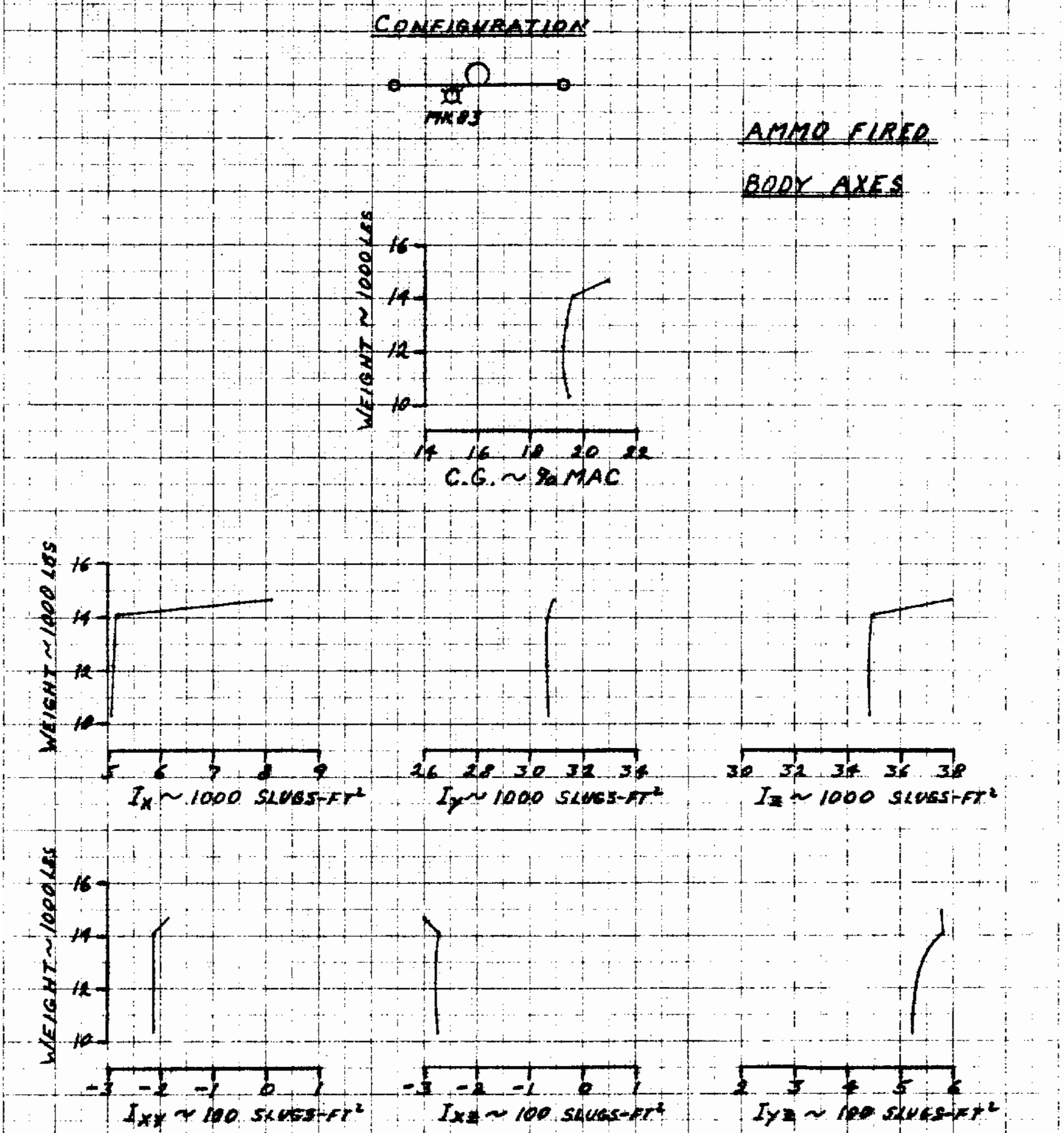
F-5A WEIGHT, C.G., & MOMENTS & PRODUCTS OF INERTIA
ASYMMETRIC LOADING

FIGURE 7 (3.1.3)



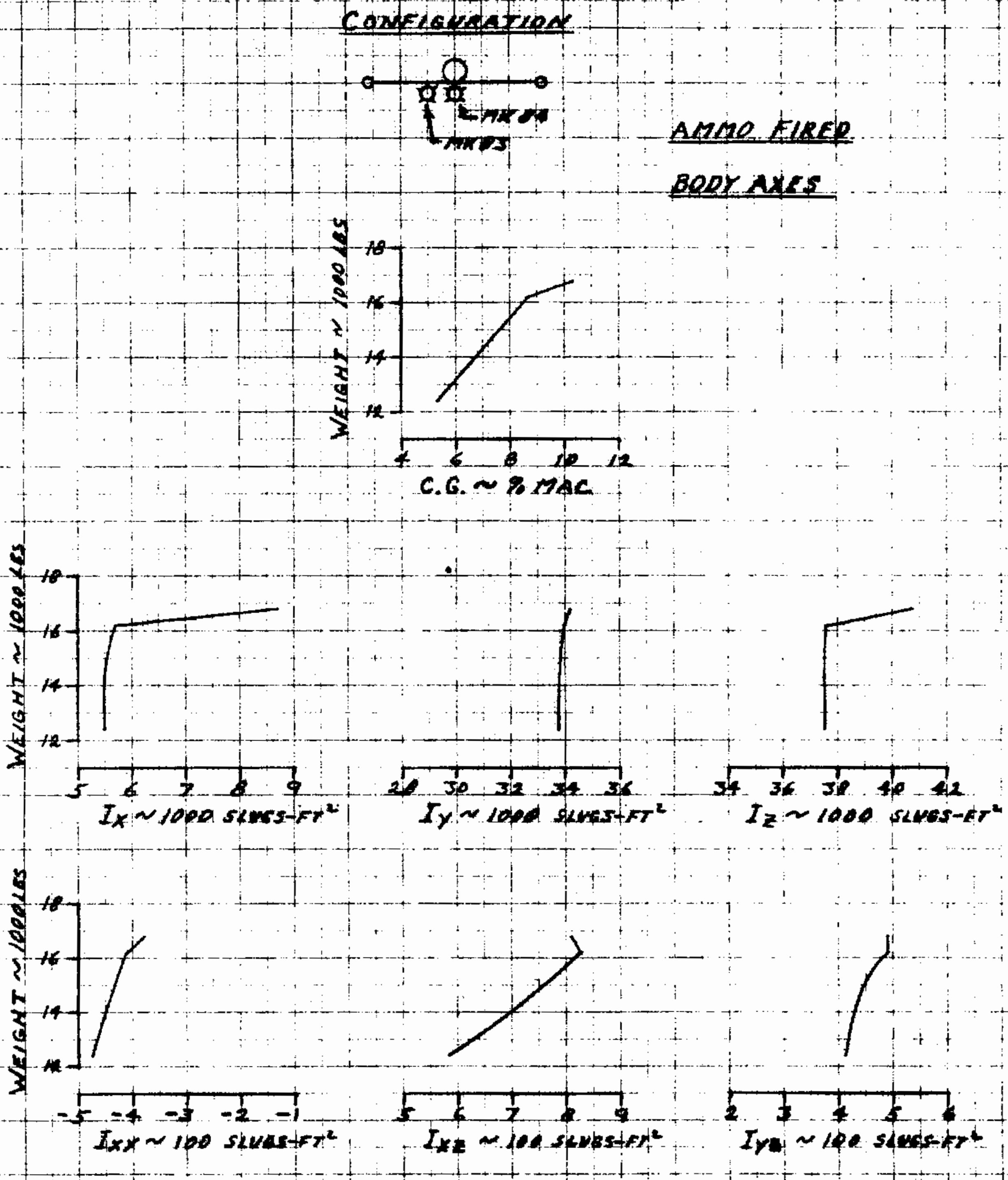
F-5A WEIGHT, C.G., & MOMENTS & PRODUCTS OF INERTIA
ASYMMETRIC LOADING

FIGURE 8 (3.1.3)



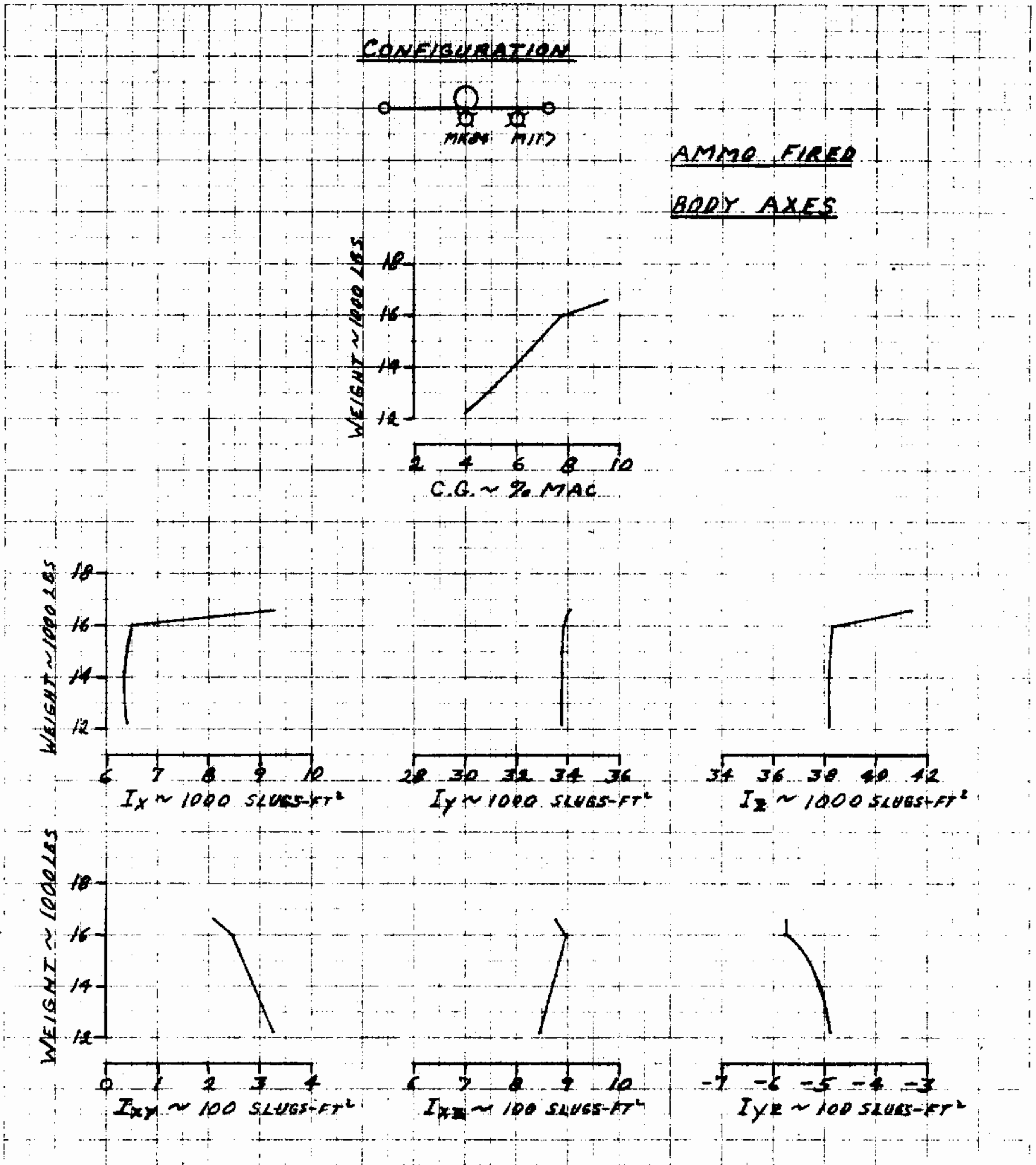
F-5A WEIGHT, C.G., & MOMENTS & PRODUCTS OF INERTIA
ASYMMETRIC LOADING

FIGURE 9 (3.1.3)



F-5A WEIGHT, C.G., & MOMENTS & PRODUCTS OF INERTIA
ASYMMETRIC LOADING

FIGURE 10 (3.1.3)



F-5A WEIGHT, C.G. & MOMENTS & PRODUCTS OF INERTIA
ASYMMETRIC LOADING

FIGURE 11 (3.1.3)

Requirement

Paragraph 3.1.5 Configurations. The requirements of this specification shall apply for all configurations required or encountered in the applicable Flight Phases of 1.4. A (crew-) selected configuration is defined by the positions and adjustments of the various selectors and controls available to the crew except for rudder, aileron, elevator, throttle and trim controls. Examples are: the flap control setting and the yaw damper ON or OFF. The selected configurations to be examined must consist of those required for performance and mission accomplishment. Additional configurations to be investigated may be defined by the procuring activity.

Paragraph 3.1.6 State of the airplane. The State of the airplane is defined by the selected configuration together with the functional status of each of the airplane components or systems, throttle setting, weight, moments of inertia, center-of-gravity position, and external store complement. The trim setting and the positions of the rudder, aileron, and elevator controls are not included in the definition of Airplane State since they are often specified in the requirements.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.6.1 Airplane Normal States. The contractor shall define and tabulate all pertinent items to describe the Airplane Normal (no component or system failure) State(s) associated with each of the applicable Flight Phases. This tabulation shall be in the format and shall use the nomenclature shown in 6.2. Certain items, such as weight, moments of inertia, center-of-gravity position, wing sweep, or thrust setting may vary continuously over a range of values during a Flight Phase. The contractor shall replace this continuous variation by a limited number of values of the parameter in question which will be treated as specific states, and which include the most critical values and the extremes encountered during the Flight Phase in question.

Comparison

Airplane Normal States for two configurations representative of two extreme loadings in terms of weight and external stores are shown in Table XVI (3.1.6.1). The air-to-air combat and the ground attack configurations are presented and appear in this order in the table.

Resolution

The two configurations shown in the comparison comprise the air-to-air combat and ground attack flight phases. The airplane is capable of carrying a great number of other external stores which represent the exhibited category and Flight Phases. In order to comply fully with the interpreted requirements of this paragraph, many more or perhaps all the possible loadings need to be tabulated. It is not considered practical to tabulate all the possible combinations of external stores loadings. Furthermore, tabulating the extremes may not necessarily fulfil the intent of the specification. Hence, the requirement of this paragraph is not sufficiently conclusive and should be reviewed to make the requirement more decisive.

Recommendation

It is recommended that the following sentence be added to the end of the paragraph.

"The external stores loadings, that represent the Categories and Flight Phases for which Airplane Normal States will be tabulated, shall be established with the guidance and approval of the procuring activity."

Flight Phases	Weight (Pounds)	C.G.* (% M.A.C.)	External Stores	High Lift Devices	Landing Gear	Speed Brakes	Stability Augmentation
Takeoff (TO)	13,160	13.4		flaps down	down	optional	on
Cruise (CR)	12,183	12.4		flaps up	up		
Air-to-Air Combat (CO)	11,450	12.0		flaps up	up		
Approach (PA)	10,422	17.7		flaps down	down		
Landing (L)	9,747	17.7		flaps down	down		
Takeoff (TO)	20,000	9.2	**	flaps down	down	optional	on
Cruise (CR)	18,421	7.0		flaps up	up		
Ground Attack (GA)	15,600	4.7		flaps up	up		
Approach (PA)	11,321	18.2		flaps down	down		
Landing (L)	10,521	18.0		flaps down	down		

*Ammo in except for (PA) and (L) Flight Phases with ammo fired
**Landing of takeoff stores configuration is a normal state (see resolution)

Airplane Normal States

TABLE XVI (3.1.6.1)

Requirement

Paragraph 3.1.6.2 Airplane Failure States. The contractor shall define and tabulate all Airplane Failure States, which consist of Airplane Normal States modified by one or more malfunctions in airplane components or systems; for example, a discrepancy between a selected configuration and an actual configuration. Those malfunctions that result in center-of-gravity positions outside the center-of-gravity envelope defined in 3.1.2 shall be included. Each mode of failure shall be considered. Failures occurring in any Flight Phase shall be considered in all subsequent Flight Phases.

Comparison

The following tabulates the Airplane Failure States and defines them. In addition, some examples of the malfunctions that create these failure states and result in their causation are respectively tabulated as follows:

FAILURE STATES, DEFINITION AND CAUSATION

1. Center-of-Gravity Outside the Center-of-Gravity Envelope.

   Definition: Failure modes that put the center-of-gravity outside of the maximum envelope or outside of the calculated c.g. for a specific mission.

   Causation: Unauthorized or incorrect stores loaded. Asymmetrical stores configuration resulting from hung store(s).

2. Aerodynamic Asymmetry.

   Definition: All such failures except for primary and secondary control systems failures that result in asymmetry.

   Causation: Structural failures such as wing-tip or door or flap lost in flight. Landing gear or door extending in flight.

3. Loss of Control Function.

Definition: Functional failures of primary and secondary flight control system components, including those caused by failures of support systems components.

Causation: Control stick and control cable system failure. Leading or trailing edge flap or speed brake fails in down or extended position. Hydraulic or electrical power supply failure resulting in functional failure of either primary or secondary flight control system. Stability augmentation system hard over failure. Servo valve/actuator failure. Misrigged control resulting in crossed control function or insufficient authority.

4. Jammed Control.

Definition: The failure of any primary or secondary flight control system function to respond to a pilot commanded input.

Causation: Slipped or incorrect cushion on seat survival kit preventing full aft control stick movement. Loosened or migrating airplane parts or foreign object creating a jammed control function.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.6.2.1 Airplane Special Failure States. Certain components, systems, or combinations thereof may have extremely remote probability of failure during a given flight. These failure probabilities may, in turn, be very difficult to predict with any degree of accuracy. Special Failure States of this type need not be considered in complying with the requirements of section 3 if justification for considering the Failure States as Special is submitted by the contractor and approved by the procuring activity.

Comparison

There are a number of Special Failure States potentially possible for the F-5 and T-38 airplanes. Examples are; control stick fracture, dual hydraulic system failure, or disintegration of both horizontal tail surfaces. None of these Special Failure States or any others have occurred in the more than 4,000,000 flying hours accumulated by over 2,000 F-5 and T-38 airplanes.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.7 Operational Flight Envelopes. The Operational Flight Envelopes define the boundaries in terms of speed, altitude, and load factor within which the airplane must be capable of operating in order to accomplish the missions of 3.1.1. Envelopes for each applicable Flight Phase shall be established with the guidance and approval of the procuring activity. In the absence of specific guidance, the contractor shall use the representative conditions of table I for the applicable Flight Phases.

Comparison

The operational missions being considered involve the air-to-air combat and the ground attack Flight Phases. These two external loadings represent the extremes between clean configuration with no underwing stores to full complement of underwing stores. The full complement of external stores covers 4 wing stations and 1 fuselage station for a total of 5 external store stations. The operational envelopes for these two external loadings appear respectively in Figure 1 (3.1.7) through Figure 4 (3.1.7). All other external store loadings will have operational envelopes that fall within these two extremes. The takeoff, cruise, approach and landing flight phases fall within the operational envelopes shown in these figures.

Resolution

In the first sentence of this paragraph, the boundaries of load factor could be interpreted as the structural limit load factor, n_L. The intention of the specification is to require construction of a V-n diagram as shown on Page 66 of MIL-F-8785B (ASG). However, the minimum wordings of the specification are not sufficient to ensure that everyone understands that V-n diagram or diagrams are required.

Recommendation

It is recommended that the first sentence be partially changed. The following is a rewrite of the first sentence only.

"The Operational Flight Envelopes define the boundaries in terms of speed, altitude, and load factor (V-n diagram or diagrams required to be constructed) within which the airplane must be capable of operating in order to accomplish the missions of 3.1.1."

It is also recommended that the following statement be added to the end of the paragraph to ensure that the flying qualities integrity of the airplane is maintained from the onset of design and development through flight. This will provide an airplane that the procuring activity will be pleased with and that the contractor will be proud of, resulting in mutual satisfaction and understanding, thus eliminating the need for future ill-wanted deviations from specifications as have been historically practised.

"The Operational Flight Envelopes will be constructed by the contractor and approved by the procuring acitivity at the onset of design and development of the airplane. Compliance with all applicable requirements will be in accordance with these Envelopes."

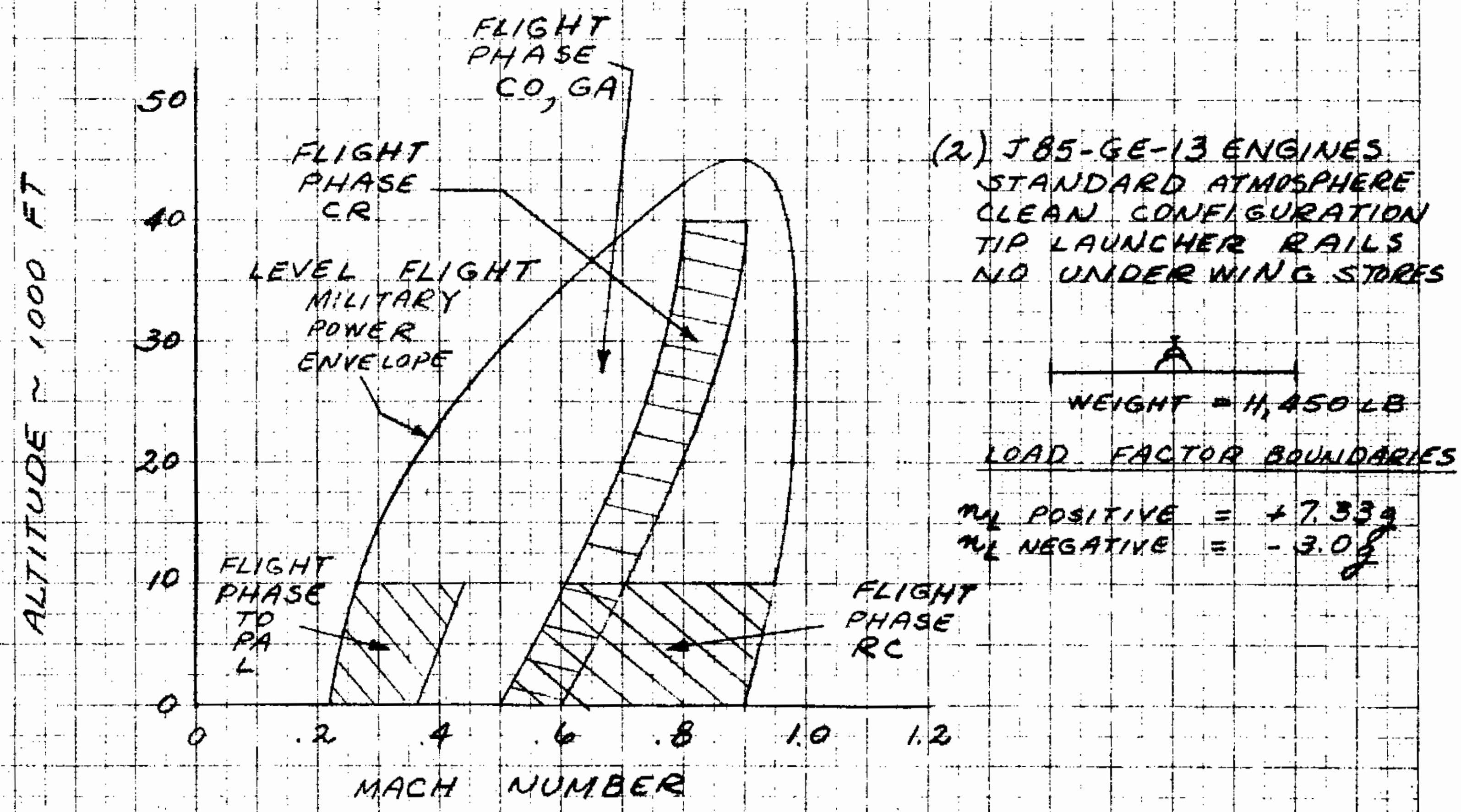
TABLE I. Operational Flight Envelope

FLIGHT PHASE CATEGORY	FLIGHT PHASE	AIRSPEED		ALTITUDE		LOAD FACTOR	
		$V_{o_{min}}$ ($M_{o_{min}}$)	$V_{o_{max}}$ ($M_{o_{max}}$)	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
	AIR-TO-AIR COMBAT (CO)	1.4 V_S	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L
	GROUND ATTACK (GA)	1.3 V_S	V_{MRT}	MSL	Medium	-1.0	n_L
	WEAPON DELIVERY/LAUNCH (WD)	V_{range}	V_{MAT}	MSL	Combat Ceiling	.5	*
A	AERIAL RECOVERY (AR)	1.2 V_S	V_{MRT}	MSL	Combat Ceiling	.5	n_L
	RECONNAISSANCE (RC)	1.3 V_S	V_{MAT}	MSL	Combat Ceiling	*	*
	IN-FLIGHT REFUEL (RECEIVER) (RR)	1.2 V_S	V_{MRT}	MSL	Combat Ceiling	.5	2.0
	TERRAIN FOLLOWING (TF)	V_{range}	V_{MAT}	MSL	10,000 ft.	0	3.5
	ANTISUBMARINE SEARCH (AS)	1.2 V_S	V_{MRT}	MSL	Medium	0	2.0
	CLOSE FORMATION FLYING (FF)	1.4 V_S	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L
	CLIMB (CL)	.85 $V_{R/C}$	1.3 $V_{R/C}$	MSL	Cruising Ceiling	.5	2.0
	CRUISE (CR)	V_{range}	V_{NRT}	MSL	Cruising Ceiling	.5	2.0
	LOITER (LO)	.85 V_{end}	1.3 V_{end}	MSL	Cruising Ceiling	.5	2.0
B	IN-FLIGHT REFUEL (TANKER) (RT)	1.4 V_S	V_{MAT}	MSL	Cruising Ceiling	.5	2.0
	DESCENT (D)	1.4 V_S	V_{MAT}	MSL	Cruising Ceiling	.5	2.0
	EMERGENCY DESCENT (ED)	1.4 V_S	V_{max}	MSL	Cruising Ceiling	.5	2.0
	EMERGENCY DECELERATION (DE)	1.4 V_S	V_{max}	MSL	Cruising Ceiling	.5	2.0
	AERIAL DELIVERY (AD)	1.2 V_S	200 kt	MSL	10,000 ft	0	2.0
	TAKEOFF (TO)	Minimum Normal Takeoff Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	CATAPULT TAKEOFF (CT)	Minimum Catapult End Airspeed	$V_{o_{min}}$ +30 kt	MSL	—	.5	n_L
C	APPROACH (PA)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	WAVE-OFF/GO-AROUND (WO)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft.	.5	2.0
	LANDING (L)	Minimum Normal Landing Speed	V_{max}	MSL	10,000 ft.	.5	2.0

*Appropriate to the operational mission.

SPEED ALTITUDE PROFILE

CLEAN

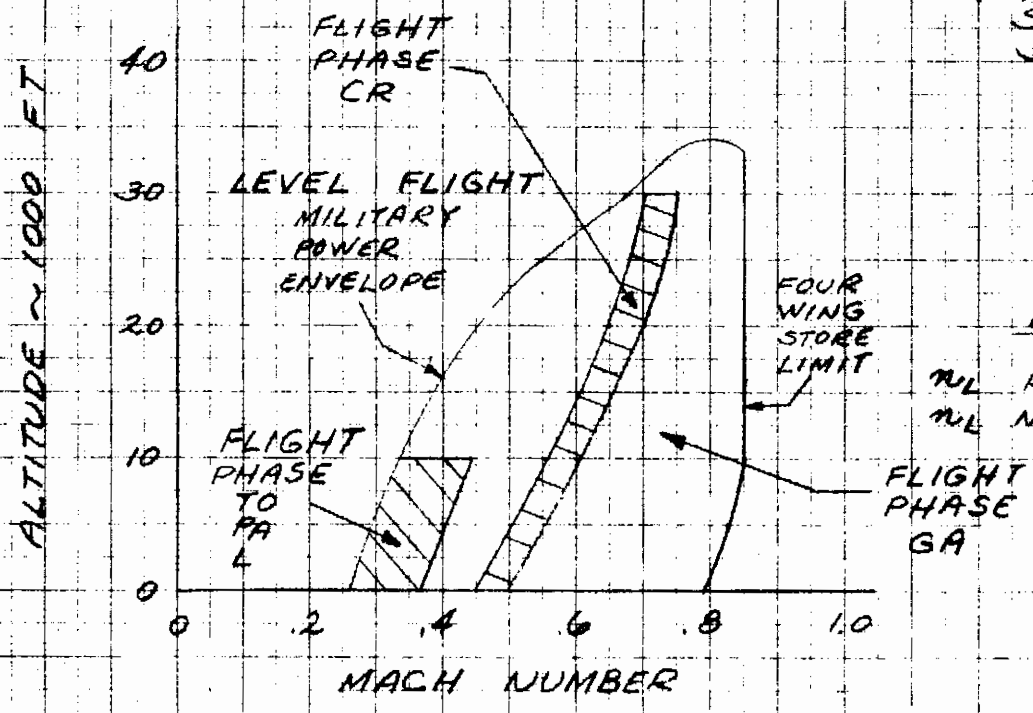


F-5 OPERATIONAL ENVELOPE

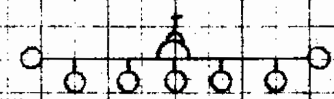
FIGURE 1 (3.1.7)

SPEED ALTITUDE PROFILE

STORES



(2) J85-GE-13 ENGINES
STANDARD ATMOSPHERE
STORES CONFIGURATION



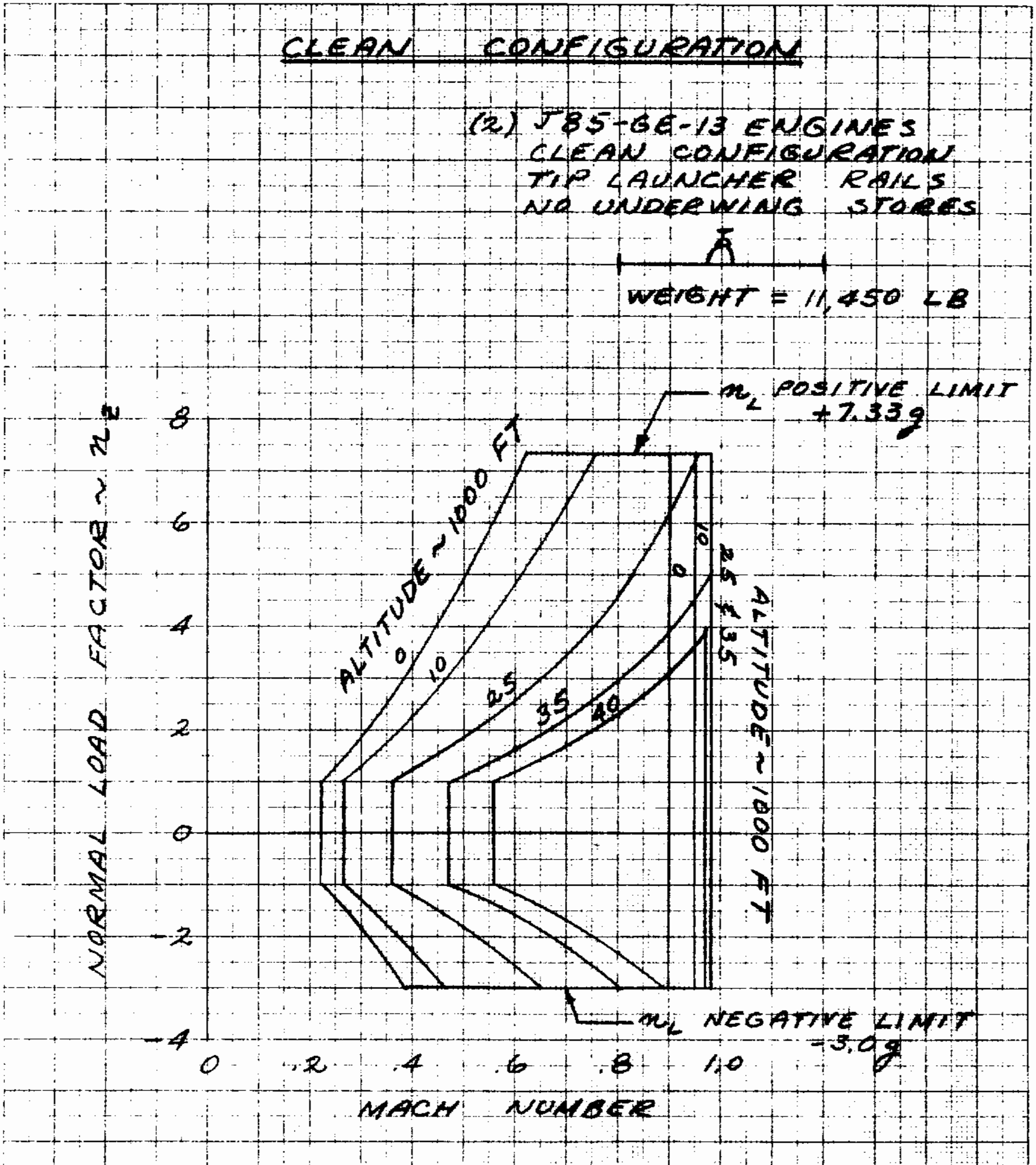
WEIGHT = 17,500 LB

LOAD FACTOR BOUNDARIES

n_L POSITIVE = +5.0 g
n_L NEGATIVE = −1.5 g

F-5 OPERATIONAL ENVELOPE

FIGURE 2 (3.1.7)



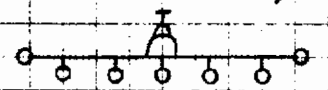
OPERATIONAL LOAD FACTOR ENVELOPE

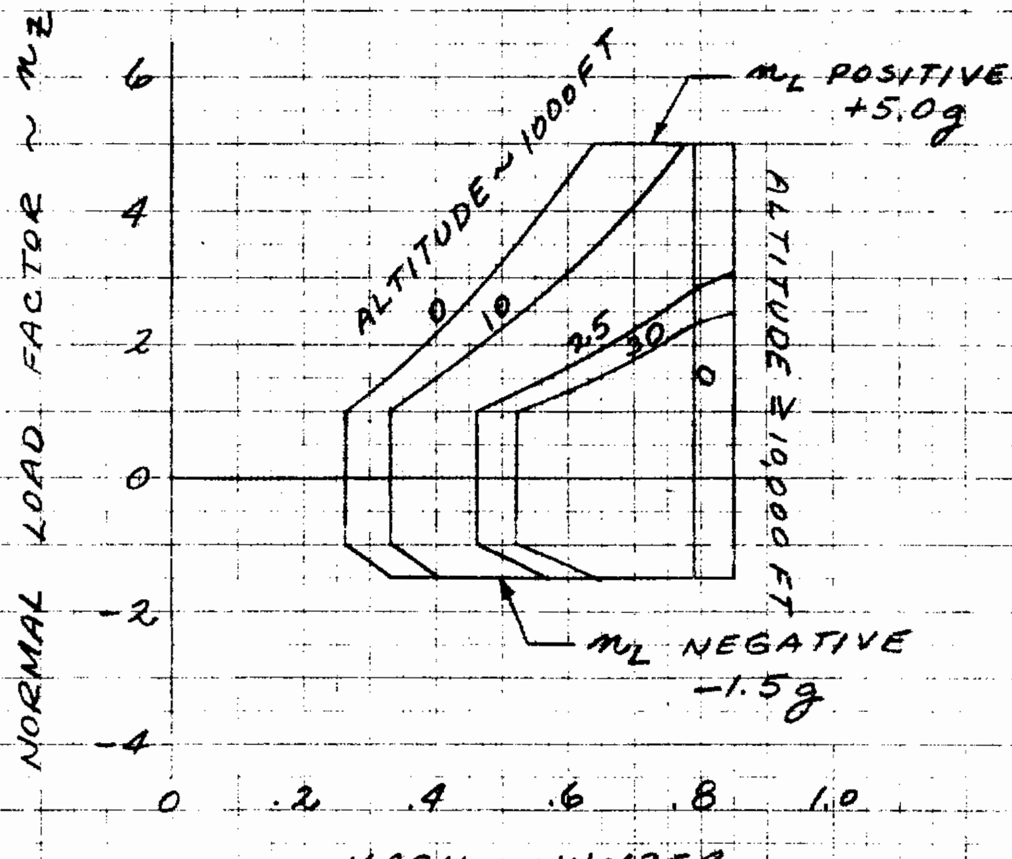
FIGURE 3 (3.1.7)

STORES CONFIGURATION

(2) J85-GE-13 ENGINES

STORES CONFIGURATION
WEIGHT = 17,500 LB





OPERATIONAL LOAD FACTOR ENVELOPE

FIGURE 4 (3.1.7)

Requirement

Paragraph 3.1.8 Service Flight Envelopes. For each Airplane Normal State the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude, and normal acceleration derived from airplane limits as distinguished from mission requirements. For each applicable Flight Phase and Airplane Normal State, the boundaries of the Service Flight Envelopes can be coincident with or lie outside the corresponding Operational Flight Envelopes, but in no case shall they fall inside those Operational boundaries. The boundaries of the Service Flight Envelopes shall be based on considerations discussed in 3.1.8.1, 3.1.8.2, 3.1.8.3, and 3.1.8.4.

Paragraph 3.1.8.1 Maximum service speed. The maximum service speed, V_{max} or M_{max}, for each altitude is the lowest of:

a. The maximum permissible speed

b. A speed which is a safe margin below the speed at which intolerable buffet or structural vibration is encountered.

c. The maximum airspeed at MAT, for each altitude, for dives (at all angles) from V_{MAT} at all altitudes, from which recovery can be made at 2,000 feet above MSL or higher without penetrating a safe margin from loss of control, other dangerous behavior, or intolerable buffet, and without exceeding structural limits.

Paragraph 3.1.8.2 Minimum service speed. The minimum service speed, V_{min} or M_{min}, for each altitude is the highest of:

a. 1.1 V_S

b. V_S + 10 knots equivalent airspeed

c. The speed below which full airplane-nose-up elevator control power and trim are insufficient to maintain straight, steady flight

d. The lowest speed at which level flight can be maintained with MRT

and for Category C Flight Phases:

e. A speed limited by reduced visibility or an extreme pitch attitude that would result in the tail or aft fuselage contacting the ground.

Paragraph 3.1.8.3 Maximum service altitude. The maximum service altitude, h_{max}, for a given speed is the maximum altitude at which a rate of climb of 100 feet per minute can be maintained in unaccelerated flight with MAT.

Comparison

The Service Flight Envelopes are presented in Figures 1 (3.1.8) and 2 (3.1.8). The external loadings presented cover the extremes as discussed in the comparison part of Paragraph 3.1.7.

The maximum service speed is shown in Figure 1 (3.1.8) for the F-5 air-to-air combat configuration. M_{max} is 1.4 at an altitude of 35,000 feet.

The minimum service speed is shown in Figure 1 (3.1.8) for the F-5 air-to-air combat configuration. M_{min} is 0.20 at sea level. This represents 1.1 V_S with landing flaps down.

The maximum service altitude, h_{max}, for the F-5 is shown in Figure 1 (3.1.8). It is 51,000 feet for Mach number of 0.95.

Resolution

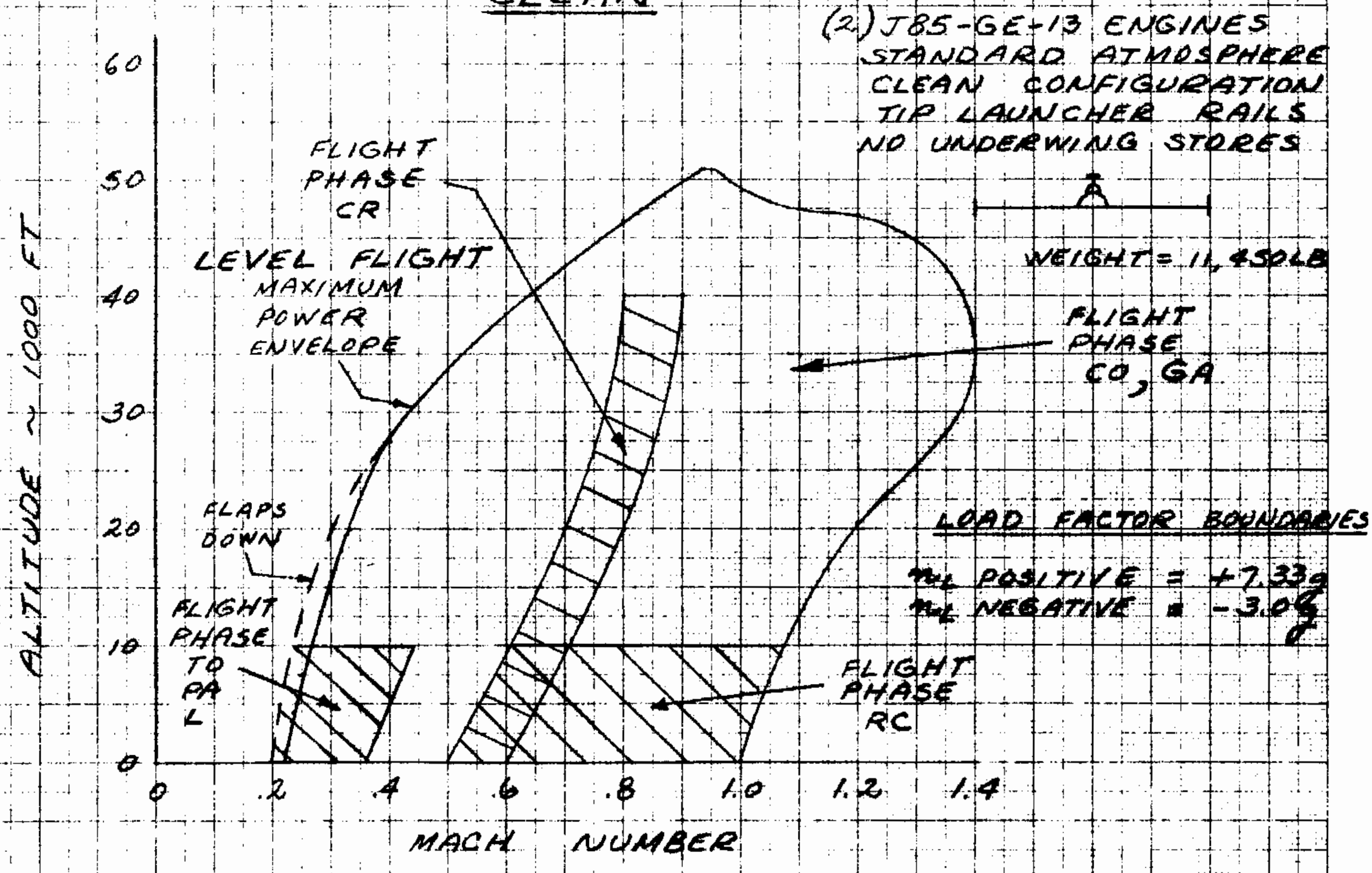
None

Recommendation

None

SPEED ALTITUDE PROFILE

CLEAN

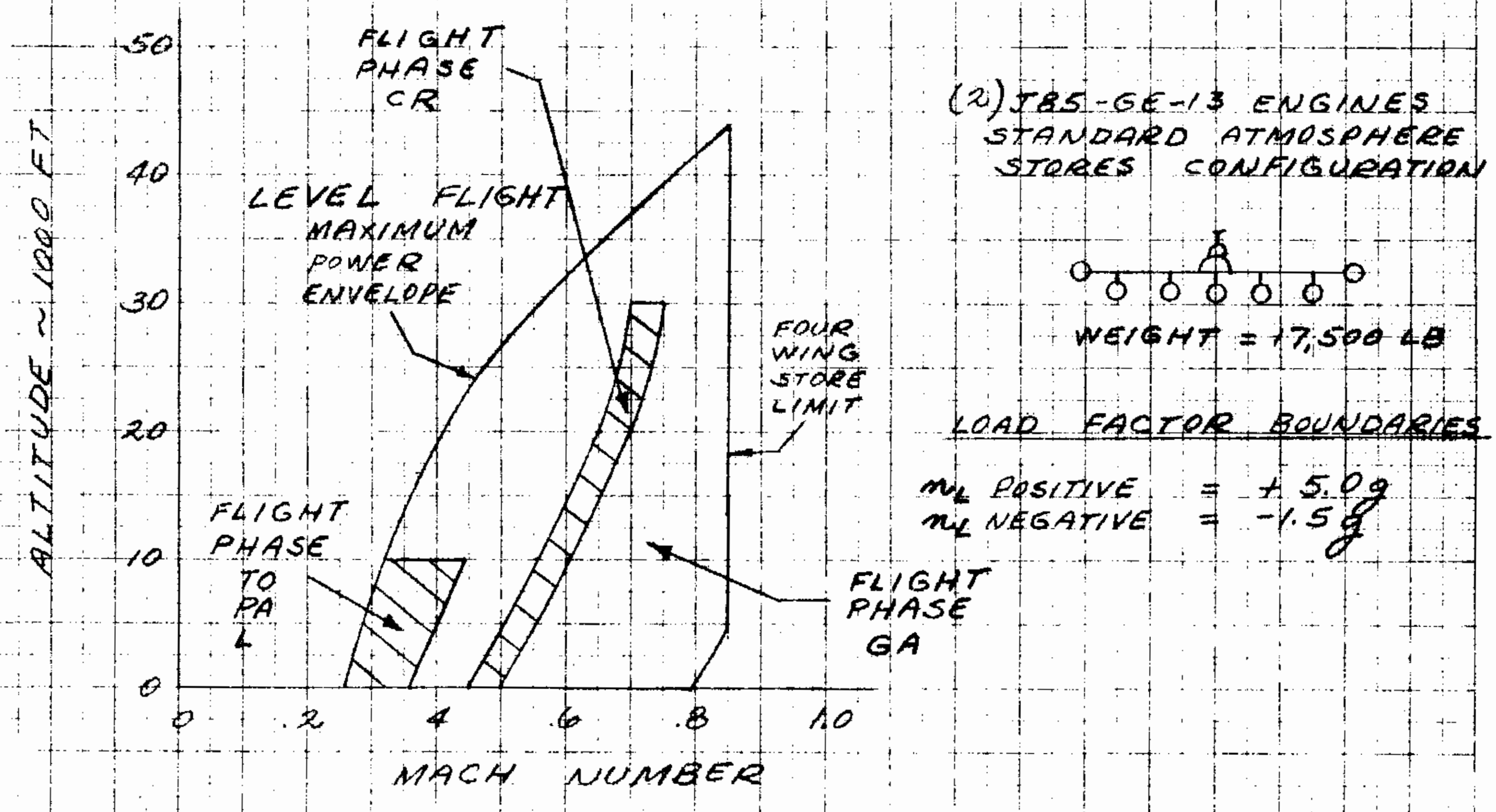


F-5 SERVICE ENVELOPE

FIGURE 1 (3.1.8)

SPEED ALTITUDE PROFILE

STORES



F-5 SERVICE ENVELOPE

FIGURE 2 (3.1.8)

Requirement

Paragraph 3.1.8.4 Service load factors. Maximum and minimum service load factors, n (+) [n (-)], shall be established as a function of speed for several significant altitudes. The maximum [minimum] service load factor, when trimmed for 1g flight at a particular speed and altitude, is the lowest [highest] algebraically of:

a. The positive [negative] structural limit load factor

b. The steady load factor corresponding to the minimum allowable stall warning angle of attack (3.4.2.2.2)

c. The steady load factor at which the elevator control is in the full airplane-nose-up [nose-down] position

d. A safe margin below [above] the load factor at which intolerable buffet or structural vibration is encountered.

Comparison

The Service load factors as a function of speed are presented for a clean underwing with no external stores in Figure 1 (3.1.8.4), and for a full complement of external stores in Figure 2 (3.1.8.4). These two configurations represent the extreme conditions relative to external stores. Every configuration and every weight will have its own load factor variance with speed. Consequently, the number of load factor versus speed envelopes are innumerable. Hence, the two extremes are presented and all others will fall within these envelopes.

Resolution

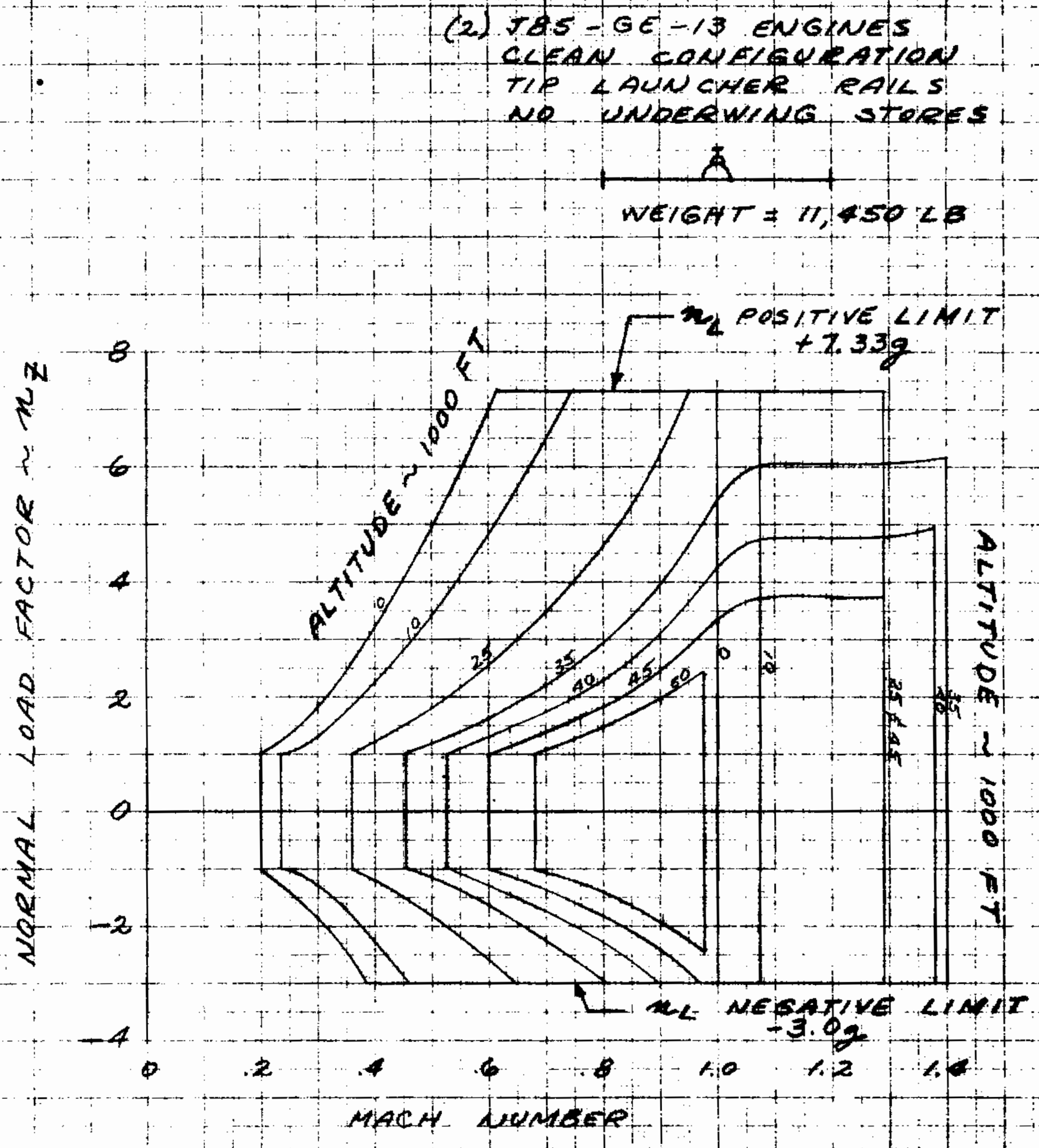
This paragraph does not specify the configurations for which service load factors as a function of speed must be constructed. In order to (1) satisfy the intended requirement of the specification, (2) provide sufficient information to the procuring activity and, (3) prohibit extended and perhaps unnecessary added effort on the part of the contractor which is time-consuming and costly, it is necessary to either specify the configurations directly in the paragraph or make allowance to establish these configurations through guidance and approval of the procuring activity.

Recommendation

The following statement is recommended to be added at the end of the paragraph:

"The specific configurations in terms of external store loadings or the absence of the external store loadings and the appropriate weight conditions for which service load factor versus speed envelopes are to be constructed shall be established with the guidance and approval of the procuring activity."

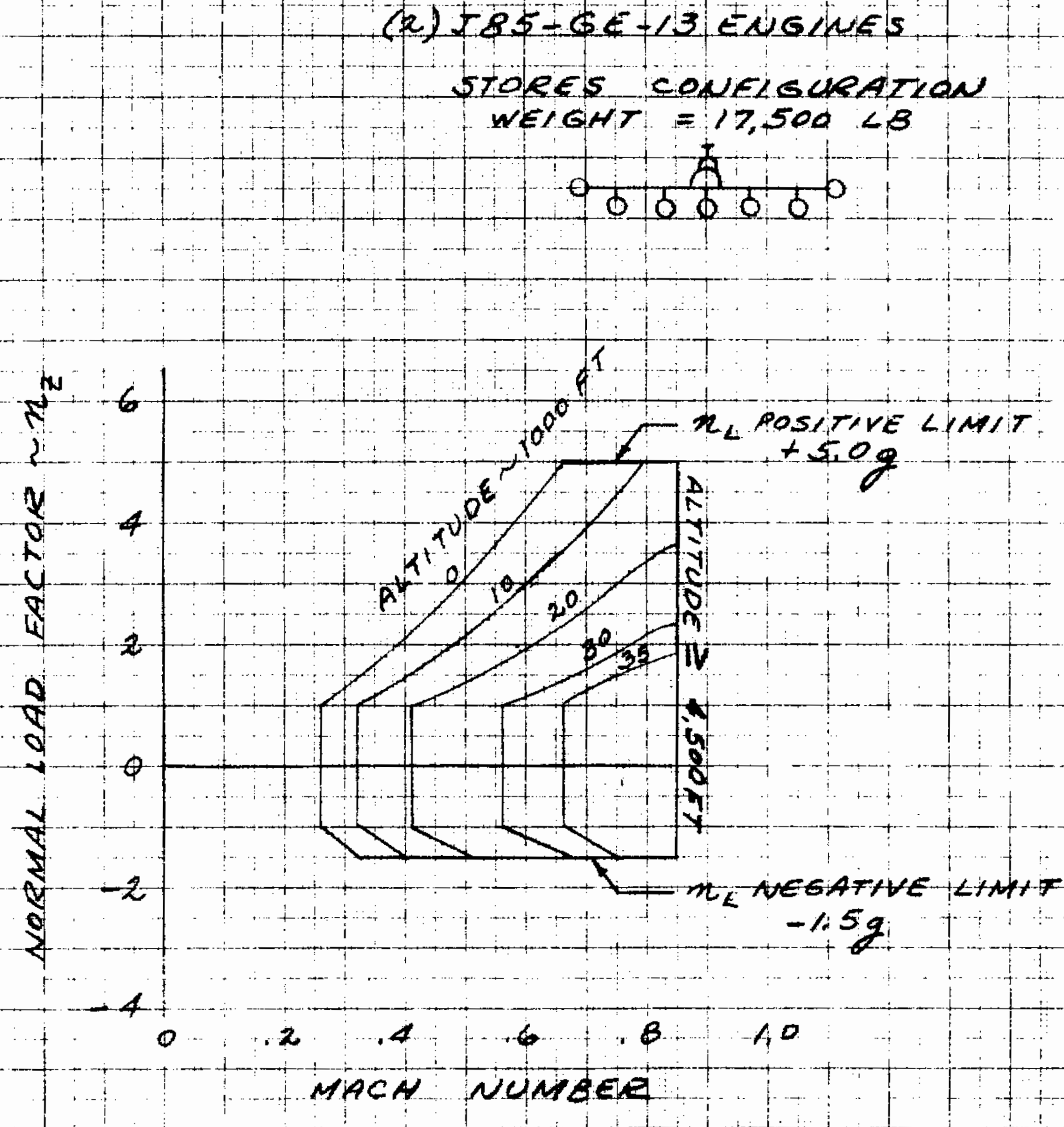
CLEAN CONFIGURATION



SERVICE LOAD FACTOR ENVELOPE

FIGURE 1 (3.1.8.4)

STORES CONFIGURATION



SERVICE LOAD FACTOR ENVELOPE

FIGURE 2 (3.1.8.4)

Requirement

Paragraph 3.1.9 Permissible Flight Envelopes. The Permissible Flight Envelopes encompass all regions in which operation of the airplane is both allowable and possible. These are the boundaries of flight conditions outside the Service Flight Envelope which the airplane is capable of safely encountering. Stalls, spins, zooms, and some dives may be representative of such conditions. The Permissible Flight Envelopes define the boundaries of these areas in terms of speed, altitude, and load factor.

Comparison

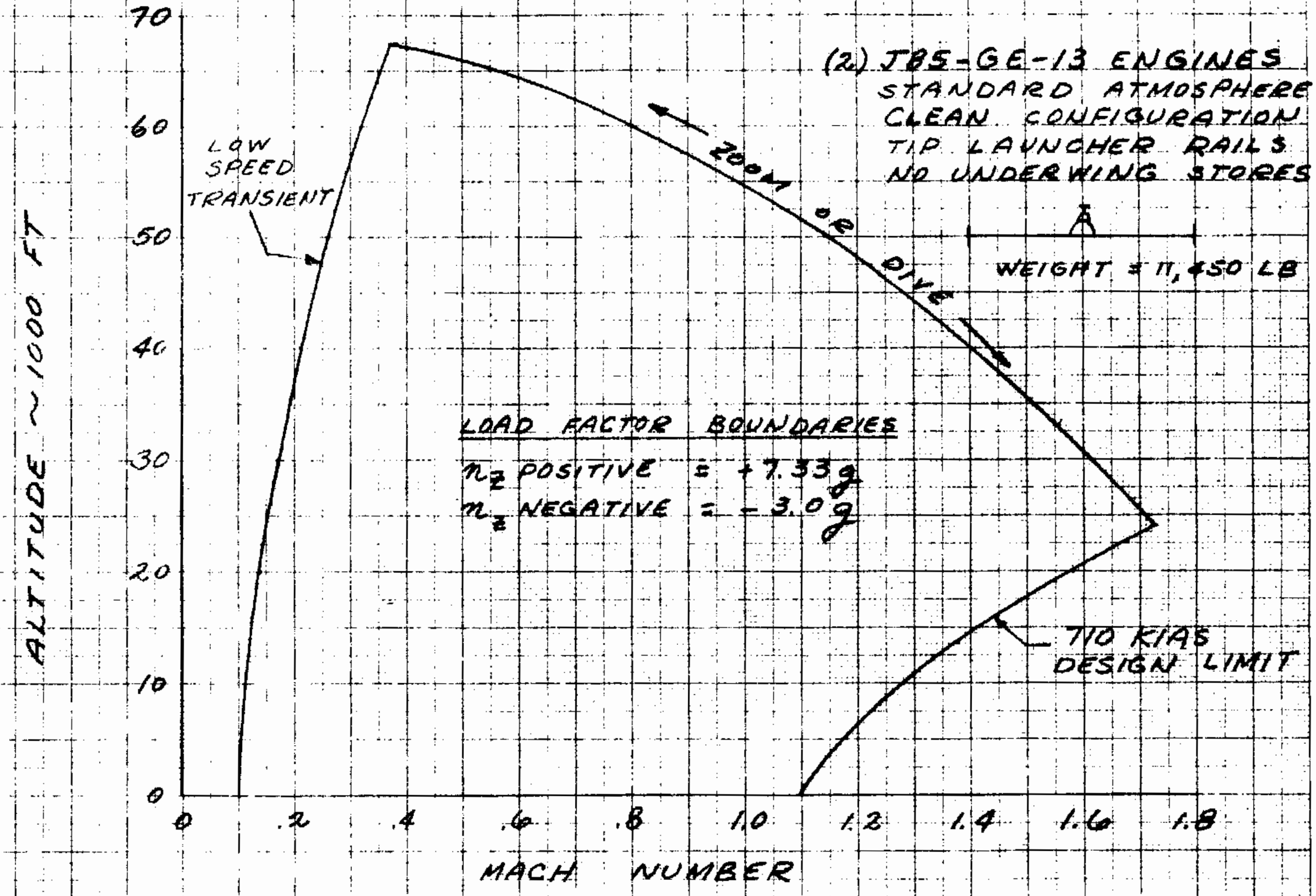
The Permissible Flight Envelopes are presented in Figures 1 (3.1.9) and 2 (3.1.9), respectively, for the clean configuration with no underwing stores and for the full complement of external stores loading. The zoom, the dive, the design limit speed and the low speed transient comprise the Permissible Flight Envelope. The low-speed transient region was obtained from flight test data of simulated air-to-air combat exercises involving an F-5A and an F-5B aircraft. The region is set by what was required for the combat exercises. The F-5 has no limits other than those imposed by thrust, drag and structural considerations.

Resolution

None

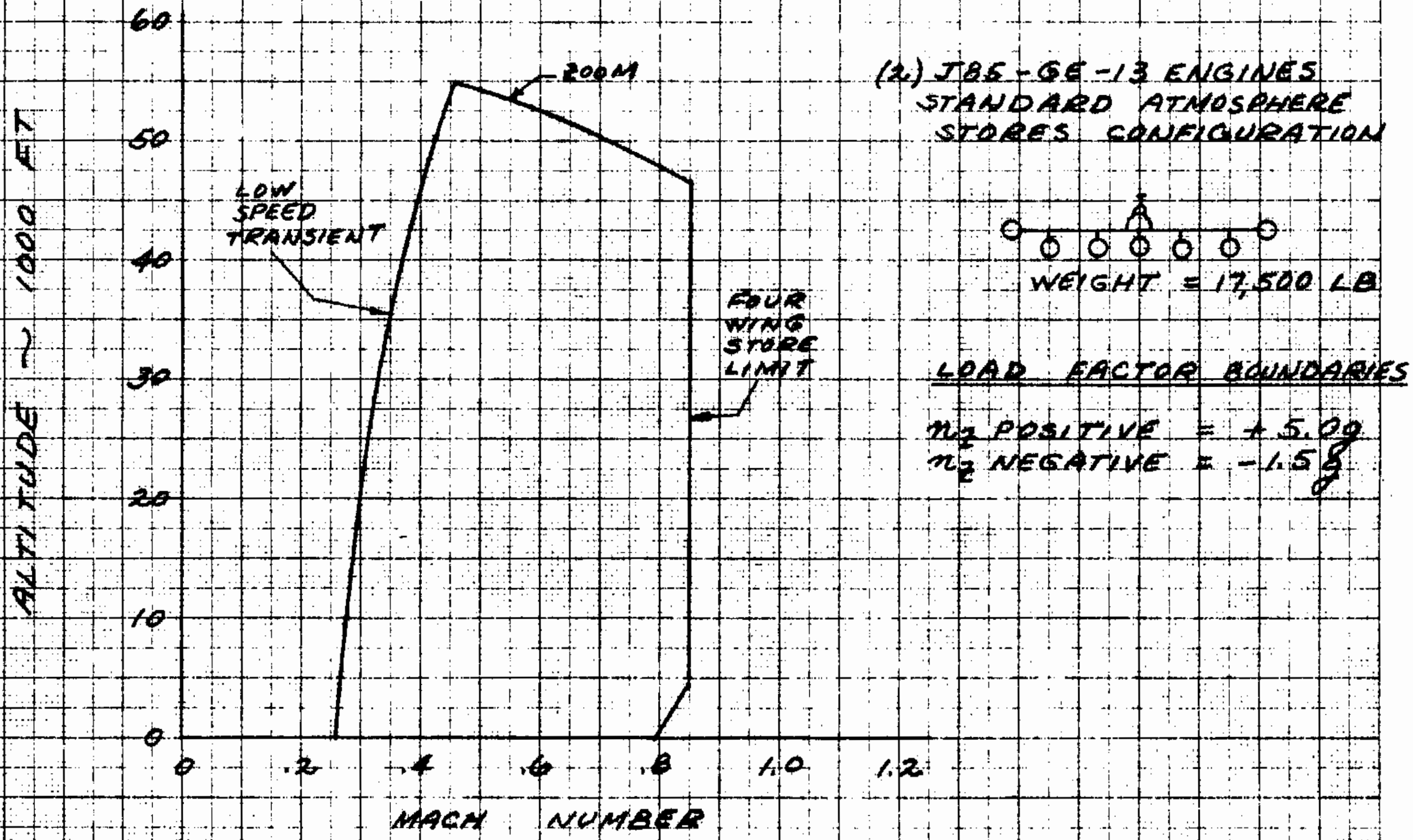
Recommendation

None



F-5 PERMISSIBLE ENVELOPE

FIGURE 1 (3.1.9)



F-5 PERMISSIBLE ENVELOPE

FIGURE 2 (3.1.9)

Requirement

Paragraph 3.1.9.1 Maximum permissible speed. The maximum permissible speed for each altitude shall be the lowest of:

a. Limit speed based on structural considerations

b. Limit speed based on engine considerations

c. The speed at which intolerable buffet or structural vibration is encountered

d. Maximum dive speed at MAT for each altitude, for dives (at all angles) from V_{MAT} at all altitudes, from which dive recovery at 2000 feet above MSL or higher is possible without encountering loss of control or other dangerous behavior, intolerable buffet or structural vibration, and without exceeding structural limits.

Comparison

The maximum permissible speed is shown as a function of altitude in Figure 1 (3.1.9). M_{max} is 1.72 at an altitude of 24,000 feet.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.9.2 Minimum permissible speed. The minimum permissible speed in 1g flight is V_S as defined in 6.2.2 or 3.1.9.2.1.

Comparison

V_S, stall speed at 1g flight, is a function of (1) the weight of the airplane, (2) the center of gravity of the airplane, (3) the external store loading, and (4) the flap settings. Therefore, a unique minimum permissible speed in 1g flight does not really exist; however, a variation of V_S with the above parameters exists. Such a variation is usually made available in Reference 1. The F-5 airplane, flaps up, clean configuration, weight of 10,000 pounds and center of gravity of 15% MAC, has a V_S equal to 134 KIAS.

Resolution

The intent of the specification is to define to the pilot the minimum permissible speed in level flight. Some minimum airspeeds may need to be defined such as for zooms to assure recoverability. Therefore, reference to definition of V_S is not sufficient to satisfy the intent of this requirement. A table of V_S or a figure of V_S as a function of parameters that affect the stall speed needs to be specified. Naturally, this could lead to a large magnitude of additional work, that may not be altogether necessary. Therefore, the procuring activity must specify all the conditions for which V_S should be indicated. The paragraph as presently written does not give a precise specification. The procuring activity will have to consider and approve the boundaries of the permissible flight envelopes and the design configurations as well as those required to demonstrate compliance with the specification. Consequently, tables or figures of V_S will have to be established based on the above and specified as required information.

Recommendation

It is recommended that the paragraph be written as follows:

"Minimum permissible speeds in 1g flight, V_S, as defined in 6.2.2 or 3.1.9.2.1 shall be prepared for the flight conditions of 3.1.9. The airplane loadings and configurations for which V_S are to be prepared shall be established with the guidance and approval of the procuring activity."

Requirement

Paragraph 3.1.9.2.1 Minimum permissible speed other than stall speed. For some airplanes, considerations other than maximum lift determine the minimum permissible speed in 1g flight (e.g., ability to perform altitude corrections, excessive sinking speed, ability to execute a wave-off (go-around), etc.). In such cases, an arbitrary angle-of-attack limit, or similar minimum speed and maximum load factor limits, shall be established for the Permissible Flight Envelope, subject to the approval of the procuring activity. This defined minimum permissible speed shall be used as V_S in all applicable requirements.

Comparison

None

Resolution

This paragraph makes allowances, subject to the approval of the procuring activity, to permit a minimum permissible speed higher than stall speed. In contrast to this, the F-5 can safely achieve speeds below stall speed when engaged in air-to-air combat, especially during zooms to zero airspeed.

Class IV airplanes, in Category A and Flight Phase (CO), air-to-air combat when engaging enemy aircraft will often, in some portion of the engagement, have to reduce airspeed below stall speed to:

(1) produce extremely high gravity aided angular velocities which are achievable at flight speeds below stall

(2) gain a position advantage in zooms against another aircraft which is limited to stall speed.

Therefore, allowances ought to be made to permit flights below stall speed as long as such flights are safe and recoverable. A pilot in air-to-air combat should be allowed to visually observe the enemy airplane at all times during the engagement, and not be required to monitor his airspeed indicator for minimum permissible airspeed, if he is to win. For this reason, provisions should be made not only to allow airspeeds below stall speed but also to ensure that they are safe and do not result in dangerous flight conditions.

Recommendation

It is recommended that airspeed below stall be allowed for special conditions. The minimum permissible speed in 1g flight is V_S for Classes I, II, and III aircraft as defined in 6.2.2 or 3.1.9.2.1. For Class IV aircraft with an air-to-air combat requirement, the minimum permissible speed should be low enough to permit full exploitation of its agility potential against an adversary. This may require the aircraft to be controllable to speeds as low as .5 V_S.

Further study of a below-stall-speed requirement is recommended and should include:

1. Survey of known and projected threat aircraft in terms of minimum permissible speed.

2. Air-to-air combat simulation of very low speed maneuvers, including below stall speed, to determine the effectiveness of this region.

3. Formulation of controllability requirements for operating below stall speed.

4. Determination of impact of below stall speed requirement on aircraft design.

Requirement

Paragraph 3.1.10 Applications of Levels. Levels of flying qualities as indicated in 1.5 are employed in this specification in realization of the possibility that the airplane may be required to operate under abnormal conditions. Such abnormalities that may occur as a result of either flight outside the Operational Flight Envelope, the failure of airplane components, or both, are permitted to comply with a degraded Level of flying qualities as specified in 3.1.10.1 through 3.1.10.3.3.

Paragraph 3.1.10.1 Requirements for Airplane Normal States. The minimum required flying qualities for Airplane Normal States (3.1.6.1) are as shown in table II.

TABLE II. Levels for Airplane Normal States

Within Operational Flight Envelope	Within Service Flight Envelope
Level 1	Level 2

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.1.10.2 Requirements for Airplane Failure States. When airplane Failure States exist (3.1.6.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 3.1.10.1 is sufficiently small. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Airplane Failure State per flight and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These determinations shall be based on MIL-STD-756 except that (a) all airplane components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the airplane, and (b) each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense). From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more flying qualities are degraded to Level 2 because of one or more failures. The contractor shall also determine the probability that one or more flying qualities are degraded to Level 3. These probabilities shall be less than the values shown in table III.

TABLE III. Levels for Airplane Failure States

Probability of Encountering	Within Operational Flight Envelope	Within Service Flight Envelope
Level 2 after failure	$<10^{-2}$ per flight	
Level 3 after failure	$<10^{-4}$ per flight	$<10^{-2}$ per flight

In no case shall a Failure State (except an approved Special Failure State) degrade any flying quality outside the Level 3 limit.

Comparison

The USAF F-5A, F-5B, and T-38A are Class IV airplanes as defined by Paragraph 1.3. The operating command maintenance, failure, accident, and incident reporting for these airplanes are employed for this validation. USAF records are not available in the detail needed for an exact comparison with the requirements on the probability of degraded flying qualities due to failures. Nevertheless, this approach has the merit of being based on actual operational experience.

The flight control subsystems are identical except for the stability augmentation and wing flap subsystems. The F-5A and F-5B airplanes have both pitch and yaw augmentation and flaps on both leading and trailing edges of the wings. The T-38A airplane contains the same yaw augmentation and trailing-edge flaps employed in the F-5A and F-5B airplanes but the T-38 is not equipped with pitch-axis augmentation or leading-edge flaps. All other features are sufficiently near to identical so that for the purpose of this validation the field use failure reporting for the T-38 airplane plus the F-5 pitch-axis augmentation and leading-edge flaps reporting will be employed to derive the probability of encountering Levels 2 and 3 after failure for comparison with Table III. There are two reasons for this decision:

1. More T-38 airplanes have been delivered and are being flown more hours per month per airplane than for the F-5 airplane.

2. The vast majority of F-5 airplanes are operated by foreign countries, so the available failure, incident, and accident investigation reports are not as complete or thorough as for USAF-operated airplanes.

AFM 66-1 Maintenance Data Collection reporting by USAF Air Training Command for the twelve-month period October 1969 through September 1970 is employed in this validation. This reporting encompasses 4,264 F-5 and 562,598 T-38 airplane flying hours. There were no accident or incident reports affecting the flight control system that resulted in handling qualities worse than Level 3.

Aircraft accidents and incidents reported in accordance with AFR 127-4 by the Air Training Command for the same period and number of flying hours stated in the above paragraphs are also employed.

All copies of F-5 and T-38 narrative Unsatisfactory Reports in Northrop's files and all narrative reports received from Northrop Field Service Representatives were reviewed. This was accomplished to provide current knowledge to permit factoring the purely statistical and non-narrative data presented by the AFM 66-1 reporting. It also provided supplemental information relating to the accidents and incidents reported via the AFR 127-4 system. The purpose of this review was to develop an understanding of the magnitude of the number of failures, by subsystem, that did not result in a flight control deficient condition. This review also provided an insight into failures that resulted from improper maintenance and incorrect operation of the airplanes. Over 2,000 narrative reporting documents were reviewed.

The following rules are employed in determining the number of failures counted.

1. Failures due to improper maintenance are not counted.

2. Failures due to improper operation are not counted.

3. Bird strikes and other airborne collisions with objects not from the damaged aircraft are not counted.

4. Failures of the following F-5 and T-38 subsystems only are counted:

   4.1 Engine gearbox to airframe-mounted accessory gearbox driveshaft

   4.2 Airframe-mounted accessory drive gearbox (drives hydraulic pump)

   4.3 Hydraulic power supply

   4.4 Stability augmentation

   4.5 Aileron

   4.6. Horizontal tail

   4.7 Rudder

   4.8 Leading-edge flaps

   4.9 Trailing-edge flaps

   4.10 Speed brake

   4.11 Airframe, wing, and landing gear door failures that result in damage to flight control or primary aerodynamic surfaces.

Failures of the electrical power supply are not counted because this subsystem in both F-5 and T-38 airplanes is redundant. The remaining active power supply subsystem, of two per airplane, is capable of supplying the total airplane electrical load. Automatic switching disconnects the failed system and connects the remaining power supply subsystem to supply both loads. Northrop has never received a report of a dual failure of the electrical power supply subsystem in more than 4,000,000 F-5 and T-38 flight hours. The probability of failure of the automatic switching functions is less than 5.78×10^{-7} per flight.

Failure rates are computed, based on the above rules, for failures reported via AFM 66-1 and AFR 127-4 reporting. These failure rates, identified by the first three digits of the work unit codes listed in USAF Technical Orders 1F-5A-06 and 1T-38A-06, are tabulated by subsystem in Table 1(3.1.10.2).

The computations to determine the probability of encountering Levels 2 and 3 after failure and remaining within the Operational Flight Envelope are based on assessing the failures resulting in flight abort as contributing to Level 3 and the non-abort failures discovered by the aircrew in flight as contributing to Level 2. The criteria of Paragraph 1.5 are adhered to in performing this assessment.

The process of assessing flight-abort-causing and aircrew-flight-discovered but non-abort-causing failures for use in calculating Levels 2 and 3 requires judgment, experience, and insight. These characteristics have been employed to develop the factors used to determine the incremental failure rates. The incremental failure rates are then summed for use in the probability calculations.

Each subsystem (incremental) failure rate (failures per airplane flight hour) is computed as follows:

$$\lambda = \frac{N}{T}$$

where λ is the failure rate, N is the number of failures, and T is the subsystem flight hours during the period that N failures occurred. A subsystem flight hour, in this report, is equal to an airplane flight hour.

The abort and non-abort factor formulas employ the expression "127-4 λ" to represent the failure rate for failures reported via Accident and Incident reporting system AFR 127-4 and "66-1 λ" for failures reported via Maintenance Data Collection reporting system AFM 66-1.

The failure rates that have been computed for each subsystem, based on AFR 127-4 and AFM 66-1 reporting systems, are presented in Table 1 (3.1.10.2). The factored subsystems failure rates are tabulated in Table 2 (3.1.10.2).

The following rationale is the basis for factoring each subsystem employed in the calculations.

Landing Gear Doors, Work Unit Code (WUC) 114.

Abort Failure Rate: The AFM 66-1 and AFR 127-4 reporting systems each indicate approximately the same failure rates due to landing gear doors. Flight aborts due to landing gear doors are the result of:

1. failure to obtain an indication of gear up and locked after repeated recycling the gear

2. landing gear doors breaking off in flight.

Under 1, the pilot aborted for precautionary reasons.

Under 2, the pilot aborted as the result of landing gear doors separation suspected by pilot as having caused damage resulting in flying qualities deficiencies.

The arithmetic mean will be used for the Level 3 calculation for this subsystem. This will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda) + (66\text{-}1\lambda)}{2}$$

Aircrew-Discovered Non-Abort Failure Rate: The failures that constitute the bulk of this reporting are primarily due to improperly adjusted landing gear doors and landing gear door sequencing or indicating switches. This results in the pilot recycling the landing gear up and down after the initial extension or retraction in order to obtain the proper indication. It is estimated that 90 percent of the T-38 AFM 66-1 reporting, for these non-abort aircrew reported failures, does not detrimentally affect the flying qualities of the airplane. Level 2 will be mathematically expressed as follows:

$$0.1x(66\text{-}1\lambda)$$

Wing, WUC 116

Abort Failure Rate: The air aborts due to the wing that affect the flying qualities of the T-38 airplane have primarily been due to the delamination of the wing tip resulting in roll and/or yaw axes control deficiencies. The wing tip on the T-38 airplane contains more honeycomb-core-bonded-to-skin area than does the F-5 airplane. The T-38 AFR 127-4 narrative reporting is quite accurate and inclusive in reporting air aborts due to wing problems. The AFR 127-4 reporting will be used for determining the aborts due to the wing. Level 3 will be mathematically expressed as follows:

$$1.0x(127\text{-}4\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: All AFM 66-1 reported failures in this category will be counted in determining the failure rate contributing to Level 2. This will be mathematically expressed as follows:

$$1.0x(66\text{-}1\lambda)$$

Gearbox and Driveshaft, WUC 117

Abort Failure Rate: During the twelve-month failure reporting period employed in this validation there was an excessive number of gearbox failures of overhauled units due to problems encountered with a new overhaul contractor. This condition should not reflect upon the airplane. Also, all failures reported are not always verified by the shop as being deficient or resulting in a failure mode that degrades the flying qualities of the airplane. The flight abort failure rate for this subsystem will be computed by subtracting the incident and accident rate reported in accordance with AFR 127-4 from the AFM 66-1 abort rate, then taking 10% of this value plus the AFR 127-4 rate. This will be mathematically expressed as follows:

$$\left[(66\text{-}1\lambda)-(127\text{-}4\lambda)\right]0.1+(127\text{-}4\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: A review of the Unsatisfactory (narrative) Reports indicates that 50% of the failure rate derived from AFM 66-1 reporting will be applicable for this subsystem. This will be mathematically expressed as follows:

$$0.5\text{x}(66\text{-}1\lambda)$$

Aileron, WUC 141

Abort Failure Rate: The flight abort failure rate for this subsystem is considered to be the arithmetic mean of the AFR 127-4 and AFM 66-1 abort reporting. Level 3 will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda)+(66\text{-}1\lambda)}{2}$$

Aircrew-Discovered Non-Abort Failure Rate: A large percentage of the aileron problems reported for this category are due to improper adjustment of the rigging. Level 2 will be assumed and mathematically expressed as follows:

$$0.1\text{x}(66\text{-}1\lambda)$$

Horizontal Tail, WUC 142

Abort Failure Rate: Of the three basic flight control axes there is no redundancy for the pitch axis. Non-performance of the roll or yaw control subsystems may be accommodated by the remaining one of either of these two subsystems. The aircrew is extremely aware of this situation and is more critical of apparent changes in the feel and response of the pitch control subsystem. Approximately 90% of pitch-axis problems reported are due to misrigging and incorrect pilot calculation to determine rotational speed and takeoff ground distance. Level 3 will be mathematically expressed as follows:

$$\left[(66\text{-}1\lambda)-(127\text{-}4\lambda)\right]0.25+(127\text{-}4\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: Most of the horizontal tail (pitch axis) problems reported are due to improper rigging. This includes the flap to horizontal tail and speed brake to horizontal tail inter-connect functions. The arithmetic mean will be used for the calculation for this subsystem. This will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda) + (66\text{-}1\lambda)}{2}$$

Rudder, WUC 143

Abort Failure Rate: The principal causes of rudder subsystem failures are; improper rigging, hydraulic leaks, and access door screw fastenings improperly secured that project sufficiently to hinder rudder movement. There were no aborts reported by the AFR 127-4 accident and incident reporting system. One-half of the AFM 66-1 abort failure rate is representative of the failure rate for this subsystem. Level 3 will be mathematically expressed as follows:

$$0.5(66\text{-}1\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: Review of narrative reporting, via Unsatisfactory Reports, indicates that Level 2 calculation for this subsystem should be mathematically expressed as follows:

$$[(66\text{-}1\lambda)-(127\text{-}4\lambda)]\,0.25 + (127\text{-}4\lambda)$$

Leading-Edge Flaps, WUC 144

Abort Failure Rate: There were no F-5 airplane abort-causing failures reported via AFR 127-4 or AFM 66-1 reporting systems. Based on previous years' experience, 100% of the AFM 66-1 reporting would be indicated for Level 3 for this subsystem. This is mathematically expressed as follows:

$$1.0\text{x}(66\text{-}1\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: Review of the narrative reporting documents indicates that 100% of the F-5 AFM 66-1 failure rate should be used to determine Level 2 for this subsystem. This will be mathematically expressed as follows:

$$1.0\text{x}(66\text{-}1\lambda)$$

Trailing-Edge Flaps, WUC 145

Abort Failure Rate: There were no failures resulting in an abort reported via AFR 127-4. A review of narrative reporting documents indicates that 10% of the AFM 66-1 abort failure rate should be used. This will be mathematically expressed as follows:

$$0.1x(66\text{-}1\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: There were no failures reported via AFR 127-4 for this category. Review of narrative failure reporting documents indicates that 10% of the AFM 66-1 failure rate for this category should be used. This will be mathematically expressed as follows:

$$0.1x(66\text{-}1\lambda)$$

Speed Brake, WUC 146

Abort Failure Rate: A narrative document review indicates that the arithmetic mean of the AFR 127-4 and AFM 66-1 reporting systems should be used for this category. The Level 3 factor will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda) + (66\text{-}1\lambda)}{2}$$

Aircrew-Discovered Non-Abort Failure Rate:: The majority of the failures reported by narrative documents indicate they are caused by improper maintenance. Twenty percent of the AFM 66-1 reported failures will be used for this category. This will be mathematically expressed for the Level 2 factor as follows:

$$0.2x(66\text{-}1\lambda)$$

Hydraulic Power, WUC 451

Abort Failure Rate. Approximately 90% of the failures resulting in an abort due to this subsystem are due to maintenance errors of omission and commission. The Level 3 factor will be mathematically expressed as follows:

$$0.1x(66\text{-}1\lambda)$$

Aircrew-Discovered Non-Abort Failure Rate: Narrative reporting indicates that about 90% of the reported failures for this category do not result in an increase of pilot workload or degrade the mission effectiveness, or else are due to improper maintenance. The Level 2 factor will be mathematically expressed as follows:

$$0.1\text{x}(66\text{-}1\lambda)$$

Stability Augmenter, WUC 521

Abort Failure Rate: A review of narrative failure reports indicates that 50% of the mean of the AFR 127-4 and AFM 66-1 reporting should be counted for this category. The Level 3 factor will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda)+(66\text{-}1\lambda)}{2}\text{x}0.5$$

Aircrew-Discovered Non-Abort Failure Rate: Narrative failure reporting documents reviewed indicate that for Level 2 the factor for this category should be 50% of the mean of the AFR 127-4 and AFM 66-1 reporting. This will be mathematically expressed as follows:

$$\frac{(127\text{-}4\lambda)+(66\text{-}1\lambda)}{2}\text{x}0.5$$

There are eleven F-5 and T-38 subsystems for which failures are counted for use in determining the Levels of Flying Qualities. All eleven are considered to be in continual operation throughout the flight; hence, the Exponential distribution is appropriate for these calculations.

$$\text{Reliability} = R = e^{-\lambda T}$$

$$\text{Probability of Failure} = P_f = 1 - R$$

When "λT" is small, < 0.02, the probability of failure calculation may be simplified to the following:

$$P_f = \lambda T$$

The probability of encountering Level 2 and Level 3 after failure and remaining within the Operational Flight Envelope is computed as follows:

Probability of Level 2 flying qualities after failure = P_2

$$P_2 = \sum \lambda_{na} \times T$$

$$P_2 = 0.00182904 \times 2.0$$

$$P_2 = 0.00366 = 3.7 \times 10^{-3}$$

Probability of Level 3 flying qualities after failure = P_3

$$P_3 = \sum \lambda a \times T$$

$$P_3 = 0.00022543 \times 2.0$$

$$P_3 = 0.000451 = 4.5 \times 10^{-4}$$

Where: $\sum \lambda na = \lambda na_1 + \lambda na_2 \cdots\cdots + \lambda na_{11}$ = the sum of the eleven subsystems factored aircrew discovered non-abort failure rates = 0.00182904, from Table 2 (3.1.10.2)

$\sum \lambda a = \lambda a_1 + \lambda a_2 \cdots\cdots + \lambda a_{11}$ = the sum of the eleven subsystems factored abort failure rates = 0.00022543, from Table 2 (3.1.10.2)

T = mission duration in hours for the longest F-5 or T-38 airplane mission = 2.0 hours. .

Field experience of more than 4,000,000 F-5 and T-38 flying hours indicates an average mission time of 1.2 hours and maximum missions of 2.0 hours.

1. The F-5 and T-38 airplanes' indicated probability of encountering Level 2 after failure and remaining within the Operational Flight Envelope is 3.7×10^{-3} per flight. This meets the Level 2 probability of less than 10^{-2} per flight that is required by Table III.

2. The F-5 and T-38 airplanes' indicated probability of encountering Level 3 after failure and remaining within the Operational Flight Envelope is 4.5×10^{-4} per flight. This fails to meet the Level 3 probability of less than 10^{-4} per flight that is required by Table III.

3. The F-5 and T-38 aircrew-discovered non-abort factored rate is employed to determine the indicated probability of encountering Level 3 after failure and remaining within the Service Flight Envelope. Operational experience was statistically applied based on the following considerations:

A. The most comprehensive and meaningful fact provided by the AFM 66-1 failure reporting and AFR 127-4 Accident and Incident reporting is the pilot decision to abort the flight or continue when a failure symptom is discovered.

B. Conversations were held with service pilots who have flown F-5 and T-38 aircraft in operational service flying. The pilots stated that they decide to abort the flight if they detect a failure sympton that may prevent the aircraft from being capable of achieving the full performance of its Service Flight Envelope as defined by Paragraph 3.1.8.

The probability of encountering Level 3 after failure and remaining within the Service Flight Envelope is 3.7×10^{-3} per flight. This meets the Level 3 probability of less than 10^{-2} per flight that is required by Table III. This 3.7×10^{-3} is the same value previously computed for the probability of encountering Level 2 after failure for the Operational Flight Envelope.

Resolution

The analysis techniques and assumptions were based on judgment and experience in the absence of specified instructions in this new requirement. The ground rules used illustrate the approach that was considered the most logical to implement.

This validation of the requirements of Paragraph 3.1.10.2 indicates agreement with two of the three numerical values specified in Table III.

The F-5 and T-38 airplanes' probability of encountering Level 3 after failure and remaining within the Operational Flight Envelope difference is not large, 4.5×10^{-4} versus less than 10^{-4} required. This difference is significant because it raises the following questions:

1. Is one family (F-5 and T-38) of one classification of airplanes representative of all airplanes that meet the requirements of Class IV as defined by Paragraph 1.3?

2. Will similarly factored abort failure rates and aircrew discovered non-abort failure rates, for several different operational airplanes that are classified within each of the remaining three classifications, specified in Paragraph 1.3, provide an indicated probability of encountering Levels 2 and 3 after failure that will agree with Table III?

Recommendation

In order to determine if airplanes meeting the four classifications specified in Paragraph 1.3 require different values and to determine what numerical values should be specified in Table III it is recommended that the following tasks be accomplished. This will also provide the necessary data to formulate uniform ground rules and refine the analysis approach.

1. One or more of the latest model airplanes for each of the four classifications, that have a maturity of at least four years and a quantity of airplanes in service in excess of fifty, shall be selected as candidates for Task 2.

2. Assess the probability of encountering Levels 2 and 3 after failure by using the same procedures employed for assessing F-5 and T-38 airplanes for each Level and Flight Envelope.

3. Establish the numerical values for each Class or for combined Classes as indicated by the result of performing the assessment in Task 2.

SUBSYSTEM FAILURE RATES
FROM AFM 66-1 AND AFR 127-4 REPORTING

WORK UNIT CODE	SYSTEM	AFR 127-4 FAILURE RATES		AFM 66-1 FAILURE RATES	
		ABORT $\lambda \times 10^{-6}$	NON-ABORT $\lambda \times 10^{-6}$	ABORT $\lambda \times 10^{-6}$	NON-ABORT $\lambda \times 10^{-6}$
114	Landing Gear Door	1.77	1.77	1.80	35.50
116	Wing	8.88	0	10.70	17.80
117	Gearbox and Driveshaft	24.88	0	76.40	165.30
141	Aileron	3.55	5.33	101.30	879.80
142	Horizontal Tail	3.55	3.55	245.30	680.80
143	Rudder	0	1.77	32.00	323.50
144	L.E. Flap	0	0	0	469.00
145	T.E. Flap	0	0	115.50	1400.60
146	Speed Brake	1.77	0	19.60	200.90
451	Hydraulic Power	0	0	252.40	1439.70
521	Stability Augmenter	1.77	5.33	37.30	1672.60

REQUIREMENTS FOR AIRPLANE FAILURE STATES

TABLE 1 (3.1.10.2)

FACTORED SUBSYSTEM FAILURE RATES

WORK UNIT CODE	SYSTEM	ABORT FACTOR	NON-ABORT FACTOR	FACTORED ABORT λ x 10^{-6}	FACTORED NON-ABORT λ x 10^{-6}
114	Landing Gear Door	$\frac{(127-4\lambda)+(66-1\lambda)}{2}$	$0.1x(66-1\lambda)$	1.78	3.55
116	Wing	$1.0x(127-4\lambda)$	$1.0x(66-1\lambda)$	8.88	17.80
117	Gearbox and Driveshaft	$[(66-1\lambda)-(127-4\lambda)]0.1 + (127-4\ )$	$0.5x(66-1\lambda)$	30.03	82.65
141	Aileron	$\frac{(127-4\lambda)+(66-1\lambda)}{2}$	$0.1x(66-1\lambda)$	52.42	87.98
142	Horizontal Tail	$[(66-1\lambda)-(127-4\lambda)]0.25 + (127-4\lambda)$	$\frac{(127-4\lambda)+(66-1\lambda)}{2}$	63.99	342.17
143	Rudder	$0.5x(66-1\lambda)$	$[(66-1\lambda)-(127-4\lambda)]0.25 + (127-4\lambda)$	16.00	82.20
144	L.E. Flap	$1.0x(66-1\lambda)$	$1.0x(66-1\lambda)$	0	469.00
145	T.E. Flap	$0.1x(66-1\lambda)$	$0.1x(66-1\lambda)$	11.55	140.06
146	Speed Brake	$\frac{(127-4\lambda)+(66-1\lambda)}{2}$	$0.2x(66-1\lambda)$	5.78	40.18
451	Hydraulic Power	$0.1x(66-1\lambda)$	$0.1x(66-1\lambda)$	25.24	143.79
521	Stability Augmenter	$\frac{(127-4\lambda)+(66-1\lambda)}{2}x0.5$	$\frac{(127-4\lambda)+(66-1\lambda)}{2}x0.5$	9.76	419.48
			TOTALS	225.43	1829.04

REQUIREMENTS FOR AIRPLANE FAILURE STATES

TABLE 2 (3.1.10.2)

Requirement

Paragraph 3.1.10.2.1 Requirements for specific failures. The requirements on the effects of specific types of failures, e.g., propulsion or flight control system, shall be met on the basis that the specific type of failure has occurred, regardless of its probability of occurrence.

Comparison

None

Resolution

The definition of the specific failures is not sufficiently clear to make the requirements comprehensible.

It is to be noted in the validation response to Paragraph 3.1.10.2 that failures of the F-5 and T-38 propulsion systems are not counted as contributing to the flying qualities of these airplanes. The standard operating procedure for dual-engine F-5 and T-38 airplanes is to abort the flight, Reference 1, if a power loss is sustained by one engine.

During the design phase of each specific airplane, a detail study of the airplane's systems and components is necessary to determine (1) those components of the flight control system that will be affected, and (2) the extent of the effect, to determine which supporting systems, subsystems, and components affect the flying qualities.

Recommendation

At the end of the paragraph, add the following:

"Those specific failures that affect the flying qualities are to be defined by the contractor and approved by the procuring activity."

Requirement

Paragraph 3.1.10.3 Exceptions

Paragraph 3.1.10.3.1 Ground operation and terminal Flight Phases. Some requirements pertaining to takeoff, landing, and taxiing involve operation outside the Operational, Service and Permissible Flight Envelopes, as at V_S or on the ground. When requirements are stated at conditions such as these, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

Paragraph 3.1.10.3.2 When Levels are not specified. Within the Operational and Service Flight Envelopes, all requirements that are not identified with specific Levels shall be met under all conditions of component and system failure except approved Airplane Special Failure States (3.1.6.2.1).

Paragraph 3.1.10.3.3 Flight outside the Service Flight Envelope. From all points in the Permissible Flight Envelopes, it shall be possible readily and safely to return to the Service Flight Envelope without exceptional pilot skill or technique, regardless of component or system failures. The requirements on stall, spin, and dive characteristics, on dive recovery devices, and on approach to dangerous flight conditions shall also apply.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2 Longitudinal flying qualities.

Paragraph 3.2.1 Longitudinal stability with respect to speed

Paragraph 3.2.1.1 Longitudinal static stability. There shall be no tendency for the airspeed to diverge aperiodically when the airplane is disturbed from trim with the cockpit controls fixed and with them free. This requirement will be considered satisfied if the variations of elevator control force and elevator control position with airspeed are smooth and the local gradients stable, with:

Trimmer and throttle controls not moved from the trim settings by the crew, and

lg acceleration normal to the flight path, and

Constant altitude

over a range about the trim speed of ±15 percent or ±50 knots equivalent airspeed, whichever is less (except where limited by the boundaries of the Service Flight Envelope). Stable gradients mean increasing pull forces and aft motion of the elevator control to maintain slower airspeeds and the opposite to maintain faster airspeeds. The term gradient does not include that portion of the control force or control position versus airspeed curve within the preloaded breakout force or friction range.

Comparison

The accelerate/decelerate maneuver was performed to demonstrate in flight the longitudinal static stability. The airplane is trimmed at a flight condition (altitude and speed). To accelerate to a higher speed, the throttle is moved forward to produce sufficient excess power to accelerate the airplane to the desired higher speed. The pilot changes the longitudinal stick position to to reestablish constant altitude and lg acceleration normal to the flight path with the throttle not moved. To perform the decelerate portion of the maneuver, the pilot does the reverse, by moving the throttle aft, waiting for the transients to damp out, then changing the longitudinal stick position to maintain constant altitude and lg acceleration normal to the flight path with the throttle not moved. If the airplane possesses stable gradients, the pilot will push forward on the longitudinal control when accelerating and pull back when decelerating.

The accelerate/decelerate maneuver, as explained above, is considered valid to compare the characteristics of the F-5 with the intended requirements of this paragraph. This paragraph requires that the throttle not be moved from the trim settings. This is incompatible with the other requirements of the paragraph and will be discussed in the resolution part. The flight test data are presented in Figures 1 (3.2.1.1) through 5 (3.2.1.1). Clean, three-store and four-store configurations, at various c.g. locations and altitudes, are presented to cover the air-to-air combat and ground attack flight phases. The longitudinal stick force (F_S) and position (δ_S) show smooth and stable gradients except for the transonic region to be discussed in 3.2.1.1.1. Consequently it is concluded that the F-5 characteristics compare favorably with the interpreted requirements of this paragraph. Pilot comments indicated no adverse characteristics.

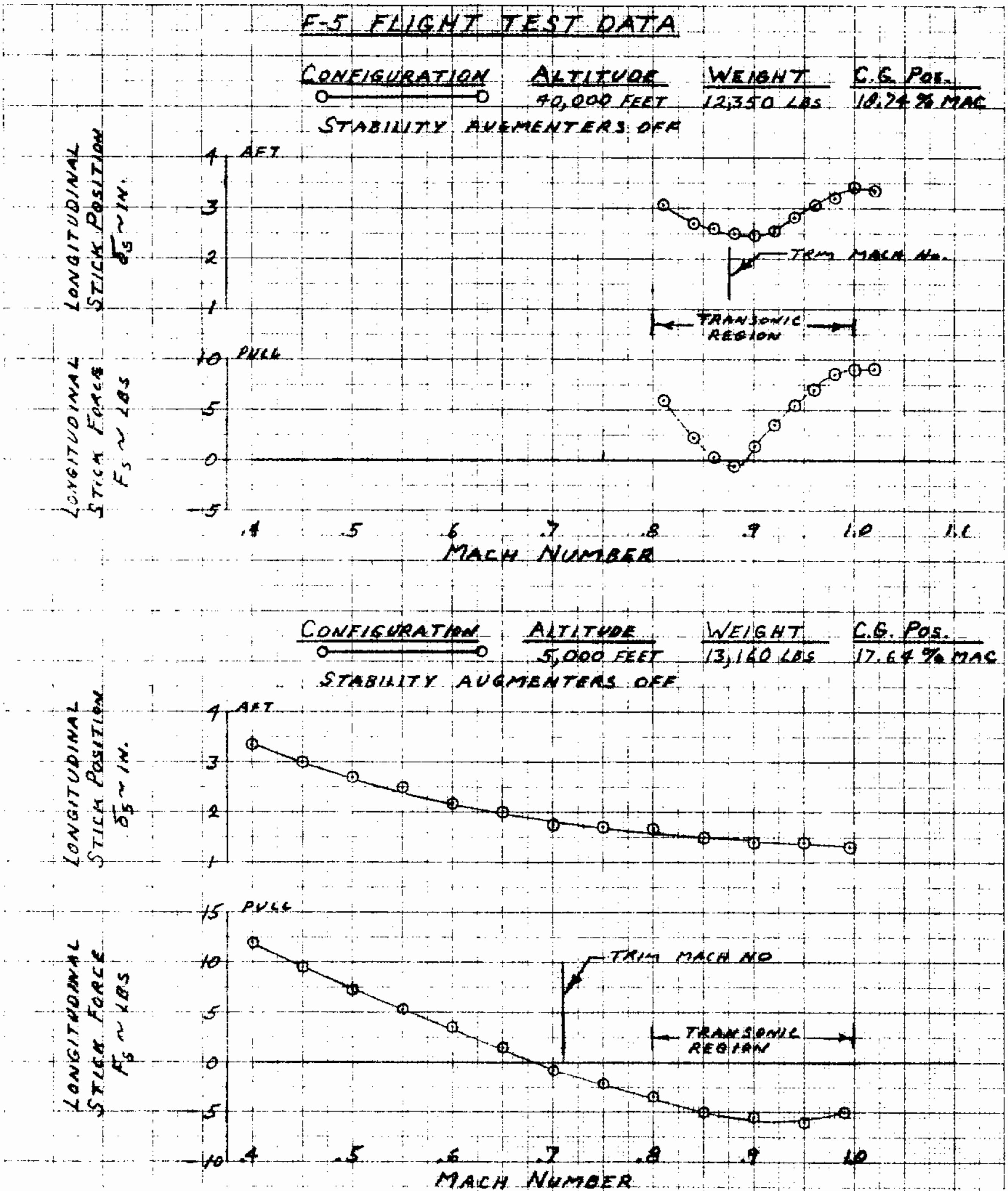
Resolution

The second paragraph, "Trimmer and throttle controls not moved from the trim settings by the crew, and" is considered incompatible with the third and fourth paragraphs, "lg acceleration normal to the flight path, and constant altitude." If the throttle setting is not changed to that which is required to obtain higher or lower speeds at constant altitude, but instead the speed is changed with the longitudinal control only, the airplane will be in either a climb or a dive at the stabilized condition. Not only will the altitude vary considerably, but the pilot will have no adequate means of maintaining constant climb or dive angle, and this will lead to introduction of errors due to load factor. The resulting data will exhibit incorrect longitudinal static stability. Compliance with the lg acceleration and constant altitude is also not achieved. To conclude, the throttle movement is necessary to change power to a different level to change speed at constant altitude and the longitudinal control movement is necessary to maintain lg flight, constant altitude and demonstrate the presence or absence of static stability. However, power effects on the static stability are present. Therefore, the level of power, which is a function of the throttle position, must be integrated into the requirement.

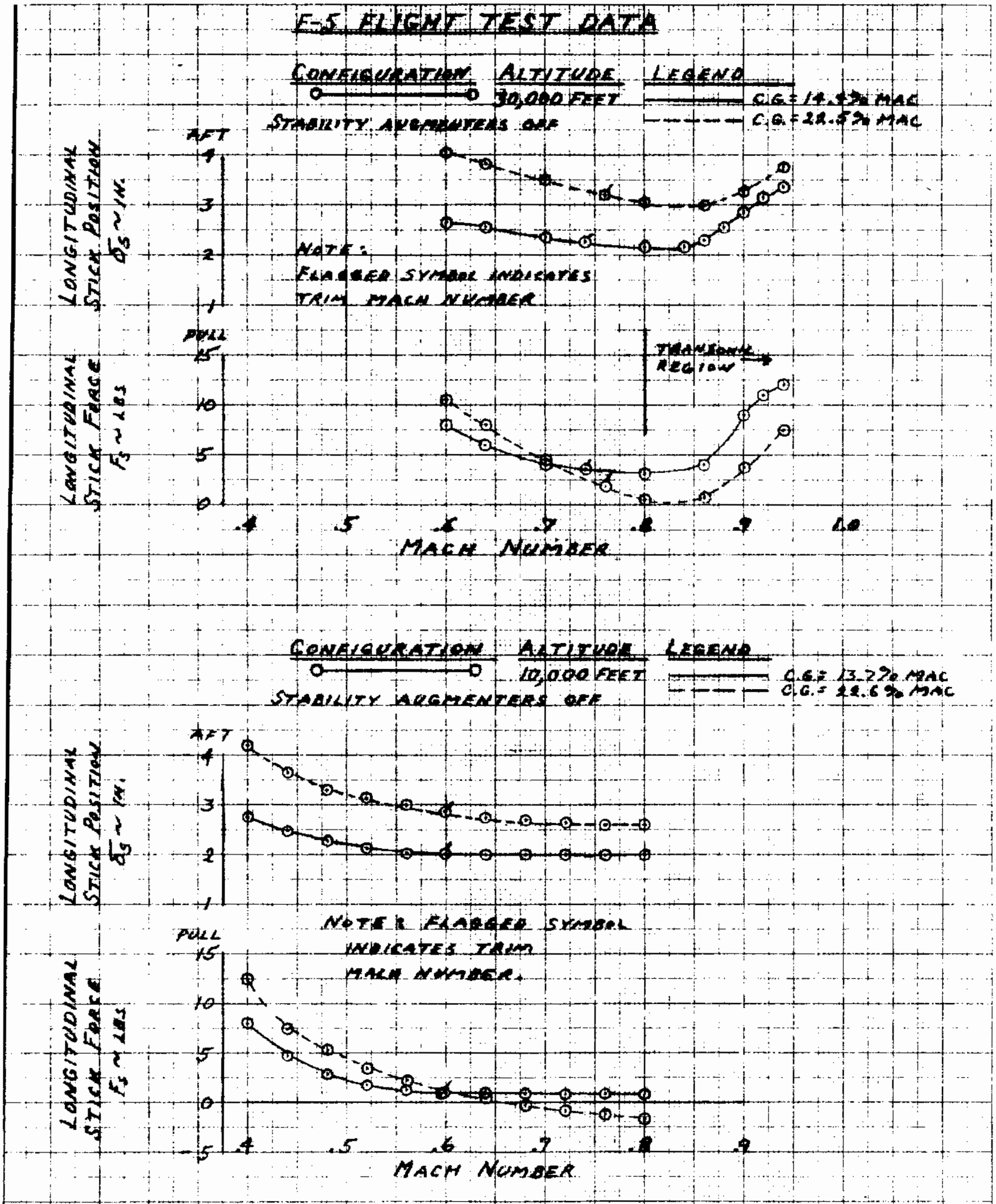
Recommendation

It is recommended that the specification of trimmer and throttle use be revised as follows:

"Trimmer control not moved from the trim setting by the crew, but throttle control initially moved, to increase or decrease speed, then held constant, and this movement must cover the extreme power variations with speed and"

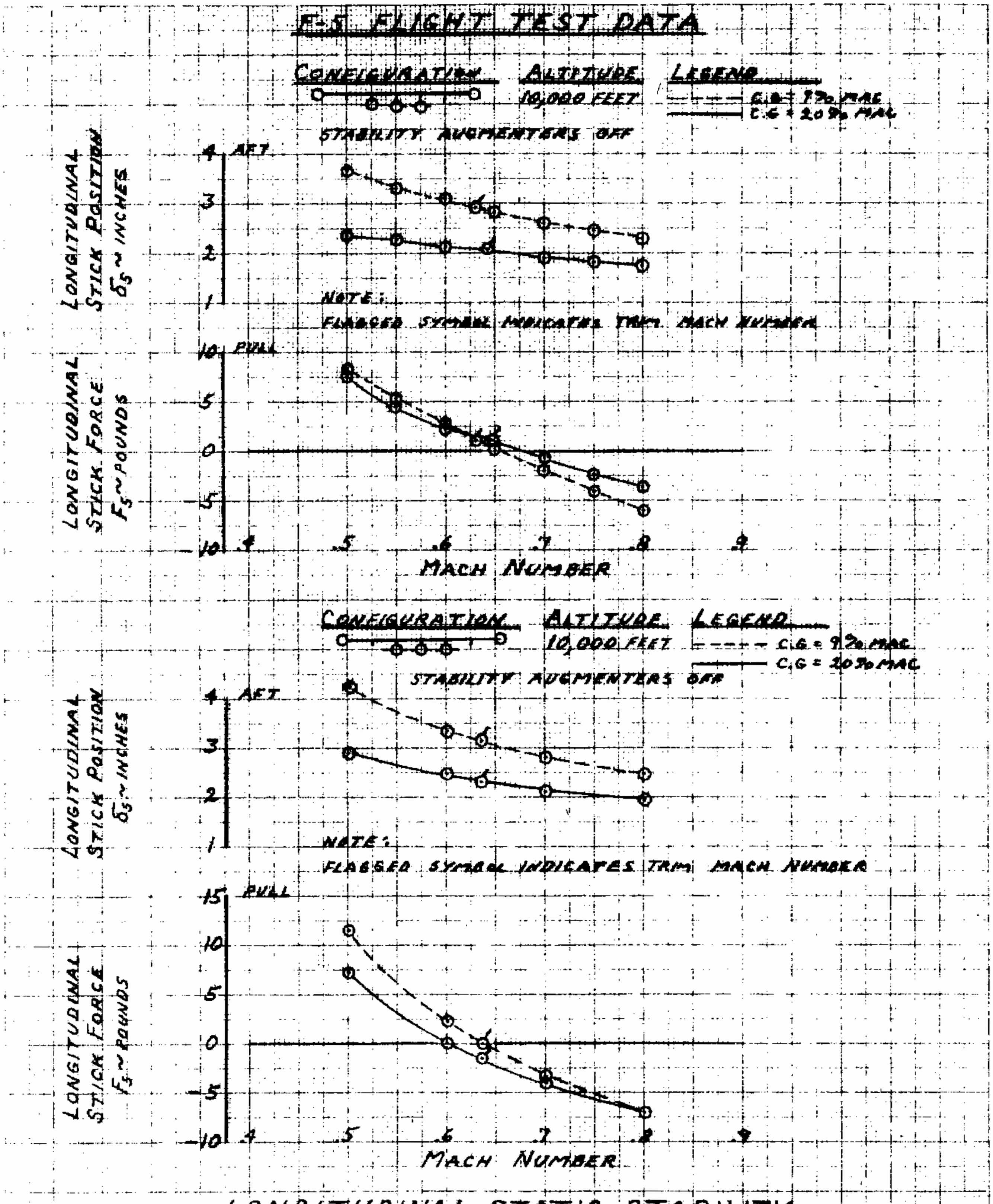


LONGITUDINAL STATIC STABILITY
FIGURE 1 (3.3.1.1)



LONGITUDINAL STATIC STABILITY

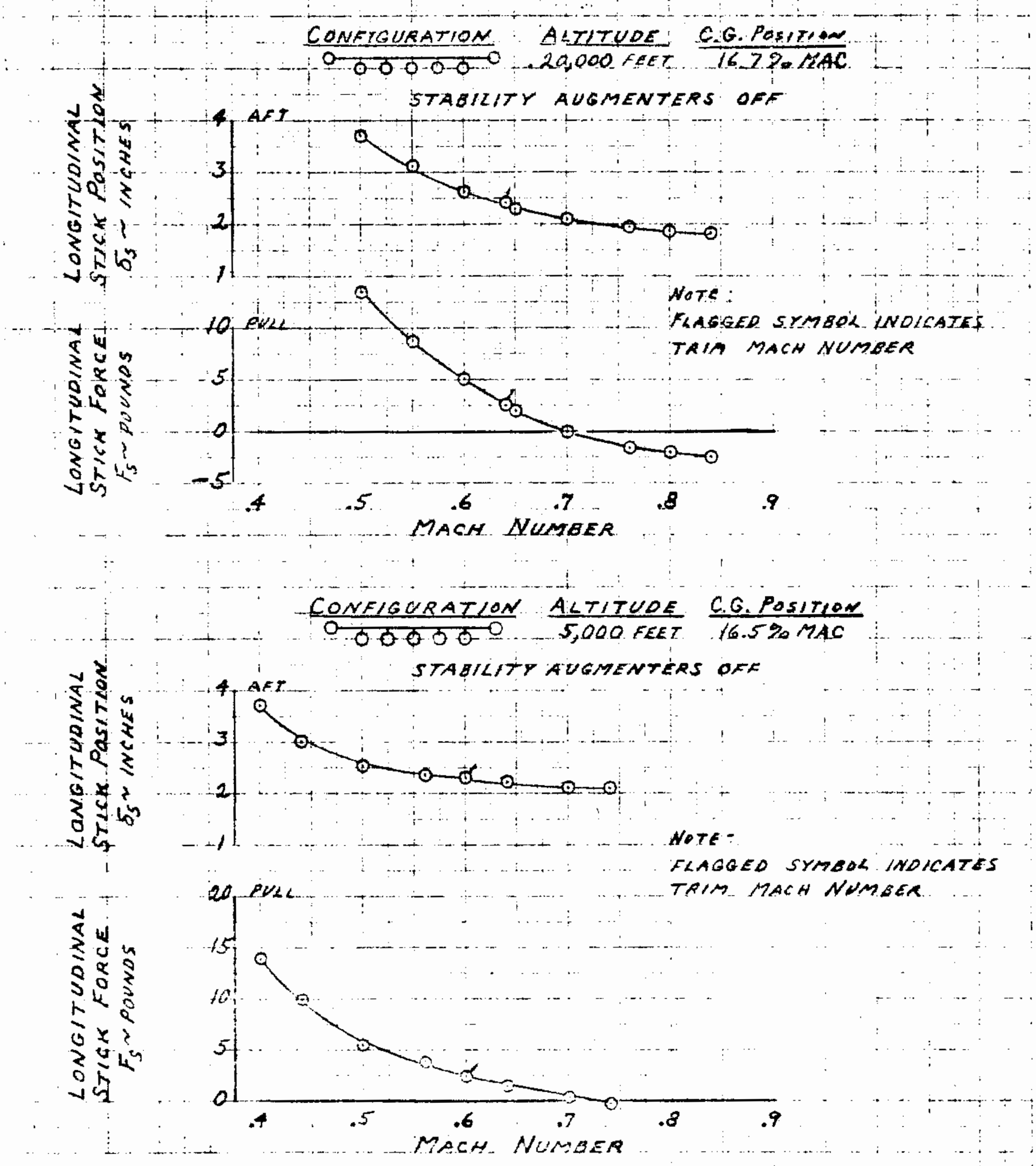
FIGURE 2 (3.2.1.1)



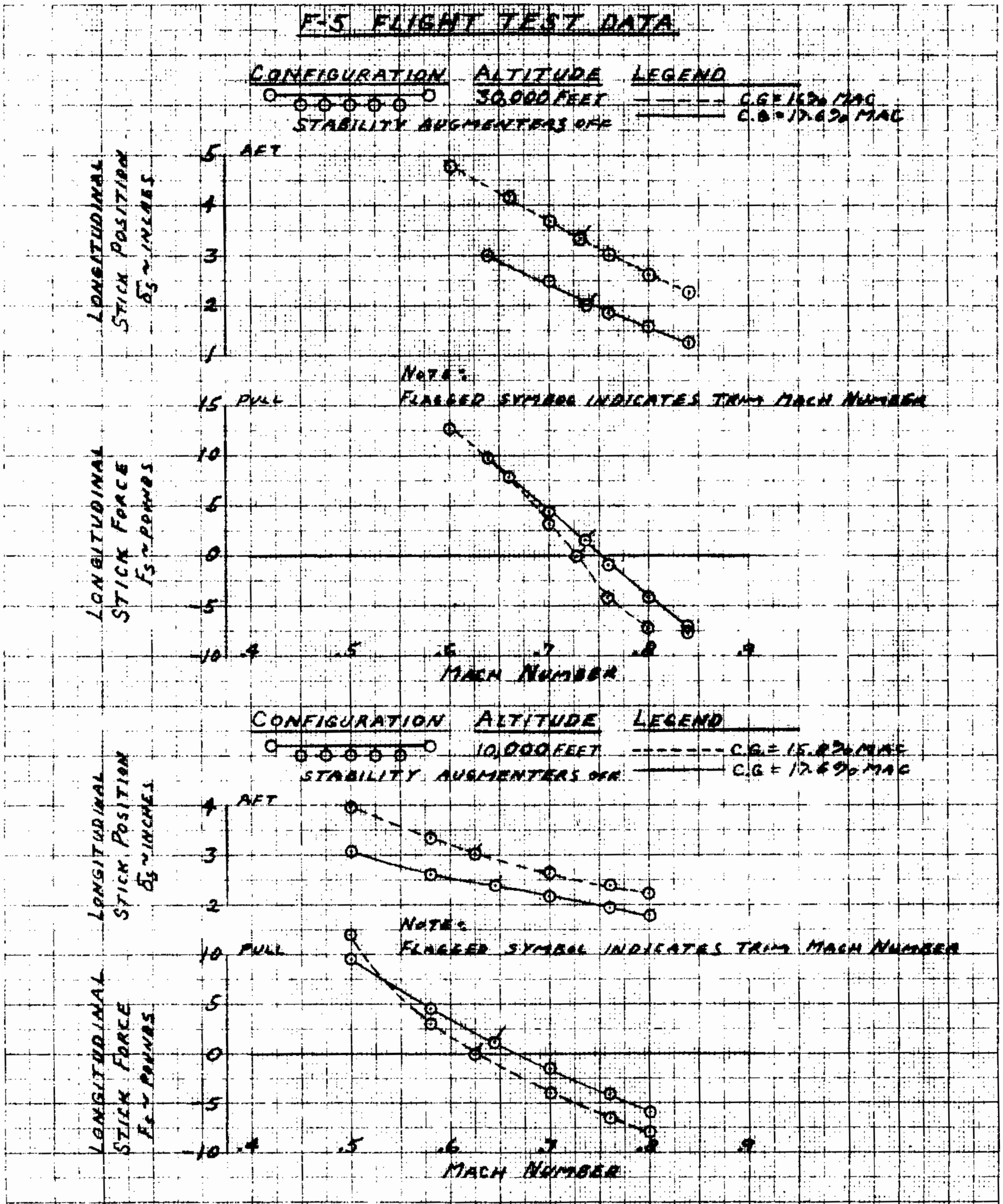
LONGITUDINAL STATIC STABILITY

FIGURE 3 (3.2.1.1)

F-5 FLIGHT TEST DATA



LONGITUDINAL STATIC STABILITY
FIGURE 4 (5.2.1.1)



LONGITUDINAL STATIC STABILITY
FIGURE 5 (3.2.1.1)

Requirement

Paragraph 3.2.1.1.1 Relaxation in transonic flight. The requirements of 3.2.1.1 may be relaxed in the transonic speed range provided any divergent airplane motions or reversals in slope of elevator control force and elevator control position with speed are gradual and not objectionable to the pilot. In no case, however, shall the requirements of 3.2.1.1 be relaxed more than the following:

a. Levels 1 and 2 - For center-stick controllers, no local force gradient shall be more unstable than 3 pounds per 0.01 M nor shall the force change exceed 10 pounds in the unstable direction. The corresponding limits for wheel controllers are 5 pounds per 0.01 M and 15 pounds, respectively.

b. Level 3 - For center-stick controllers, no local force gradient shall be more unstable than 6 pounds per 0.01 M nor shall the force ever exceed 20 pounds in the unstable direction. The corresponding limits for wheel controllers are 10 pounds per 0.01 M and 30 pounds, respectively.

This relaxation does not apply to Level 1 for any Flight Phase which requires prolonged transonic operation.

Comparion

The data presented in Figures 1 (3.2.1.1) and 2 (3.2.1.1) are used to compare the transonic flight longitudinal static stability characteristics of the F-5 with the requirements of this paragraph. The unstable gradients of longitudinal stick force and longitudinal stick position are apparent in the transonic region. The instability falls within the requirements of 3 pounds per 0.01 M. Pilot comments noted no unusual handling qualities. Consequently, it is concluded that the F-5 characteristics compare favorably with this paragraph.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.1.1.2 Elevator control force variations during rapid speed changes. When the airplane is accelerated and decelerated rapidly through the operational speed range and through the transonic speed range by the most critical combination of changes in power, actuation of deceleration devices, steep turns and pullups, the magnitude and rate of the associated trim change shall not be so great as to cause difficulty in maintaining the desired load factor by normal pilot techniques.

Comparison

Flight test maneuvers of speed brake extensions, rapid power changes, landing gear retractions and extensions, flaps operation, pullups and wind-up turns were performed with the F-5 to qualitatively check the associated trim changes. No adverse pilot comments were reported. Therefore, it is concluded that the F-5 characteristics compare favorably with this paragraph.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.1.2 Phugoid stability. The long-period airspeed oscillations which occur when the airplane seeks a stabilized airspeed following a disturbance shall meet the following requirements:

a. Level 1 ------ ζ_p at least 0.04

b. Level 2 ------ ζ_p at least 0

c. Level 3 ------ T_2 at least 55 seconds.

These requirements apply with the elevator control free and also with it fixed. They need not be met transonically in cases where 3.2.1.1.1 permits relaxation of the static stability requirement.

Comparison

No flight test data are available for a comparison of the F-5 characteristics with this paragraph. However, no adverse pilot comments regarding the phugoid mode have been reported; consequently, it is considered that the F-5 possesses acceptable phugoid stability. An analytical approach was applied to calculate the F-5 phugoid characteristics to compare with the requirements of this paragraph. Two representative altitudes and a range of Mach numbers from 0.4 to 1.2 were analyzed to encompass Levels 1 and 2. These analytical results are presented in the following tabulation.

Altitude	Mach Number	ω_{n_p}	ζ_p	Required Level	Compliance
10,000	.4	.1084	.0329	1	no
10,000	.8	.0815	.0974	1	yes
10,000	.95	.0810	.8564	1	yes
10,000	1.05	.0592	.5507	2	yes
30,000	.6	.0809	.0299	1	no
30,000	.85	.0772	.0666	1	yes
30,000	.9	.0987	.0689	1	yes
30,000	.95	.0654	.4474	1	yes
30,000	1.2	.0366	.3709	2	yes

The method of analysis used was taken from Page III-11 and Pages IV-4 and -5 of Reference 2. The transonic drag rise with speed (C_{D_u}) is the primary contributor to the high damping ratio obtained in this speed region. The natural frequency and damping ratio of the approximation to the phugoid are explicitly discussed in Reference 2.

Resolution

A partial disagreement between the F-5 and the requirements are exhibited in that two flight conditions exhibited damping ratios slightly less than .04 based on analytical results.

In the absence of specific flight test data and pilot comments, and based on engineering judgment, it is concluded that the requirement as it stands may be considered valid.

Recommendation

None

Requirement

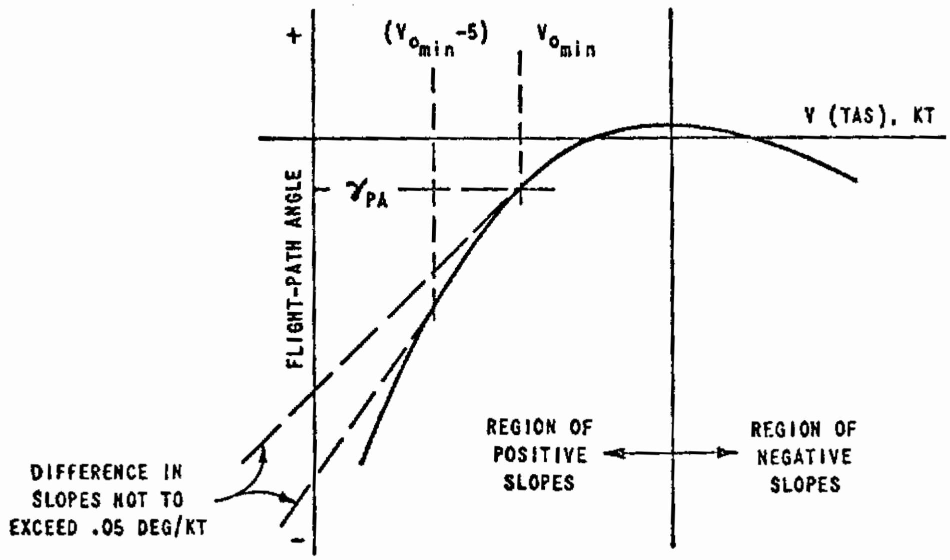
Paragraph 3.2.1.3 Flight-path stability. Flight-path stability is defined in terms of flight-path-angle change where the airspeed is changed by the use of the elevator control only (throttle setting not changed by the crew). For the landing approach Flight Phase, the flight-path-angle versus true-airspeed curve shall have a local slope at $V_{o_{min}}$ which is negative or less positive than:

a. Level 1 ------ 0.06 degrees/knot

b. Level 2 ------ 0.15 degrees/knot

c. Level 3 ------ 0.24 degrees/knot.

The thrust settting shall be that required for the normal approach glide path at $V_{o_{min}}$. The slope of the flight-path angle versus airspeed curve at 5 knots slower than $V_{o_{min}}$ shall not be more than 0.05 degrees per knot more positive than the slope at $V_{o_{min}}$, as illustrated by:



Comparison

Flight test data from level-flight unaccelerated stalls were used for comparison of F-5 flight path stability with the requirements of this paragraph. The flight path angle, γ, was obtained by the equation,

$$\gamma = \sin^{-1} \frac{\text{vertical speed}}{\text{true airspeed}}.$$

Vertical speed was obtained by differentiating pressure altitude with respect to time. No radio altimeter data were available. Figures 1 (3.2.1.3) to 3 (3.2.1.3) present the data in final form. Three different store loading configurations were analyzed, representing Category C and Flight Phase (PA). $V_{o_{min}}$ is defined as the minimum approach speed obtained from Reference 1 and $V_{o_{min}}-5$ is defined as the minimum approach speed minus 5 knots.

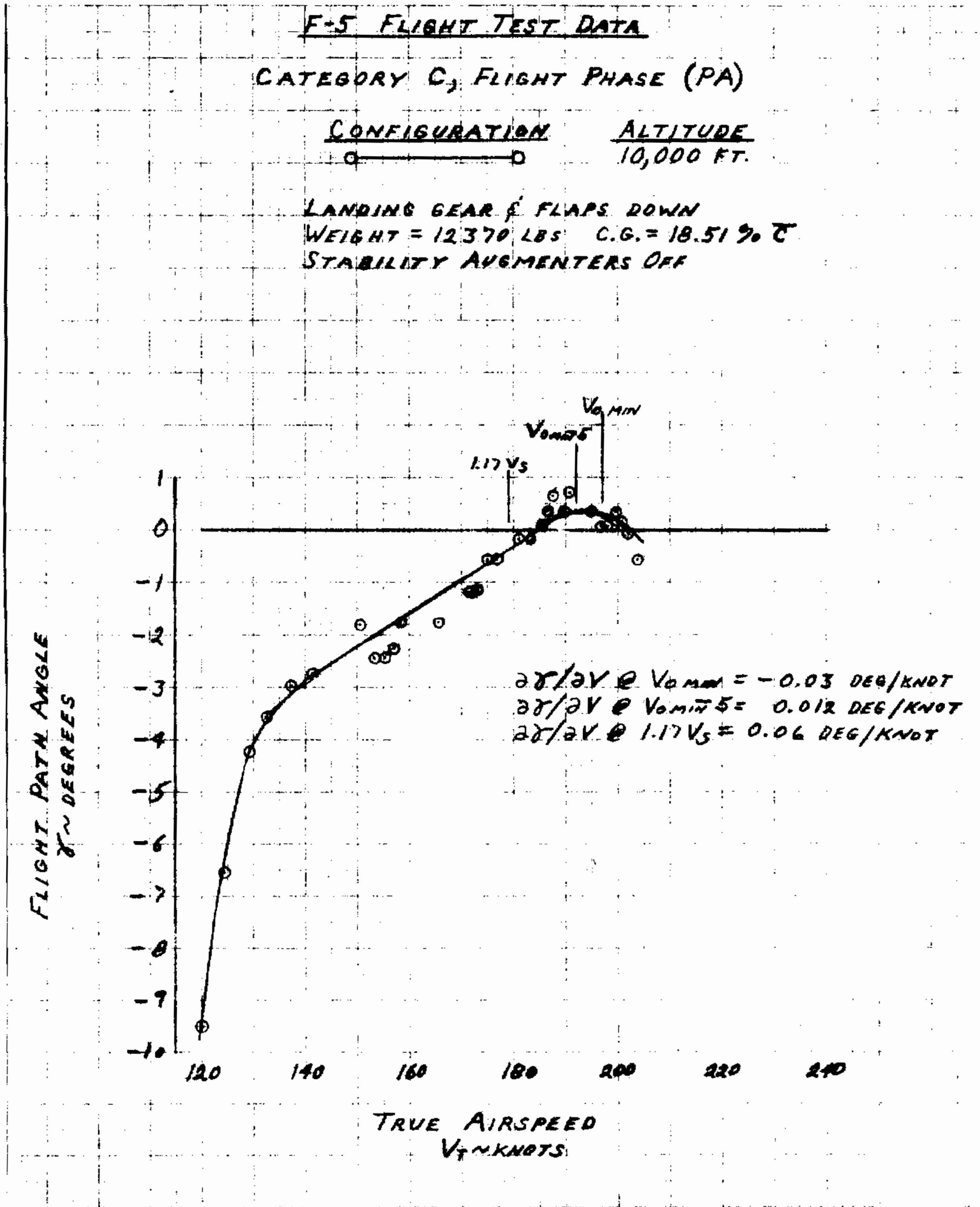
During the flight test program involving the Netherlands version of the F-5 (NF-5), tests were performed to determine the minimum acceptable approach speed. This was done to calibrate the NF-5 angle-of-attack system. Results of these tests showed the minimum acceptable approach speed to be 1.17 V_S. At this approach speed the requirements of this paragraph show agreement with the F-5 flight path characteristics, thereby validating the paragraph requirements.

Resolution

None

Recommendation

None



FLIGHT PATH STABILITY

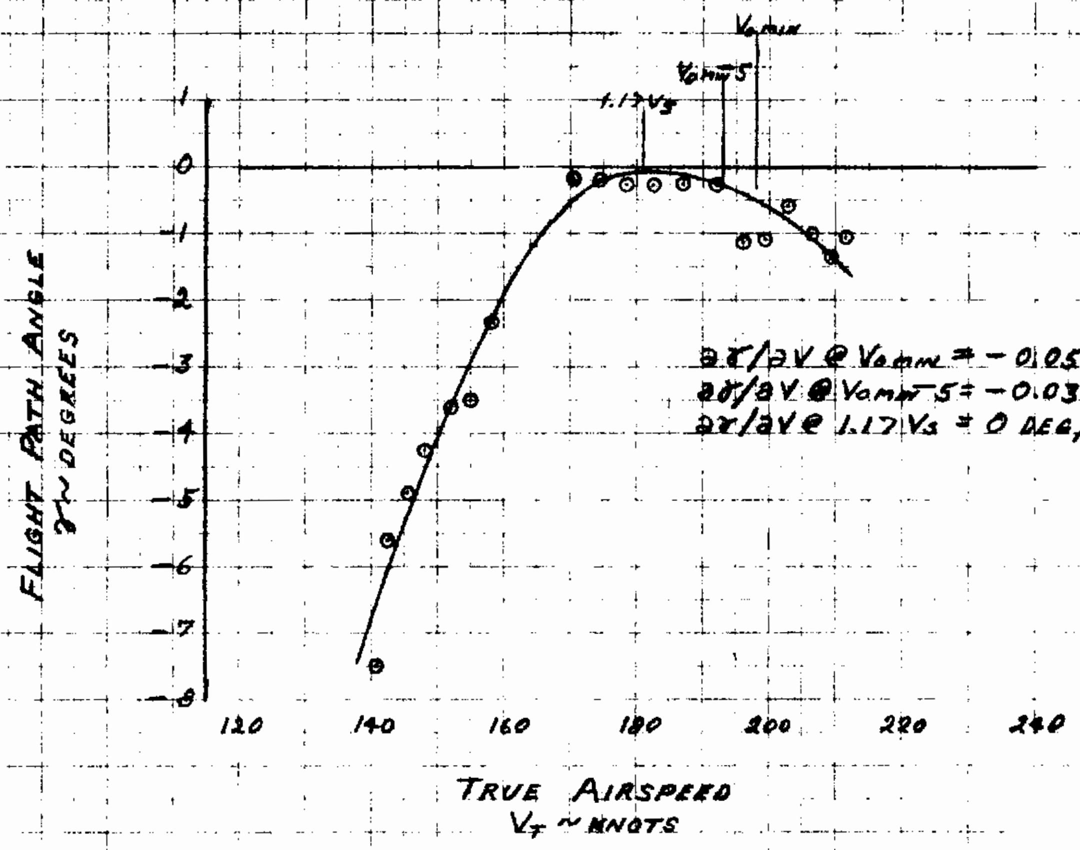
FIGURE 1 (3.2.1.3)

F-5 FLIGHT TEST DATA

CATEGORY C, FLIGHT PHASE (PA)

CONFIGURATION	ALTITUDE
	10,000 FT.

LANDING GEAR & FLAPS DOWN
WEIGHT = 12,820 LBS   C.G. = 22.26% $\bar{c}$
STABILITY AUGMENTERS OFF



FLIGHT PATH STABILITY

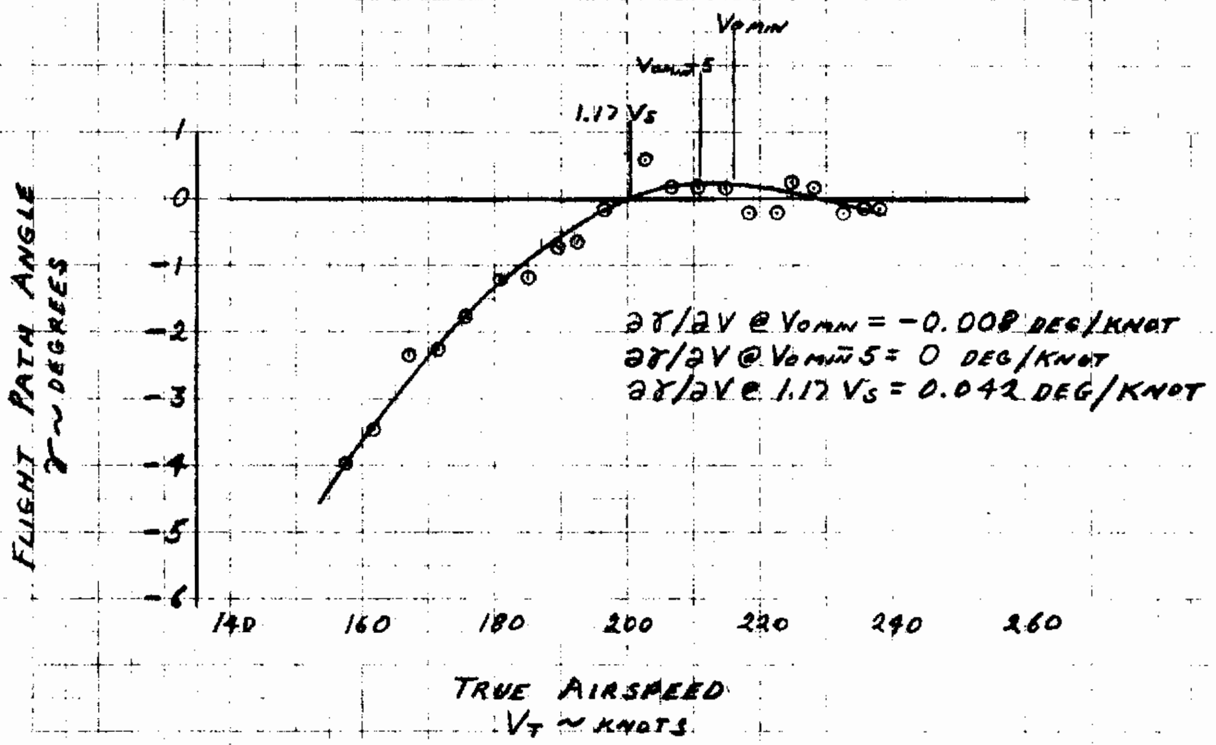
FIGURE 2 (3.2.1.3)

F-5 FLIGHT TEST DATA

CATEGORY C, FLIGHT PHASE (PA)

CONFIGURATION	ALTITUDE
	10,000 FT.

LANDING GEAR & FLAPS DOWN
WEIGHT = 14,780 LBS. C.G = 15.02% $\bar{c}$
STABILITY AUGMENTERS OFF



FLIGHT PATH STABILITY

FIGURE 3 (3.2.1.3)

Requirement

Paragraph 3.2.2 Longitudinal maneuvering characteristics

Paragraph 3.2.2.1 Short-period response. The short-period response of angle of attack which occurs at approximately constant speed, and which may be produced by abrupt elevator control inputs, shall meet the requirements of 3.2.2.1.1 and 3.2.2.1.2. These requirements apply, with the cockpit control free and with it fixed, for responses of any magnitude that might be experienced in service use. If oscillations are nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation. In addition to meeting the numerical requirements of 3.2.2.1.1 and 3.2.2.1.2, the contractor shall show that the airplane has acceptable response characteristics in atmospheric disturbances.

Comparison

This is an introductory paragraph. The comparison data are shown in paragraphs 3.2.2.1.1 and 3.2.2.1.2.

Resolution

None

Recommendation

This paragraph should specify that during the airplane design stage the contractor shall submit an analysis which indicates that the airplane has acceptable response characteristics in atmospheric disturbances. One such analytical procedure is presented and recommended in Paragraph 3.7.5.

Requirement

Paragraph 3.2.2.1.1 Short-period frequency and acceleration sensitivity. The short-period undamped natural frequency, $\omega_{n_{sp}}$, shall be within the limits shown in figures 1, 2, and 3. If suitable means of directly controlling normal force are provided, the lower bounds on $\omega_{n_{sp}}$ and n/α of figure 3 may be relaxed if approved by the procuring activity.

Comparison

Data for comparison of the F-5 characteristics with this paragraph were taken from flight tests of clean, two-store and four-external-store configurations. The dynamic longitudinal stability tests, from which $\omega_{n_{sp}}$ was extracted, were performed stick fixed and stick free with stability augmenters on and off at altitudes from 12,000 to 30,000 feet, M=0.85 and M = 0.95. The n/α data were extracted from wind-up turn maneuvers at the same flight conditions as the dynamic longitudinal stability tests. Figures 1 (3.2.2.1.1) through 3 (3.2.2.1.1) present the data reduced for comparison of the F-5 airplane characteristics with this paragraph. As noted in the figures, the solid symbols are augmenters off and should be compared with Level 2 requirements. Figures 4 (3.2.2.1.1) and 5 (3.2.2.1.1) present T-38A airplane flight test data obtained in the same fashion as the F-5 data. These dynamic longitudinal stability tests were performed at M=0.3 to M = 0.9 at 10,000 feet altitude and at M = 1.1 and 1.28 at 25,000 feet altitude.

Reduced flight test results, as presented in the figures, exhibit favorable comparison of the F-5 and T-38A airplane characteristics with the requirements of this paragraph. Pilot comments indicated acceptable qualities exist. This exhibits agreement and accord between the specification requirements and F-5/T-38 flight test data.

Resolution

None

Recommendation

None

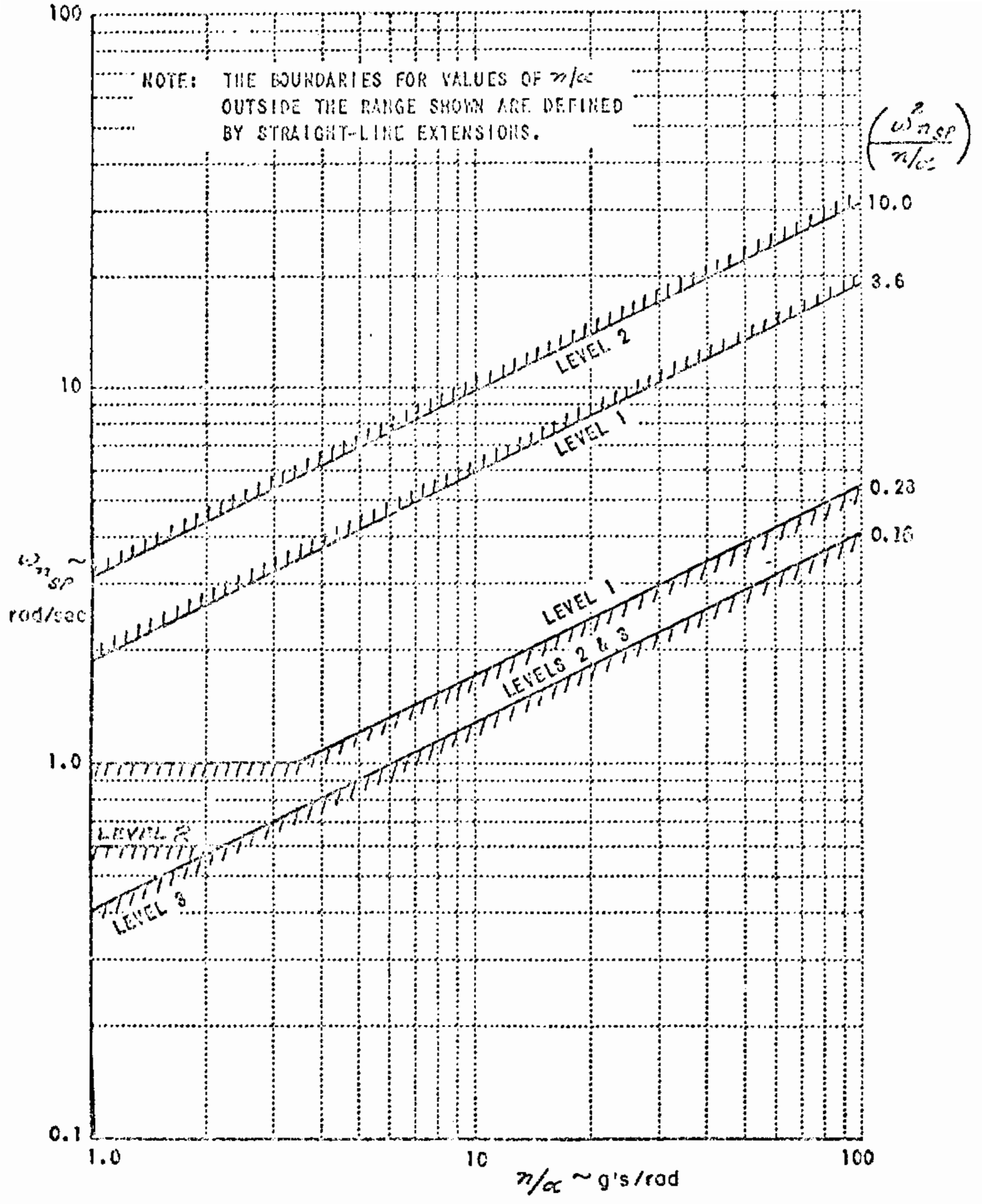


FIGURE 1. Short-Period Frequency Requirements - Category A Flight Phases

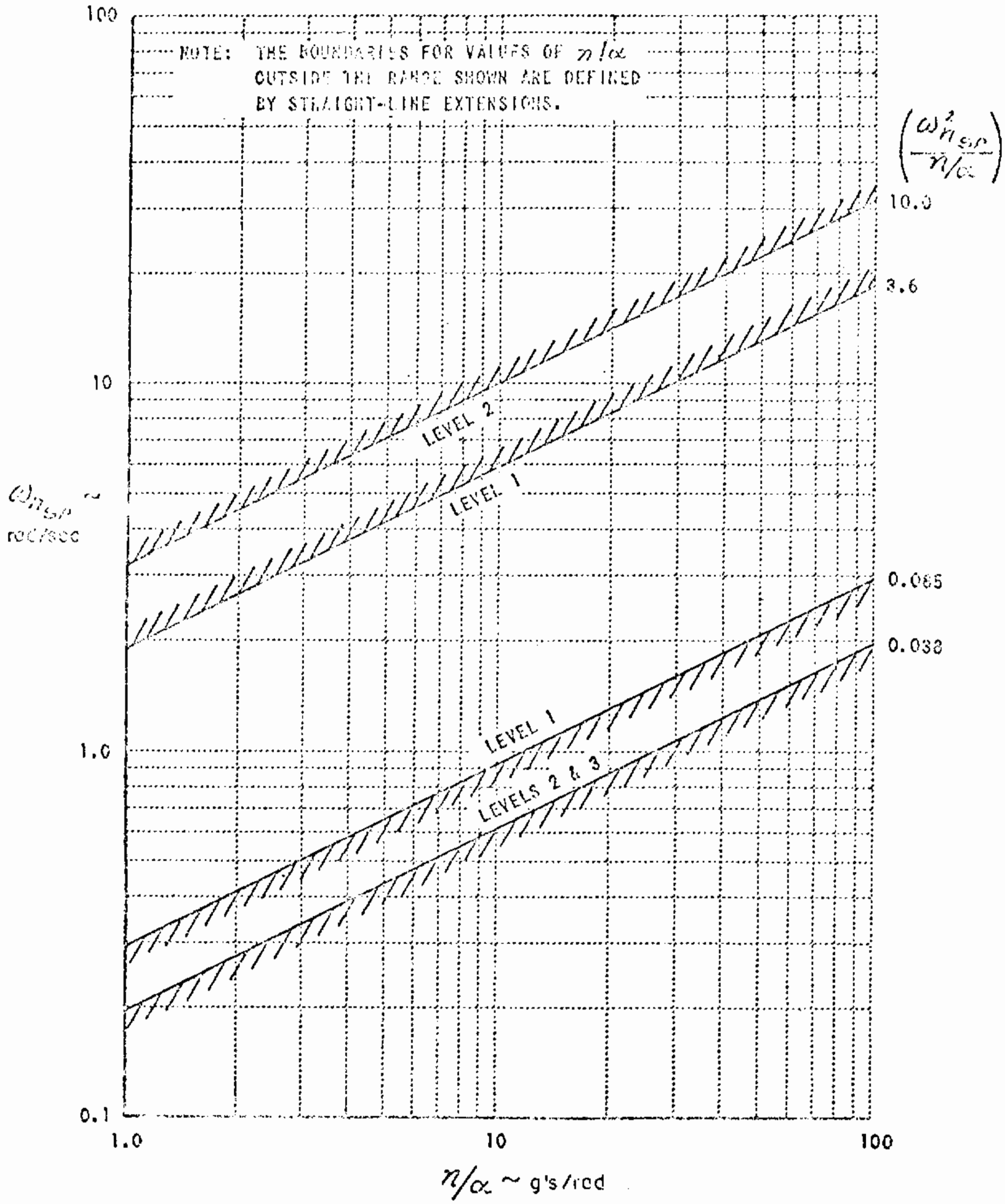


FIGURE 2. Short-Period Frequency Requirements - Category B Flight Phases

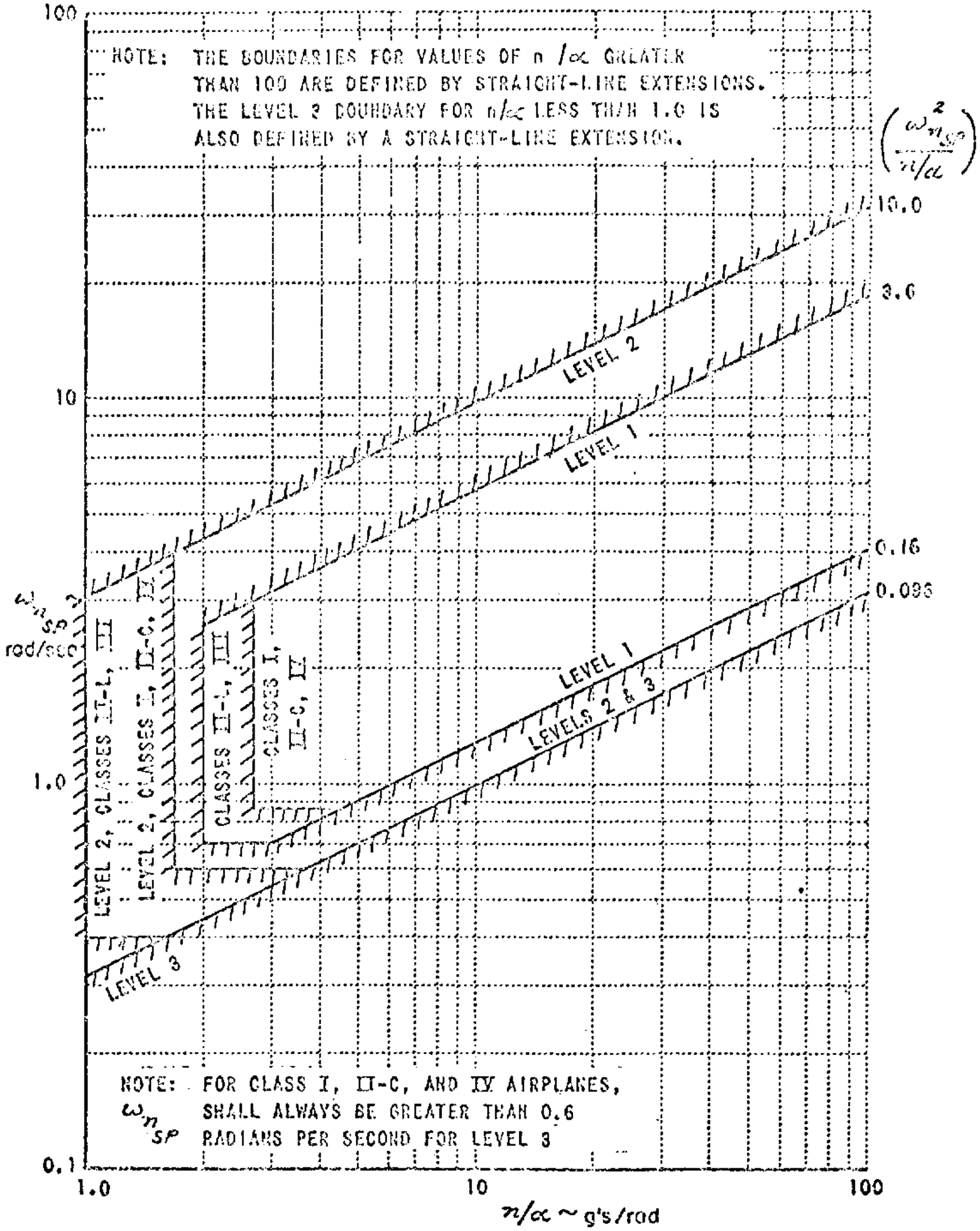
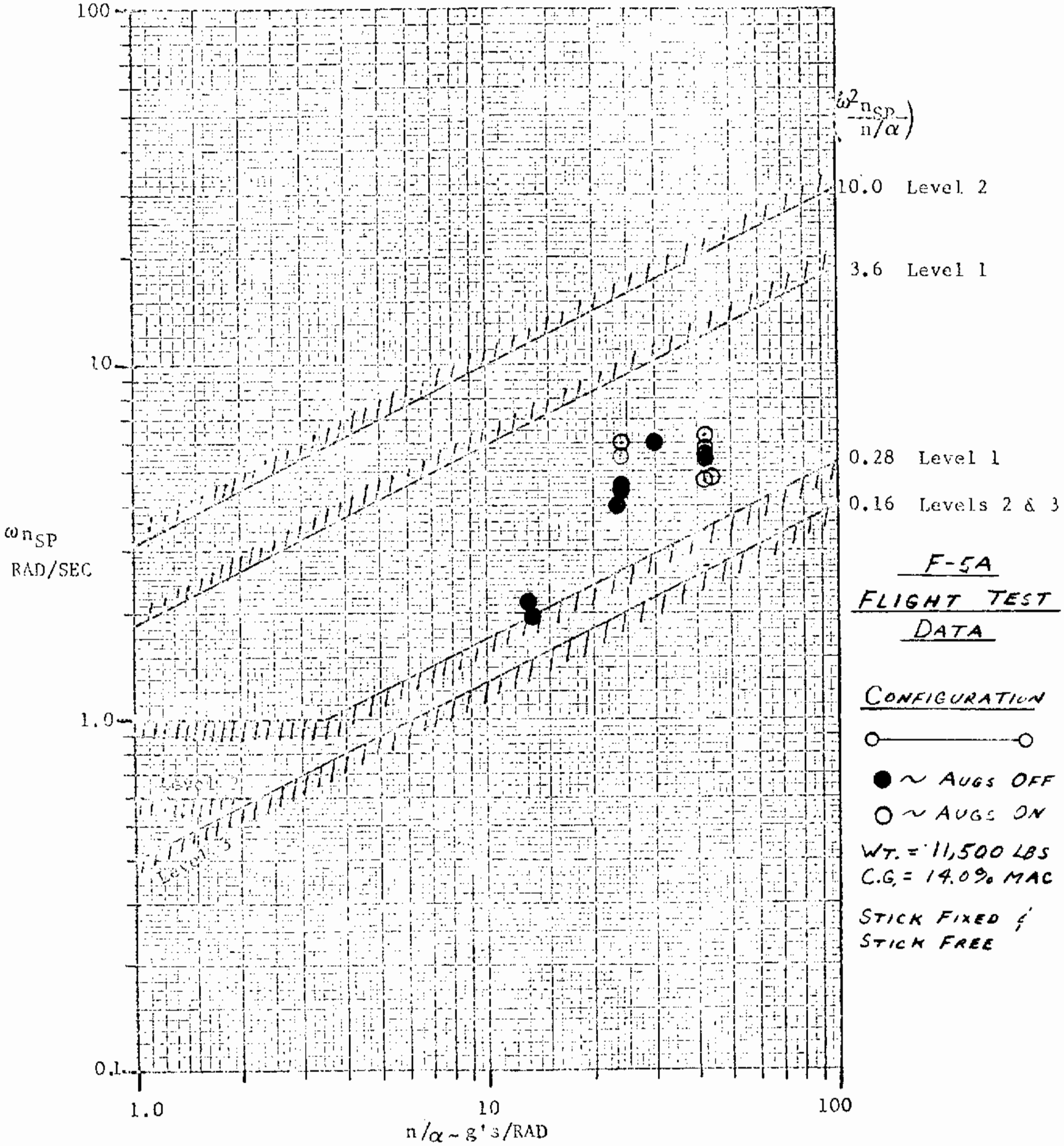


FIGURE 3. Short-Period Frequency Requirements - Category C Flight Phases

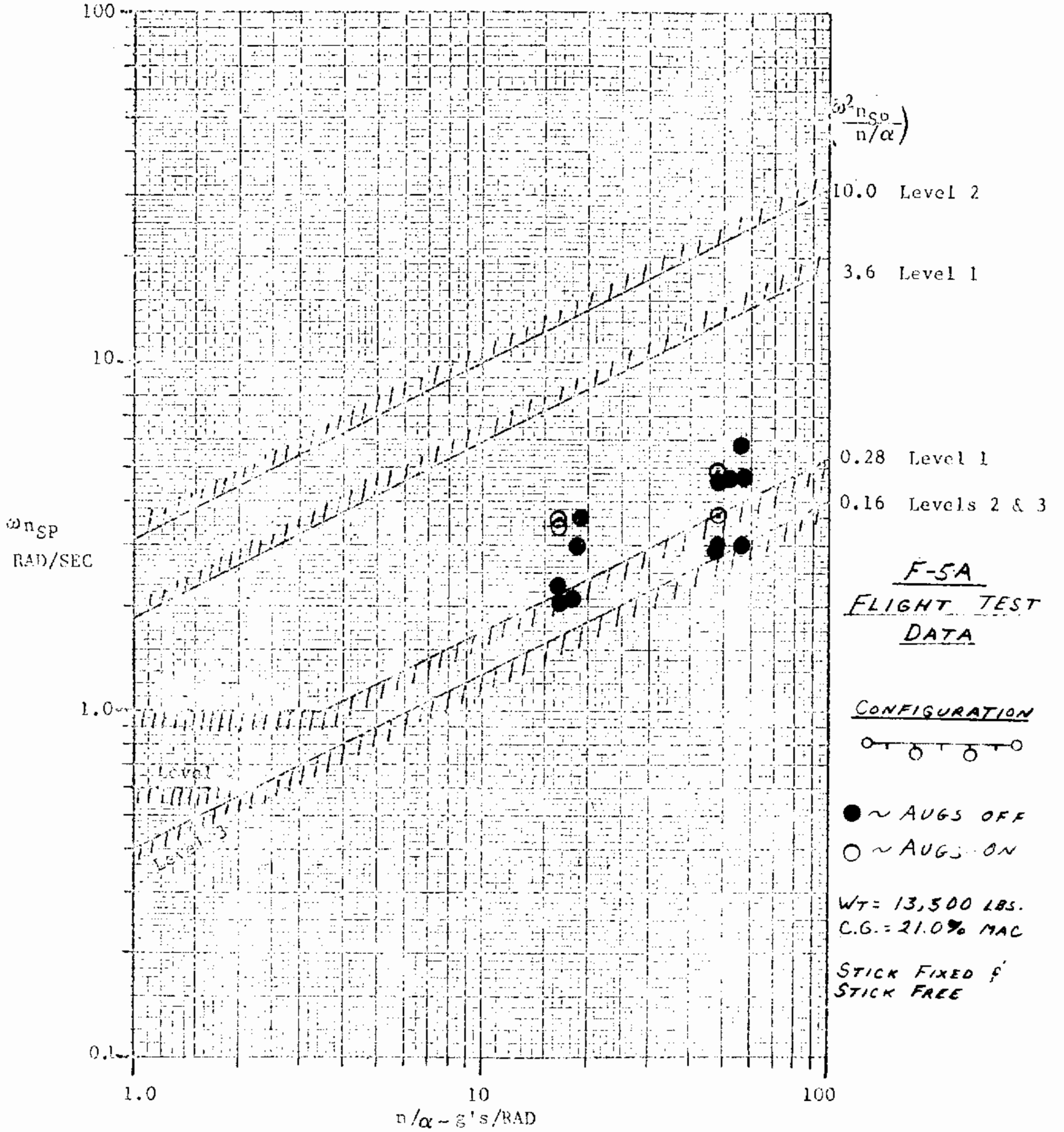
MIL-F-8785B (ASG)
Short-Period Frequency Requirements
Category A Flight Phases



SHORT PERIOD FREQUENCY AND
ACCELERATION SENSITIVITY
FIGURE 1 (3.2.2.1.1)

REF: FIGURE 1

MIL-F-8785B (ASG)
Short-Period Frequency Requirements
Category A Flight Phases

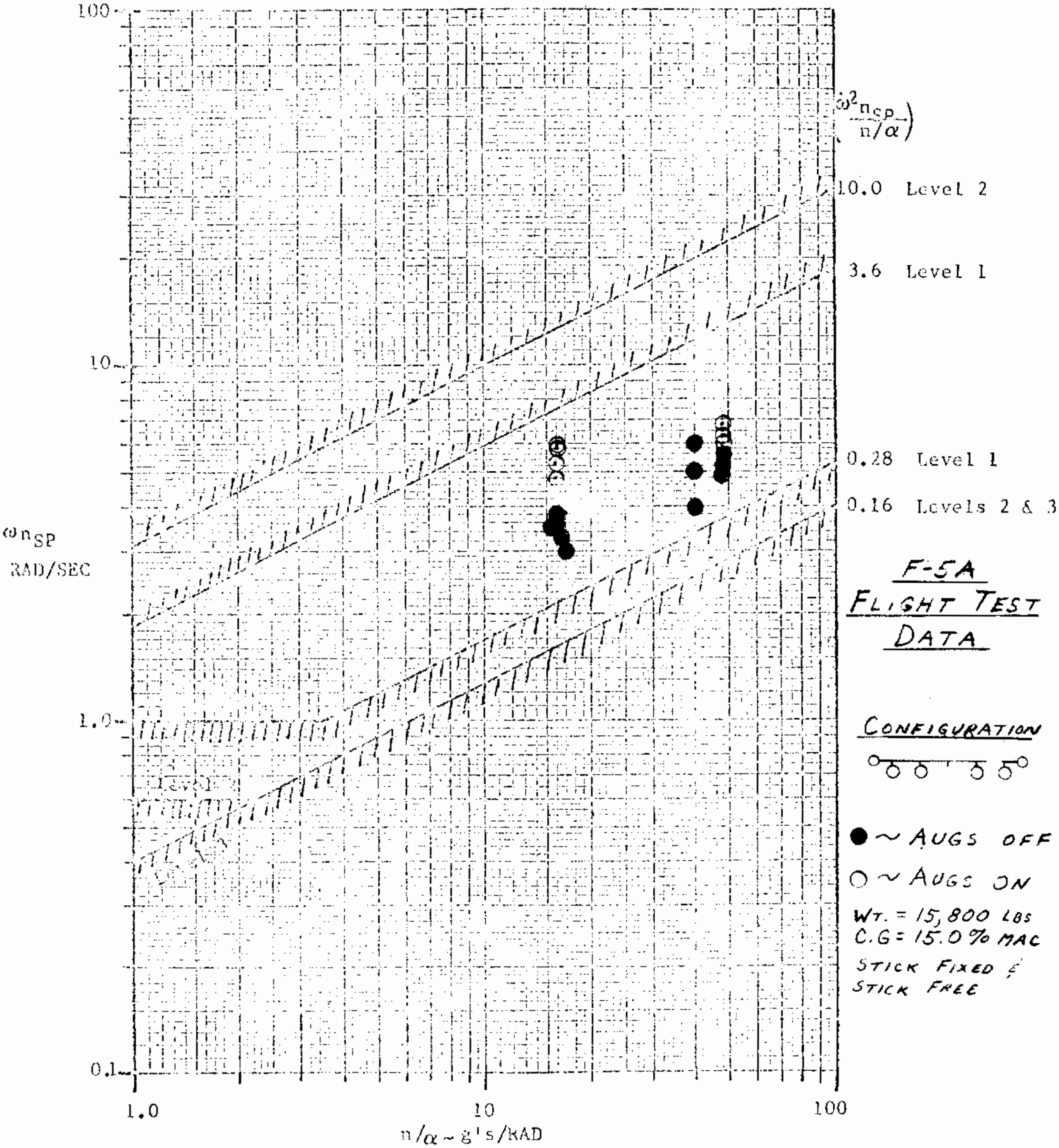


SHORT PERIOD FREQUENCY AND
ACCELERATION SENSITIVITY

FIGURE 2 (3.2.2.1.1)

REF: FIGURE 1

MIL-F-8785B (ASG)
Short-Period Frequency Requirements
Category A Flight Phases

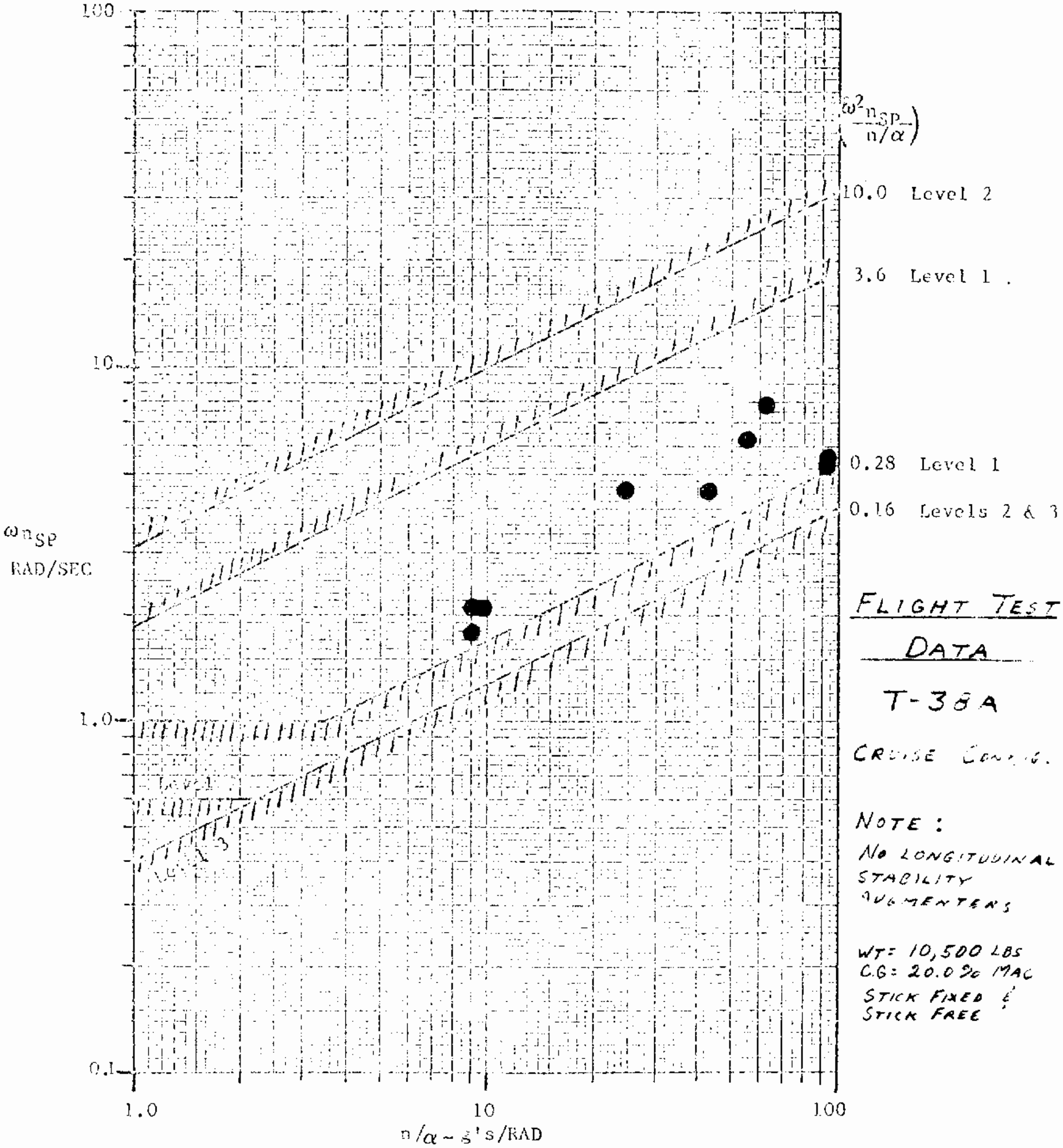


SHORT PERIOD FREQUENCY AND ACCELERATION SENSITIVITY

FIGURE 3 (3.2.2.1.1)

REF: FIGURE 1

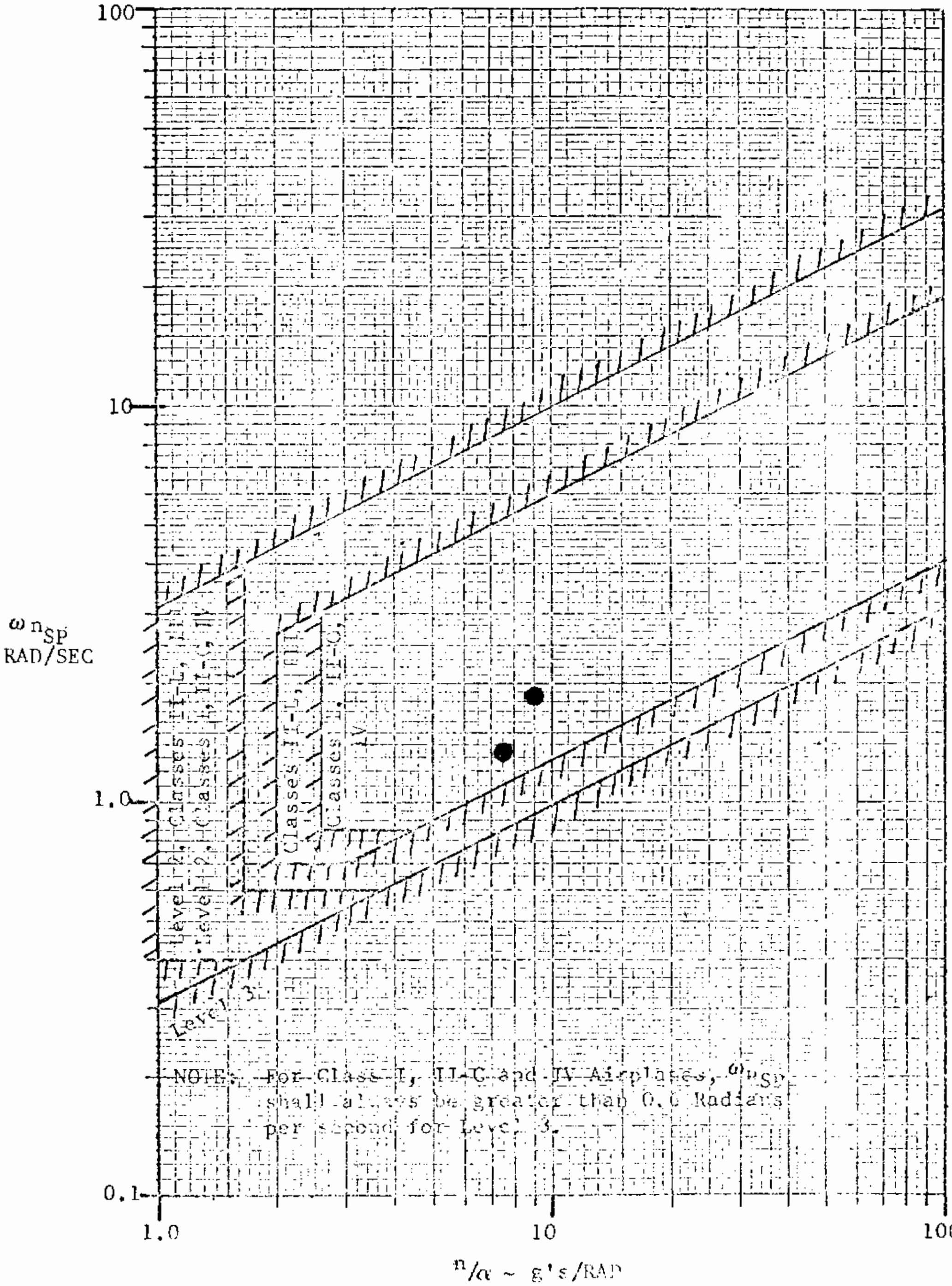
MIL-F-8785B (ASG)
Short-Period Frequency Requirements
Category A Flight Phases



SHORT PERIOD FREQUENCY AND ACCELERATION SENSITIVITY

FIGURE 4 (3.2.2.1.1)

MIL-F-8785B (ASG)
Short-Period Frequency Requirements
Category C Flight Phases



FLIGHT TEST DATA

T-38A

LANDING CONFIG.

NOTE:
NO LONGITUDINAL STABILITY AUGMENTERS
WT = 10,200 LBS
CG. = 20.0% MAC
STICK FIXED & STICK FREE

SHORT PERIOD FREQUENCY AND ACCELERATION SENSITIVITY

FIGURE 5 (3.2.2.1.1)

Requirement

Paragraph 3.2.2.1.2 Short-period damping. The short-period damping ratio, ζ_{SP}, shall be within the limits of table IV.

TABLE IV. Short-period Damping Ratio Limits

Level	Category A and C Flight Phases		Category B Flight Phases	
	Minimum	Maximum	Minimum	Maximum
1	0.35	1.30	0.30	2.00
2	0.25	2.00	0.20	2.00
3	0.15*	---	0.15*	---

*May be reduced at altitudes above 20,000 feet if approved by the procuring activity.

Comparison

Data from F-5 and T-38A flight tests were reduced to compare the damping magnitudes with the requirements of this paragraph. Figures 1 (3.2.2.1.2) through 3 (3.2.2.1.2) present, respectively, F-5 clean configuration, two-, four- and five-store configurations, and T-38A airplane results. With the stability augmentation system on, the F-5 exhibits compliance with Level 1 requirements. With the stability augmentation system off, the F-5 damping ratio generally migrates between a value of 0.6 and 0.2 depending on the Mach number, altitude, and whether the stick is fixed or free.

Similarly, the T-38A damping ratio migrates between a value of 0.5 and 0.15. The Level 2 minimum requirement is $\zeta = 0.25$. Consequently, non-compliance is partially exhibited. However, pilot comments indicated that satisfactory damping prevailed in all cases and noted that, with augmenter on, the damping was greater as exhibited in the flight test results.

A disagreement exists between the acceptable minimum characteristics of the F-5 and the requirements of this paragraph.

Resolution

The specified minimum damping ratio of 0.25 for Level 2, Category A and C Flight Phases is not considered to be minimum for acceptable handling qualities as shown by the F-5 flight test results and pilot acceptability. Good handling qualities can prevail for damping ratio as low as 0.2.

Recommendation

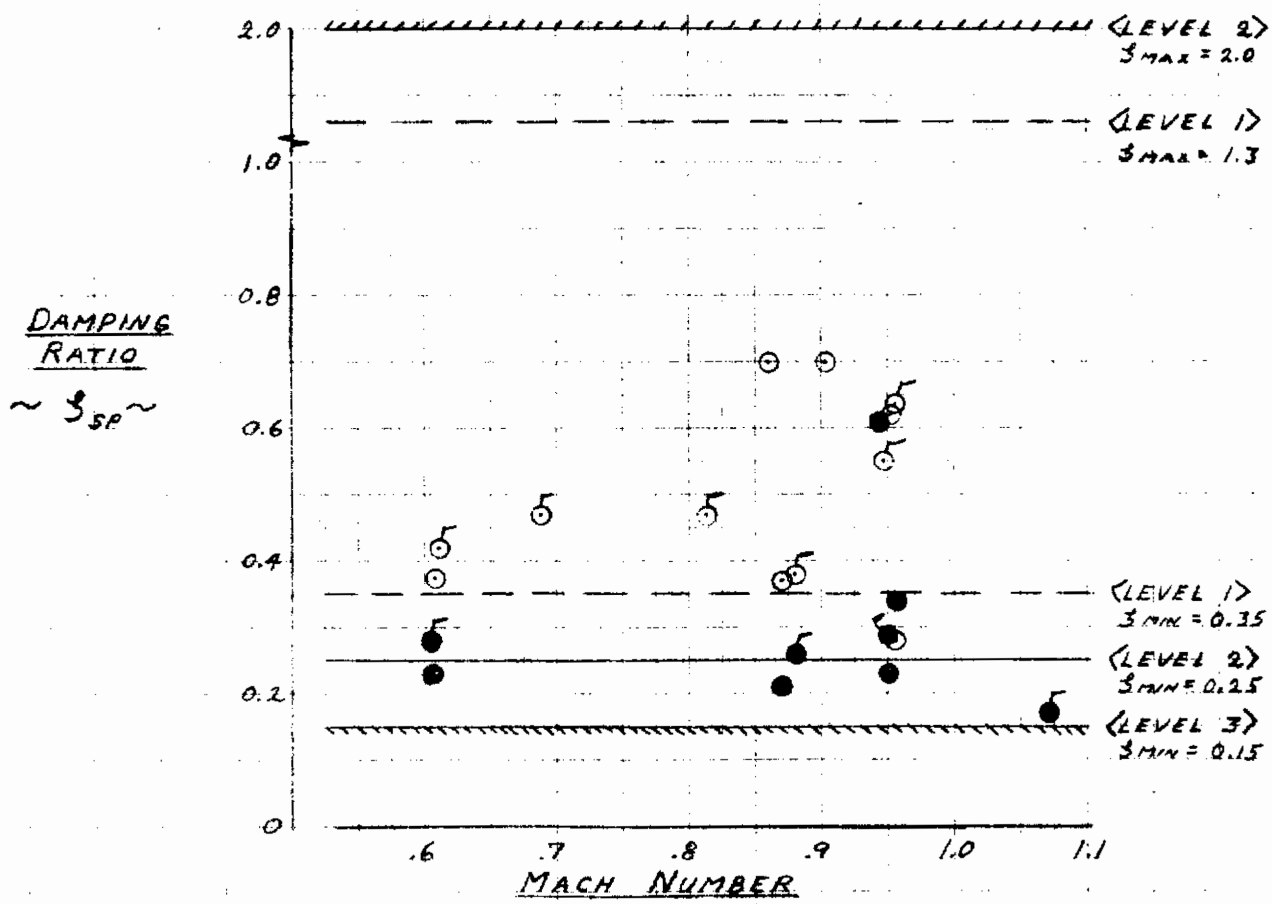
It is recommended that Table IV, Short-period Damping Ratio Limits, be changed to make the Level 2, Category A and C Flight Phases minimum requirement equal to 0.2 instead of 0.25 as currently specified.

F-5A
FLIGHT TEST DATA

CONFIGURATION — ALTITUDE 10,000 TO 30,000 FEET

FLAGGED SYMBOL ~ STICK FIXED
UNFLAGGED SYMBOL ~ STICK FREE

⊙ ~ LONGITUDINAL STABILITY AUGMENTERS ON
● ~ LONGITUDINAL STABILITY AUGMENTERS OFF



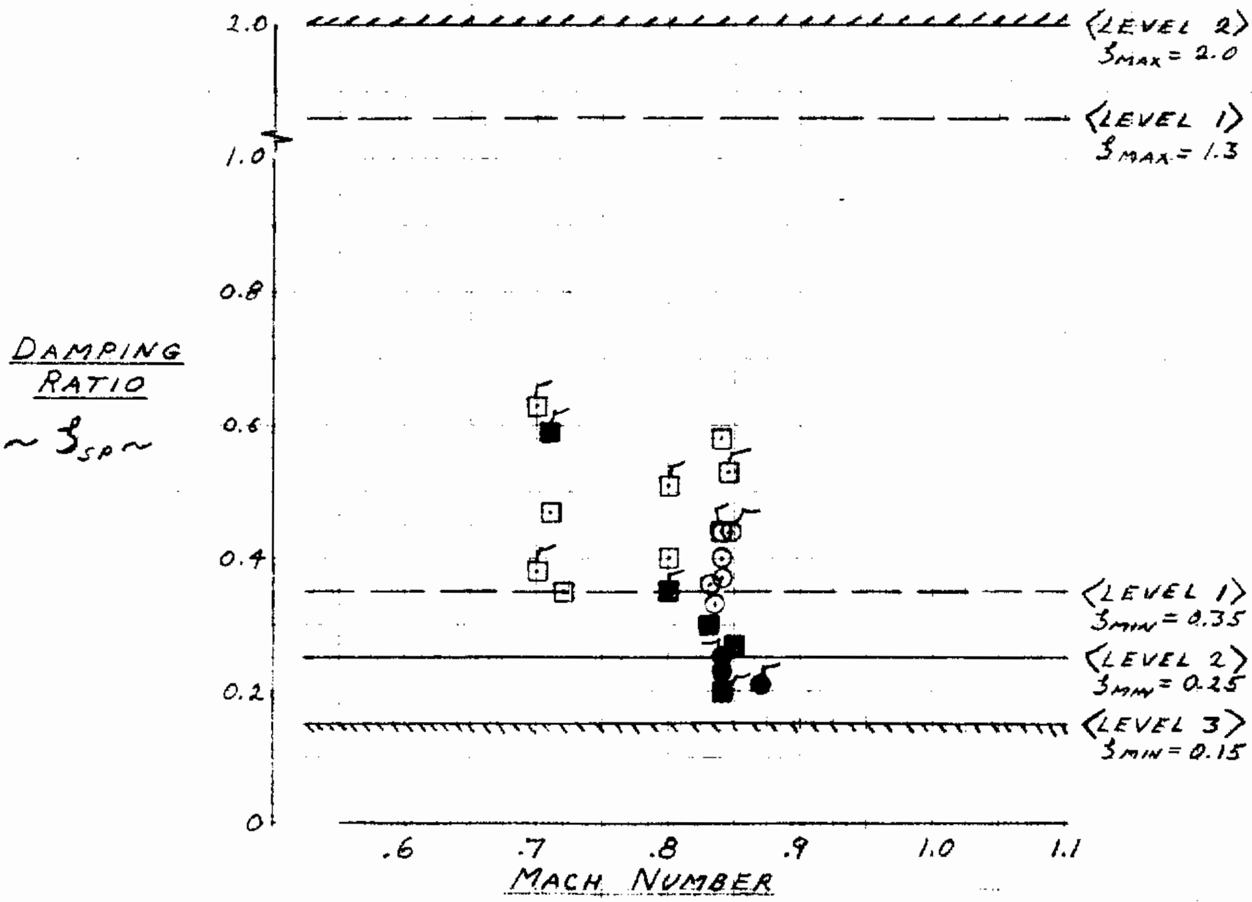
SHORT PERIOD DAMPING

FIGURE 1 (3.2.2.1.2)

F-5A FLIGHT TEST DATA

SYMBOL	CONFIGURATION	ALTITUDE
○		10,000 TO 30,000 FEET
□		

FLAGGED SYMBOL ~ STICK FIXED
UNFLAGGED SYMBOL ~ STICK FREE
CLEAR SYMBOLS INDICATE STABILITY AUGMENTERS ON
FILLED SYMBOLS INDICATE STABILITY AUGMENTERS OFF



SHORT PERIOD DAMPING

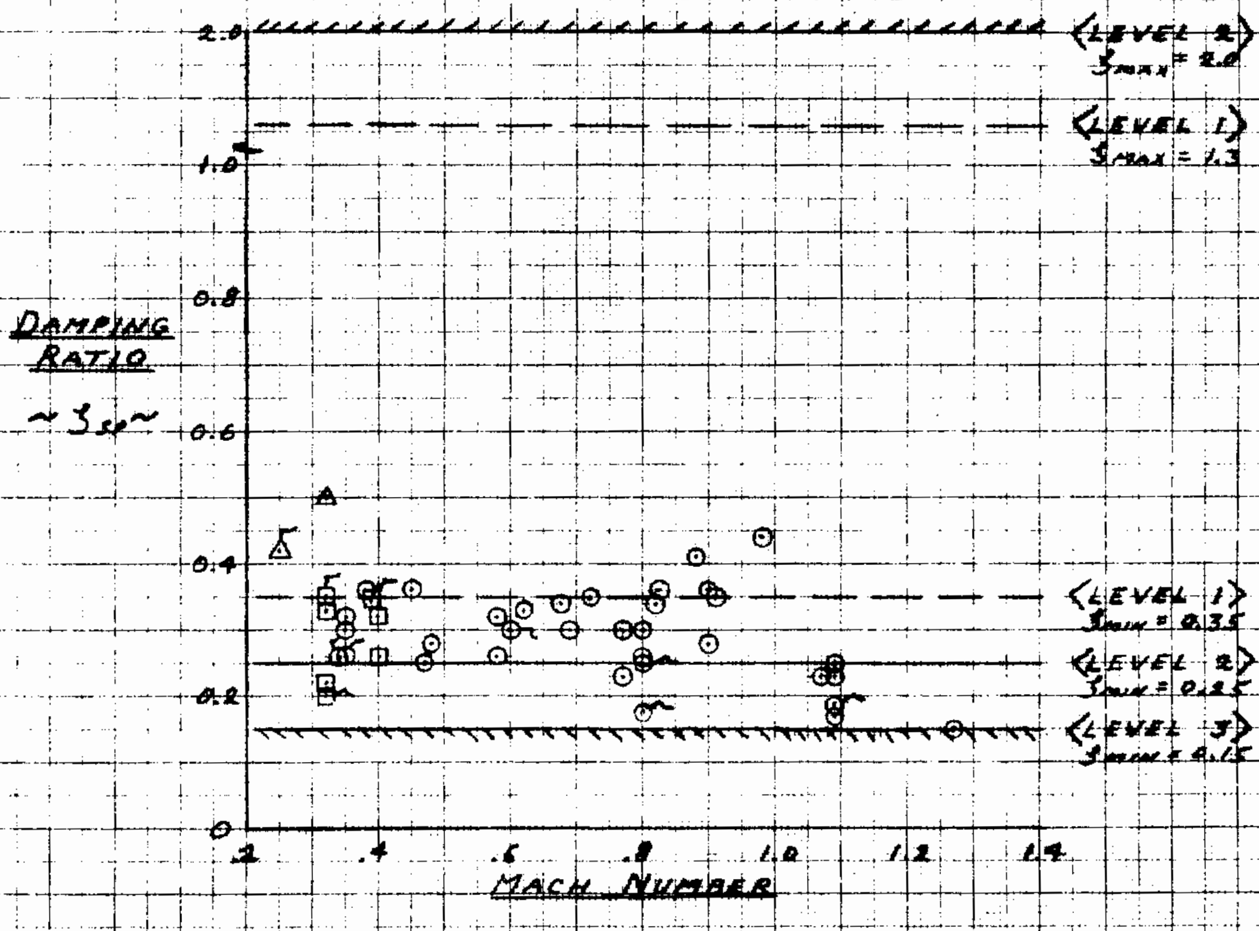
FIGURE 2 (3.2.2.1.2)

T-38A
FLIGHT TEST DATA

SYMBOL	CONFIGURATION	ALTITUDE
○	CRUISE	10,000 TO 25,000 FEET
□	TAKEOFF	10,000 FEET
△	LANDING	10,000 FEET

FLAGGED SYMBOL ~ STICK FIXED
UNFLAGGED SYMBOL ~ STICK FREE

NOTE:
NO LONGITUDINAL STABILITY AUGMENTATION SYSTEM AVAILABLE ON T-38 AIRPLANES.



SHORT PERIOD DAMPING

FIGURE 3 (3.2.2.1.2)

Requirement

Paragraph 3.2.2.1.3 Residual oscillations. Any sustained residual oscillations shall not intefere with the pilot's ability to perform the tasks required in service use of the airplane. For Levels 1 and 2, oscillations in normal acceleration at the pilot's station greater than ±0.05 g will be considered excessive for any Flight Phase, as will pitch attitude oscillations greater than ±3 mils for Category A Flight Phases requiring precision control of attitude. These requirements shall apply with the elevator control fixed and with it free.

Comparison

Stick-free flight test data of the F-5 obtained from a longitudinal stick pulse at one flight condition were utilized to compare with the requirements of this paragraph. Figure 1 (3.2.2.1.3) exhibits a favorable comparison of normal load factor measured at the c.g. It is expected that the residual oscillations at the pilot's station will fall within the ±0.05 g requirement. No flight test data are available for comparison with the pitch attitude oscillations requirement. The flight test data acquisition system used did not allow ±3 mils to be discernible.

In flight demonstration for compliance with this requirement, the sensitivity of the pitch attitude instrumentation and data acquisition must be considered.

Resolution

None

Recommendation

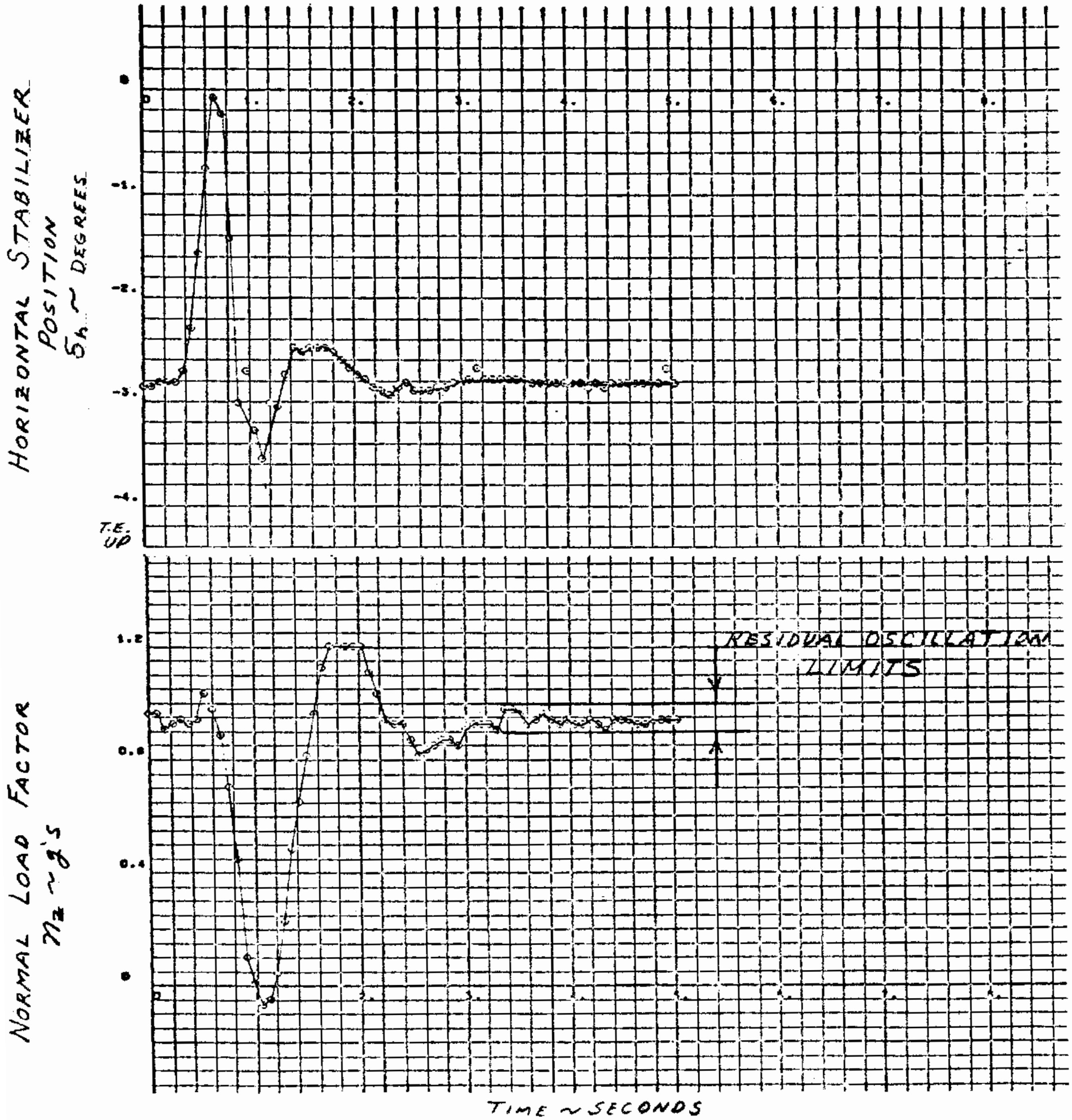
None

F-5 FLIGHT TEST TIME HISTORY OF A LONGITUDINAL STICK PULSE

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
	18,600 FEET	0.704	17,560 LBS.	7.9% M.A.C

CATEGORY A, FLIGHT PHASE (GA)

NOTE: STABILITY AUGMENTERS ON
STICK FREE



RESIDUAL OSCILLATIONS

FIGURE 1 (3.2.2.1.3)

Requirement

Paragraph 3.2.2.2 Control feel and stability in maneuvering flight. In steady turning flight and in pullups at constant speed, increasing pull forces and aft motion of the elevator control and airplane-nose-up deflection of the elevator surface are required to maintain increases in normal acceleration throughout the range of service load factors defined in 3.1.8.4. Increases in push force, forward control motion, and airplane-nose-down deflection of the elevator surface are required to maintain reductions of normal acceleration in pushovers.

Comparison

Wind-up turn and "British Pushover"* maneuvers data from F-5 flight tests of clean, two-store, and four-store configurations were utilized for comparison with the requirements of this paragraph. Figures 1 (3.2.2.2) through 15 (3.2.2.2) present these data for a Mach number range of 0.5 to 0.95, and for an altitude range of 10,000 to 30,000 feet. All data presented demonstrate increasing pull forces, aft motion of the control stick, and trailing edge up motion of the horizontal stabilizer causing airplane-nose-up motion for increases in normal load factor and the reverse for decreases in normal load factor. Therefore, a favorable comparison of the F-5 control feel and stability characteristics with the requirements of this paragraph is demonstrated.

Resolution

None

Recommendation

None

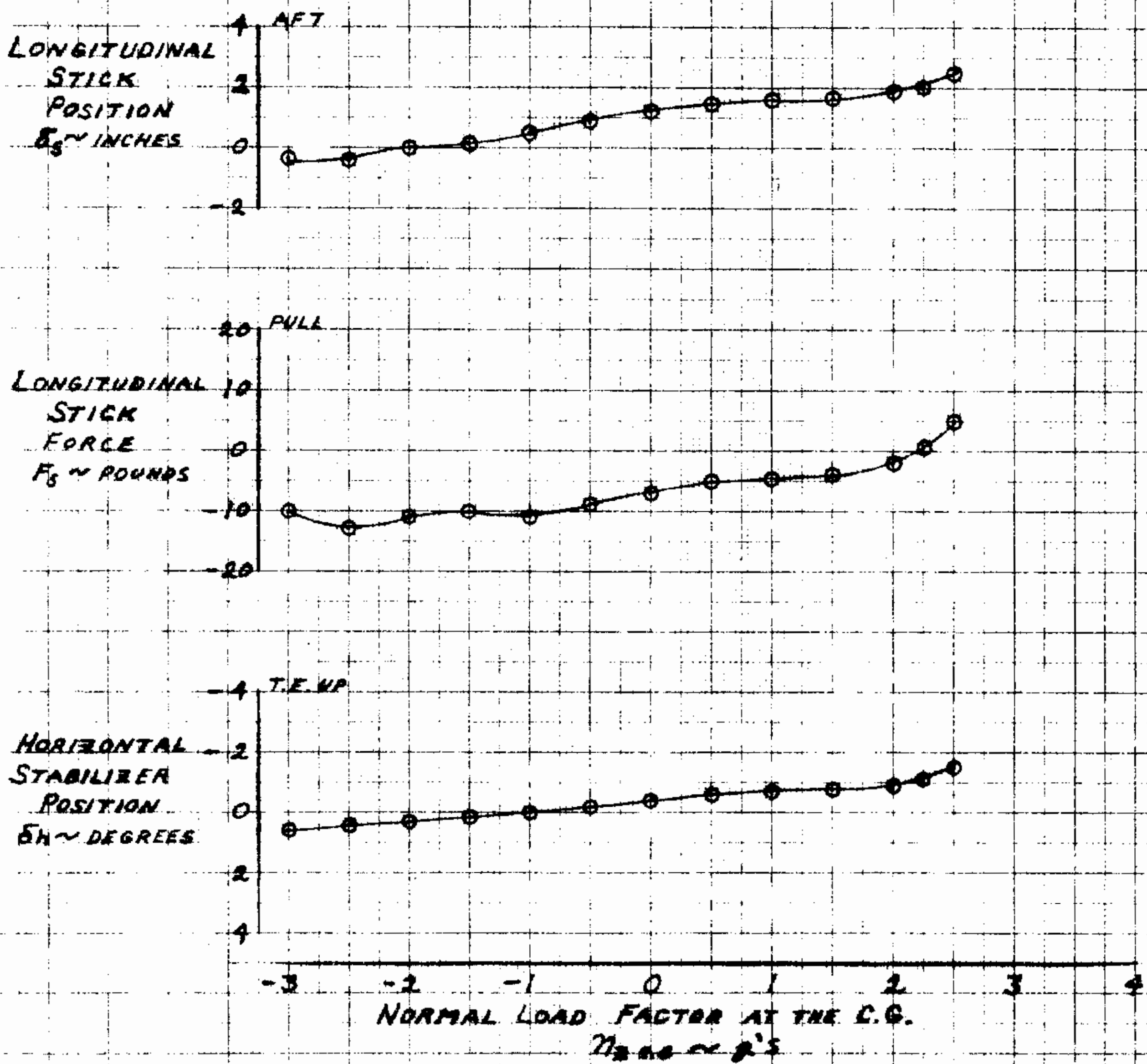
*A "British Pushover" is a pushover maneuver conducted to provide steady state longitudinal maneuvering data at a constant load factor less than 1 g while maintaining essentially constant Mach number, altitude and attitude.

F-5 FLIGHT TEST DATA

BRITISH PUSHOVER MANEUVER

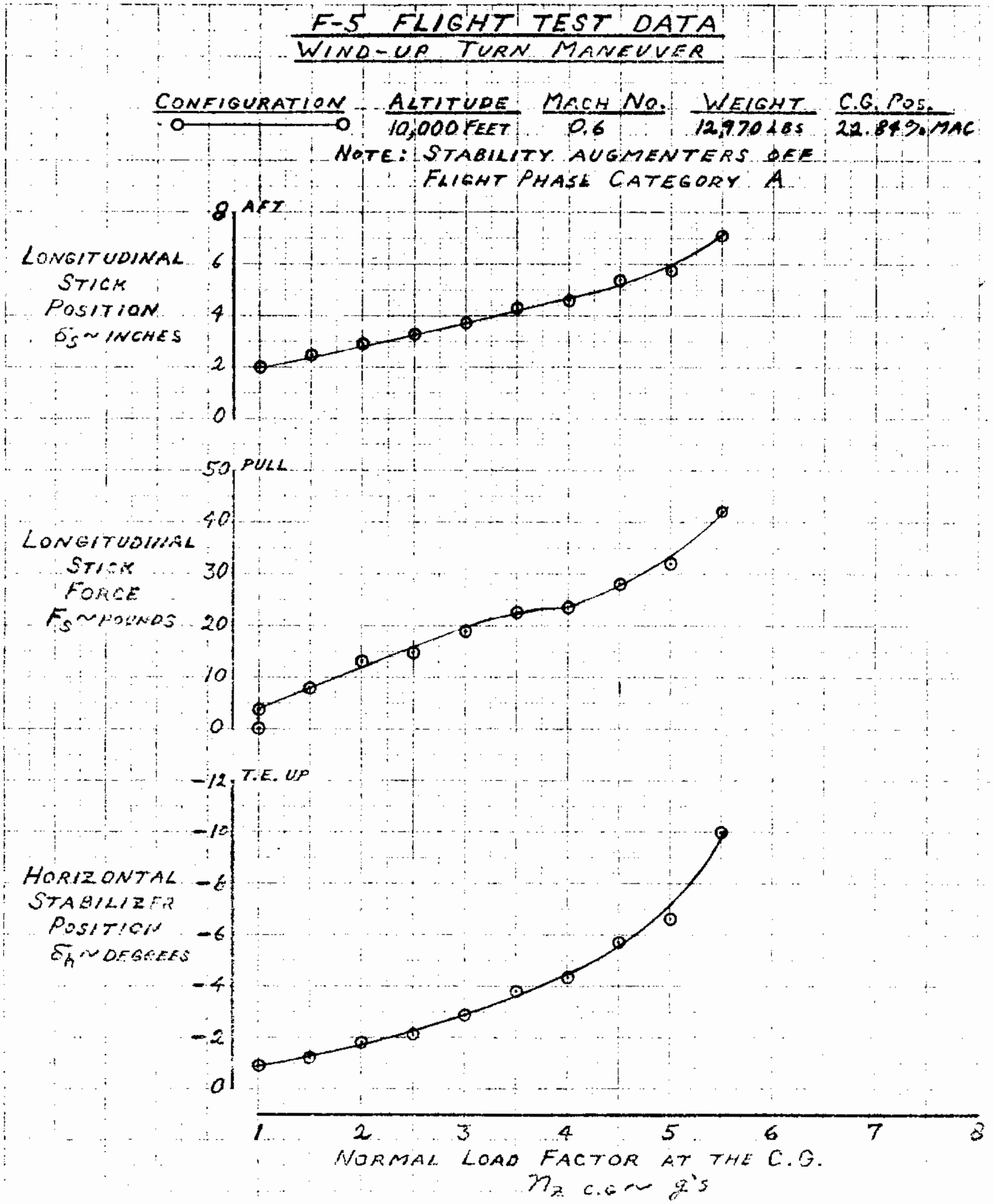
CONFIGURATION	ALTITUDE	MACH NO.	WEIGHT	C.G. POS.
O———O	10,000 FEET	0.8	13,410 LBS	22.48% MAC

NOTE 1
STABILITY AUGMENTERS OFF
FLIGHT PHASE CATEGORY A



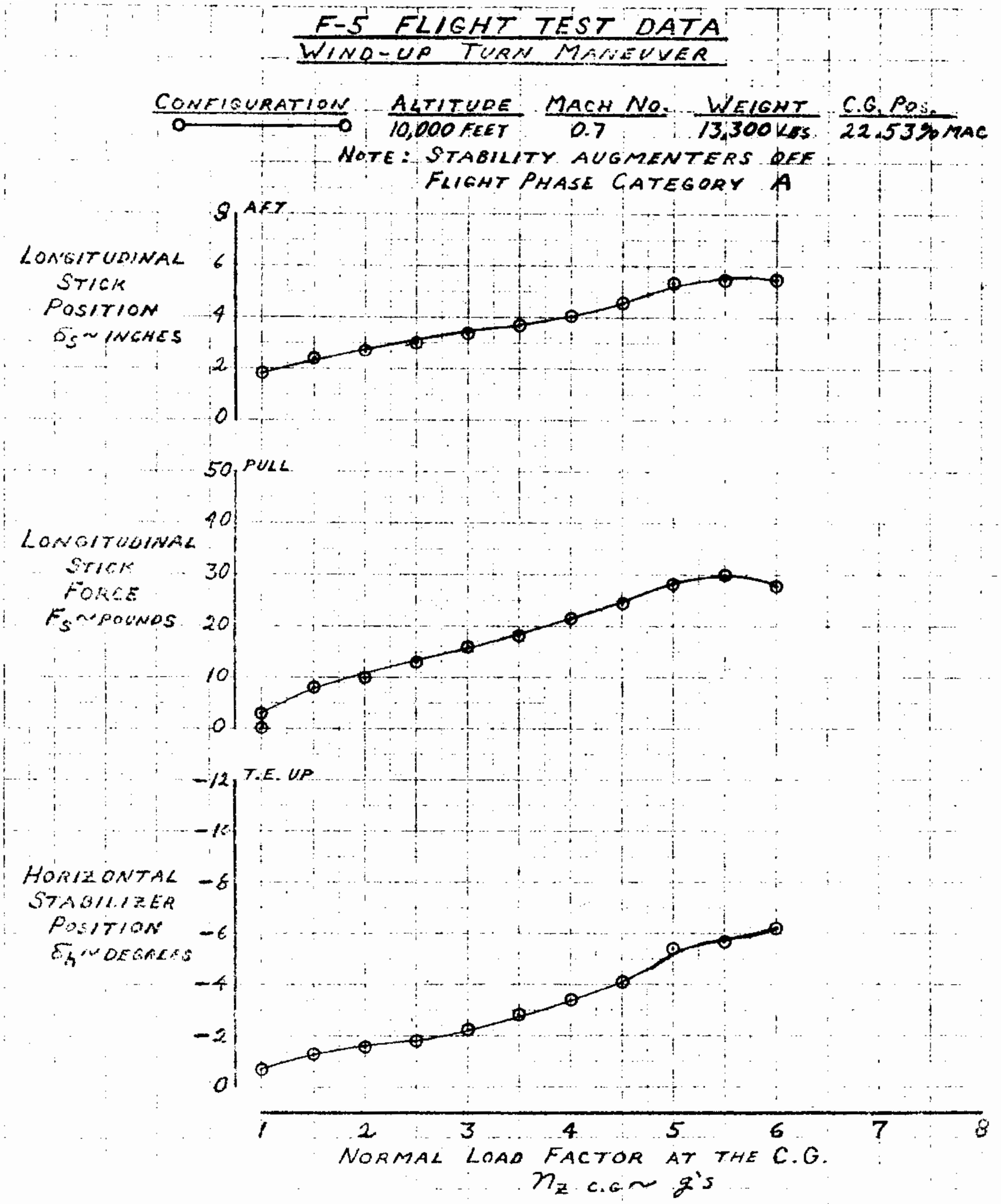
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 1 (3.2.2.2)



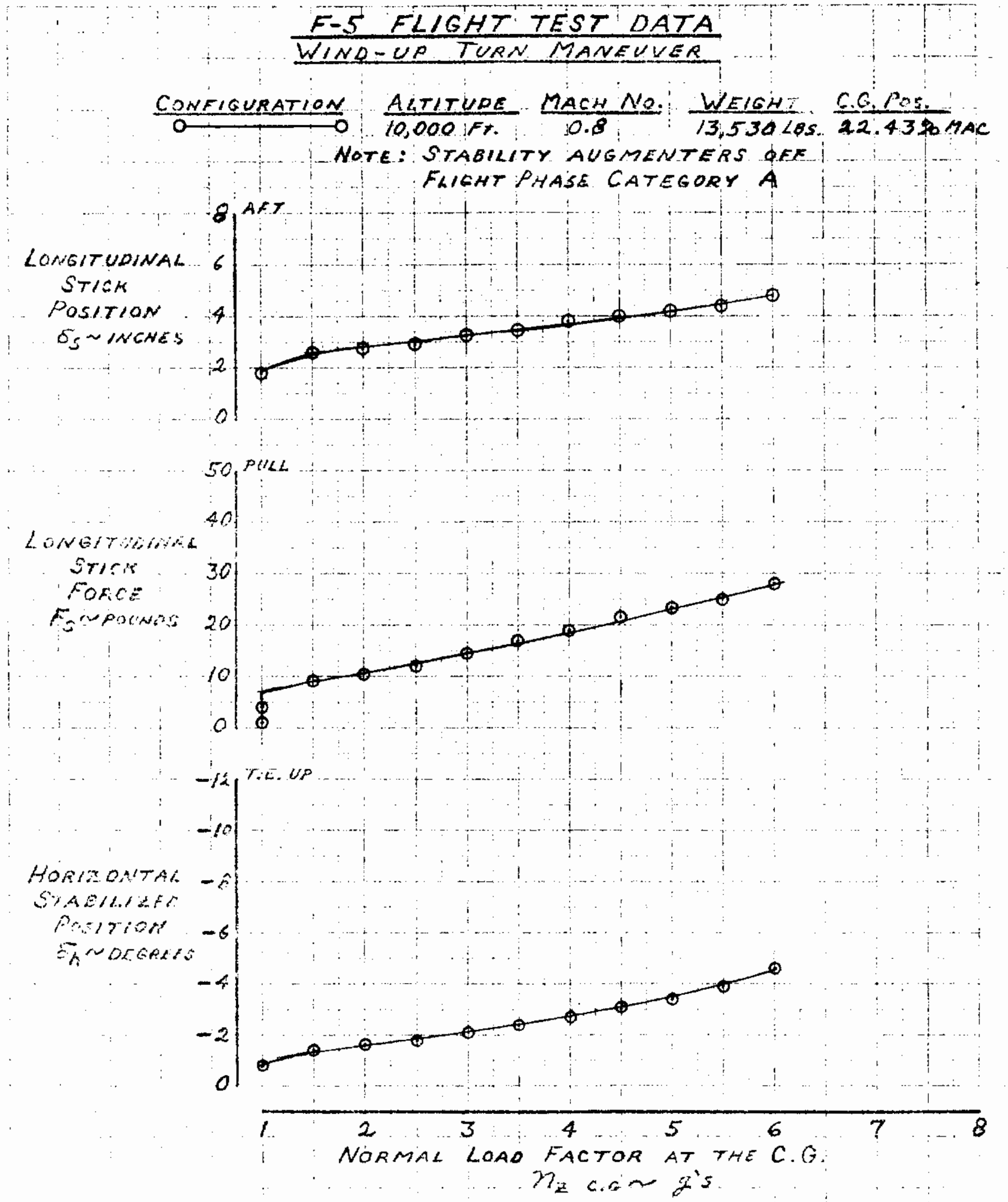
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 2 (3.2.2.2)



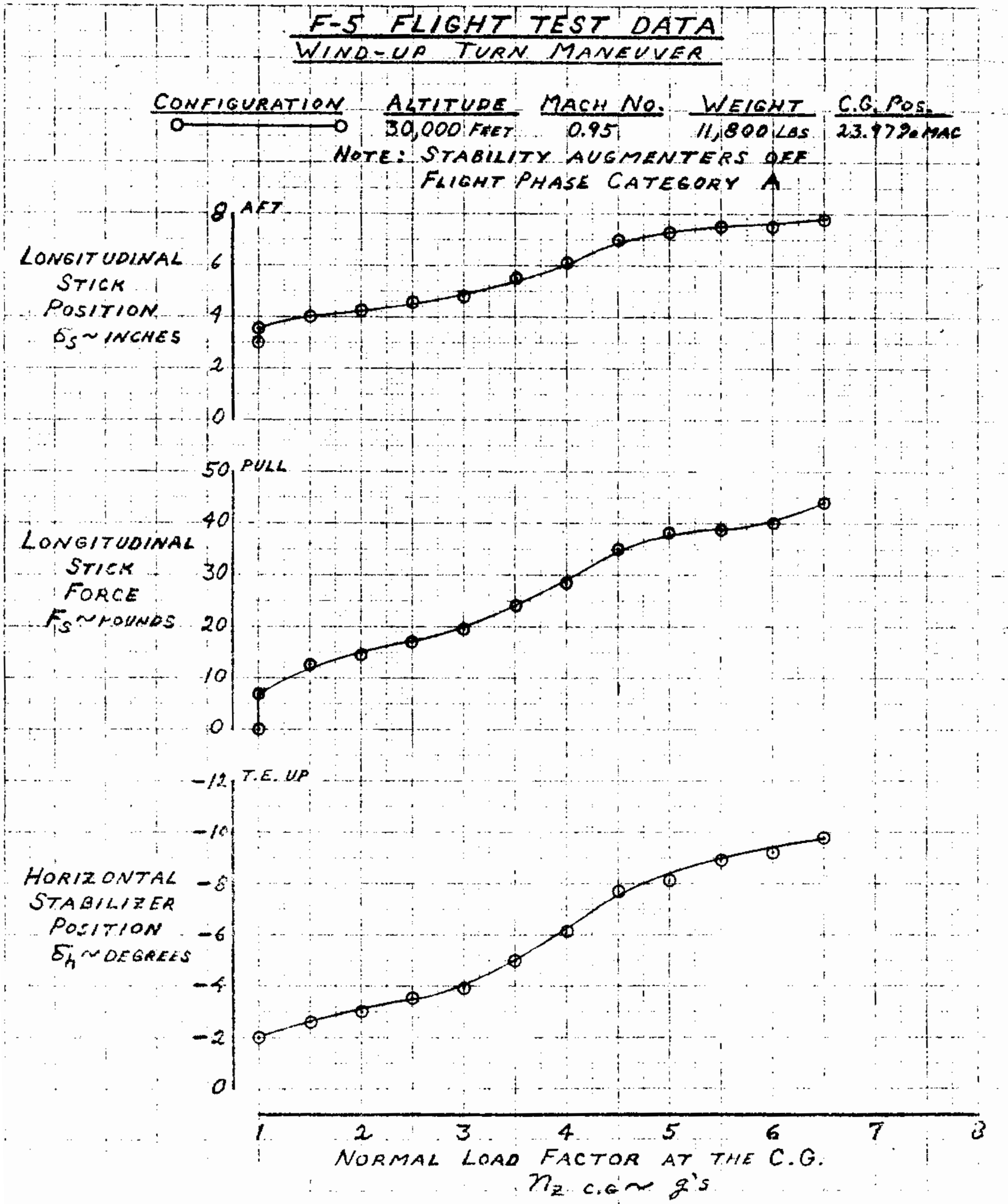
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 3 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 4 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

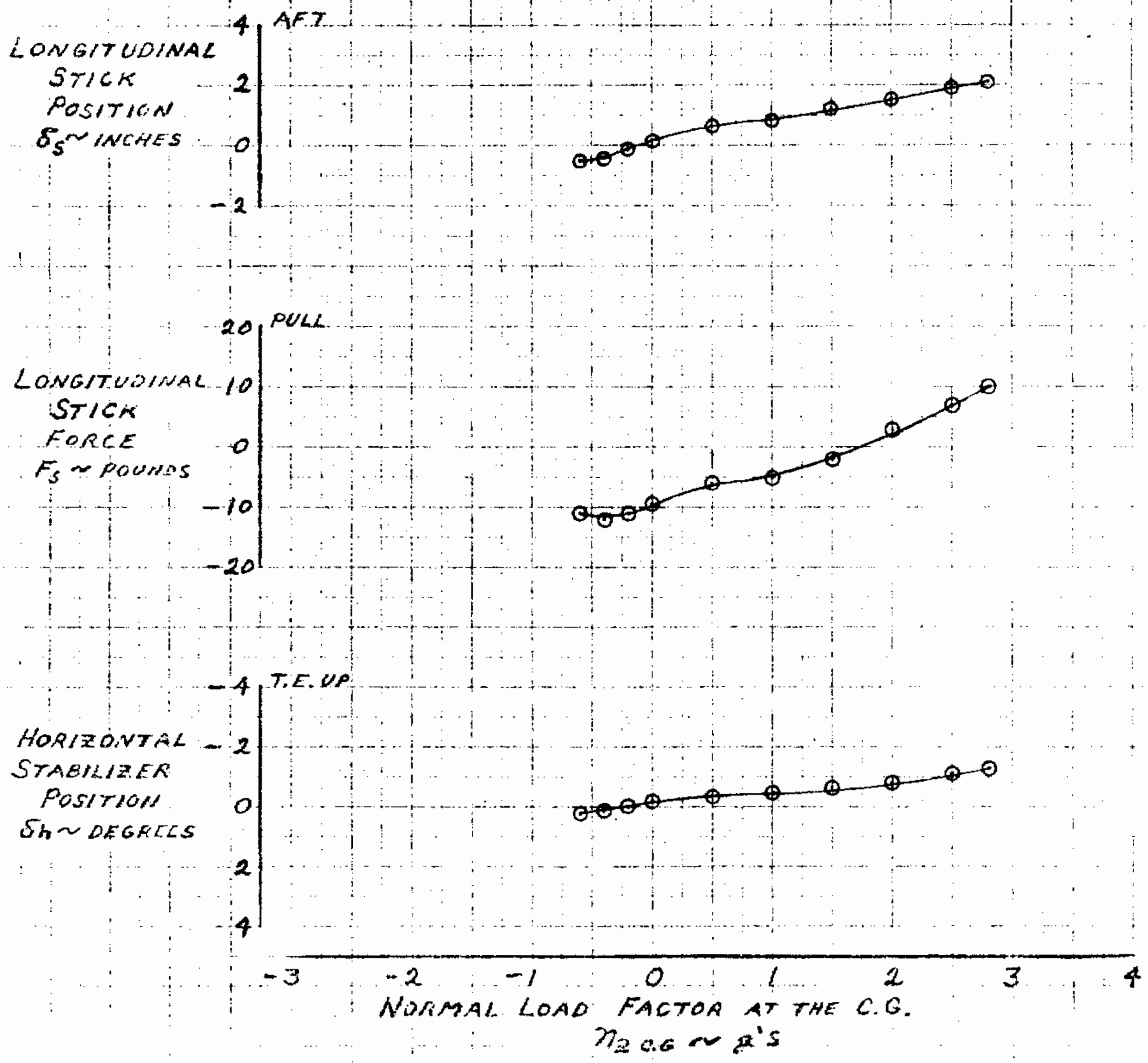
FIGURE 5 (3.2.2.2)

F-5 FLIGHT TEST DATA

BRITISH PUSHOVER MANEUVER

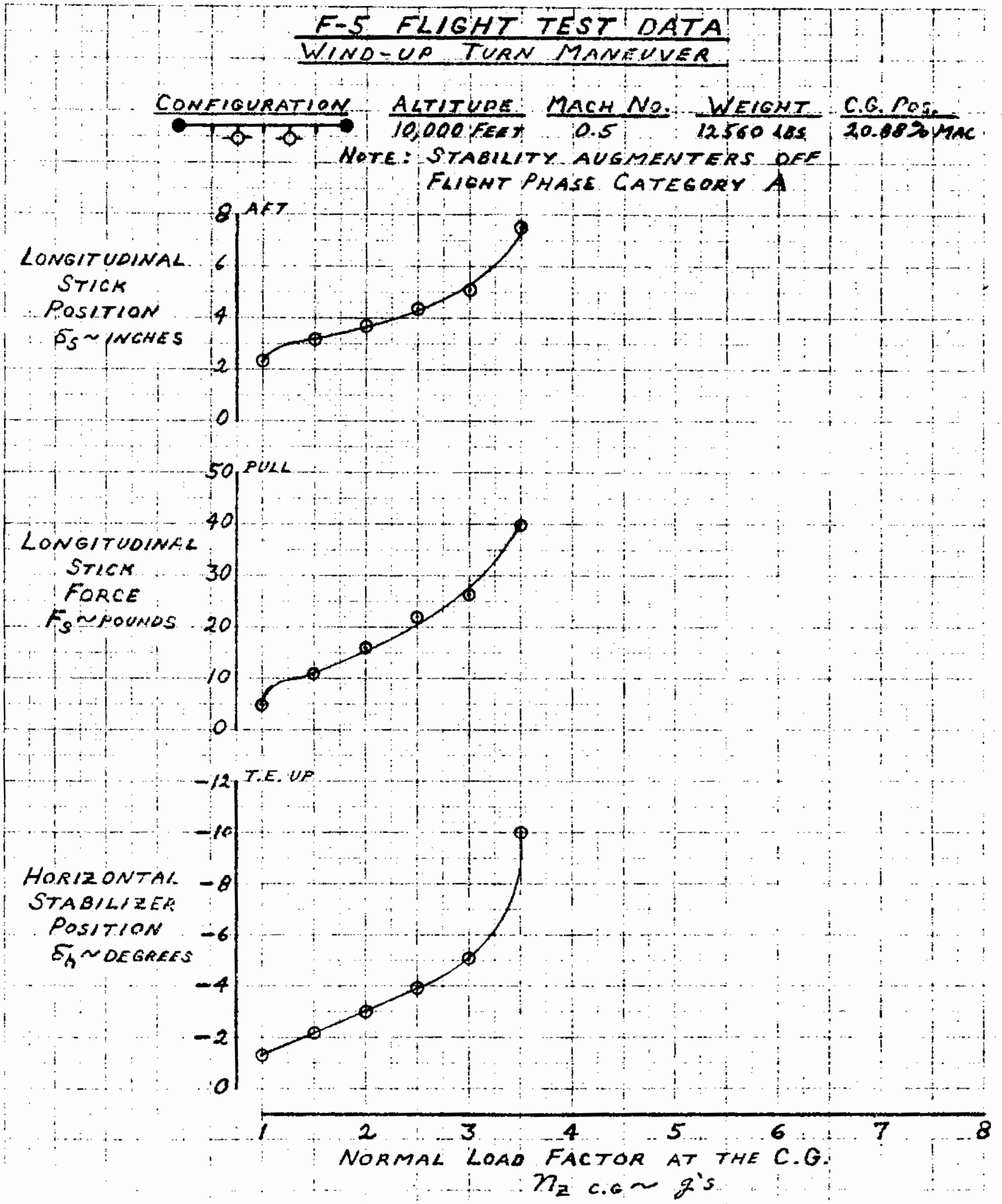
CONFIGURATION	ALTITUDE	MACH NO.	WEIGHT	C.G. POS.
	10,000 FEET	0.80	14,260 LBS	19.4% MAC

NOTE:
STABILITY AUGMENTERS OFF
FLIGHT PHASE CATEGORY A



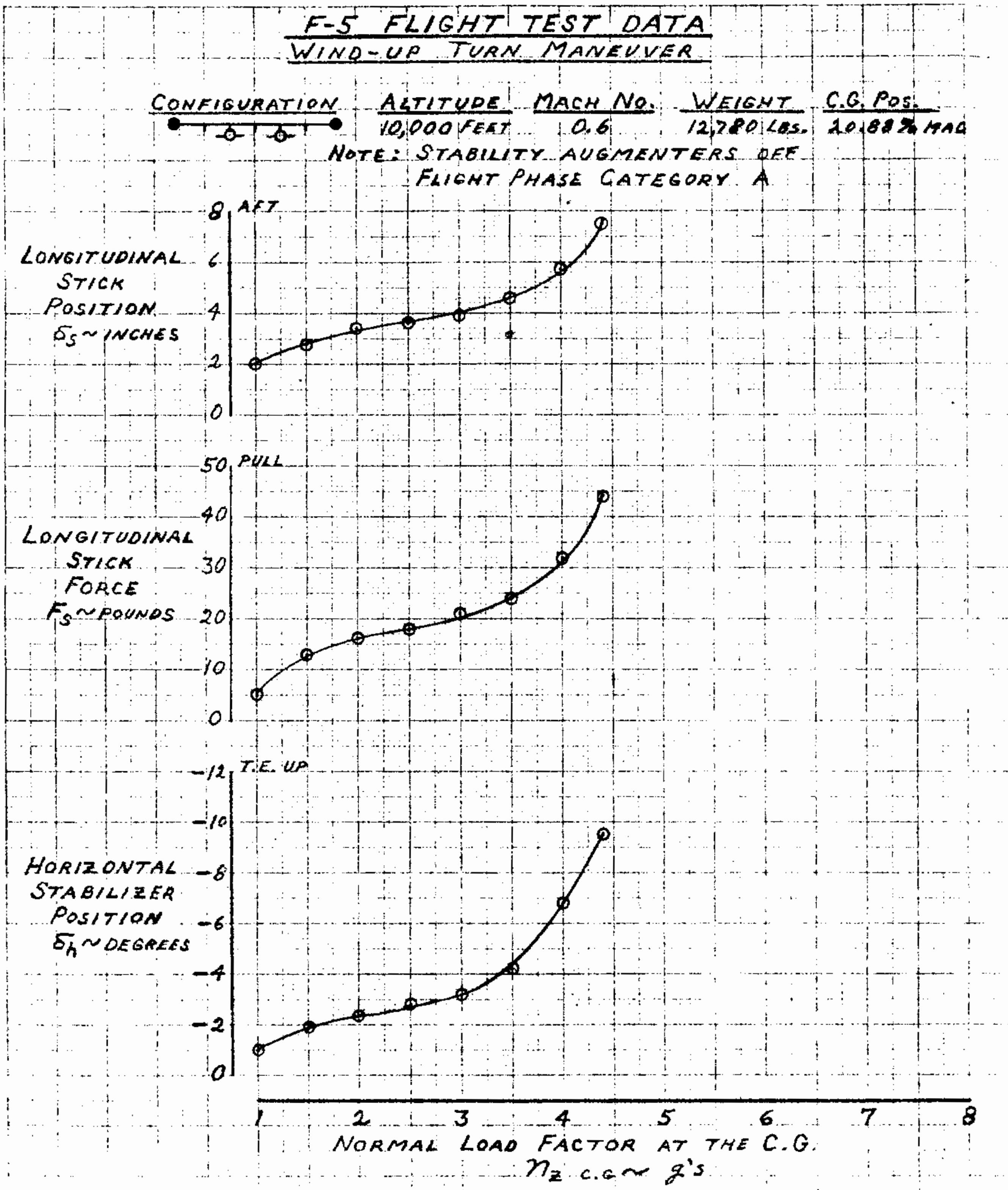
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 6 (3.2.2.2)



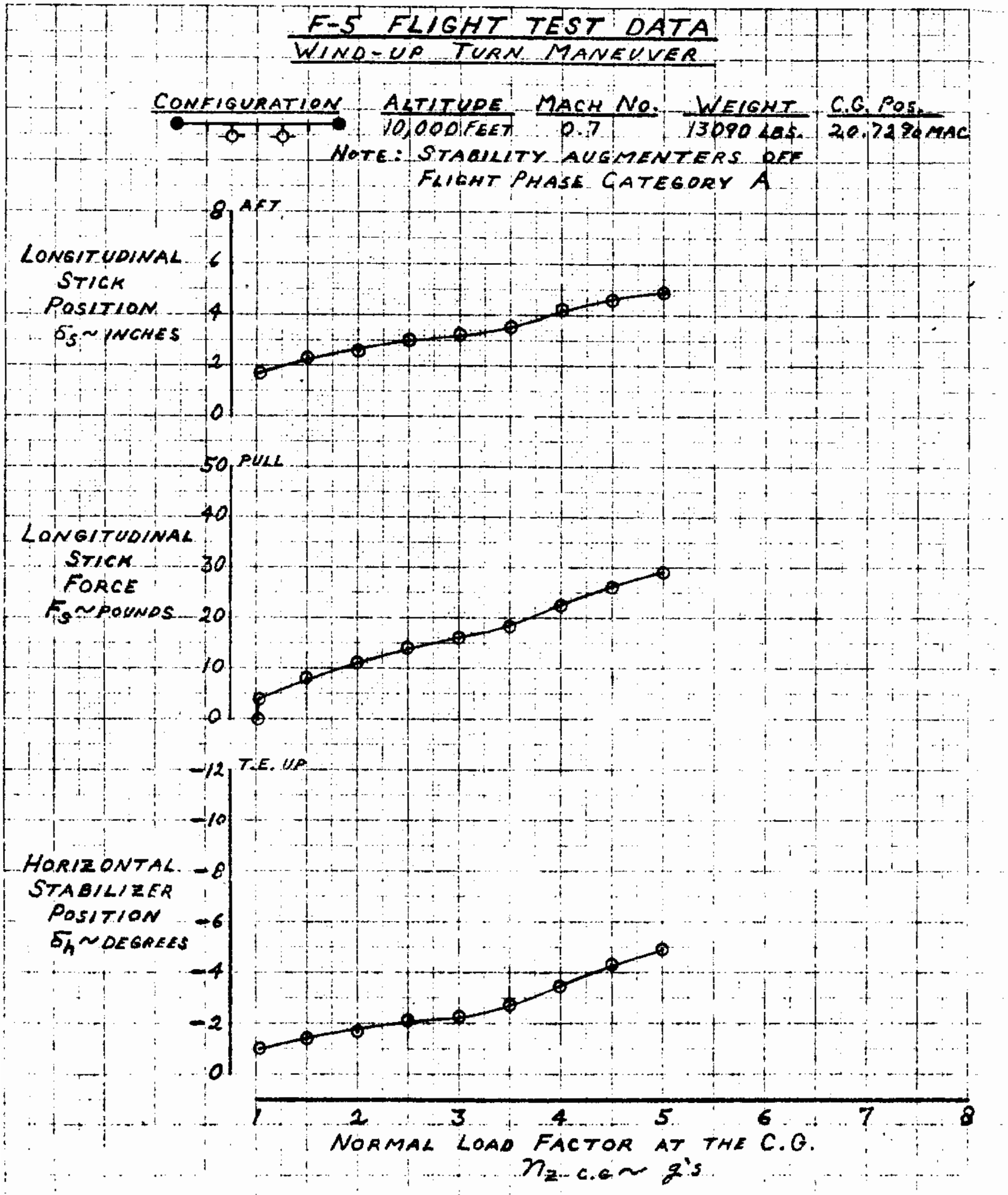
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 7 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 8 (3.2.2.2)



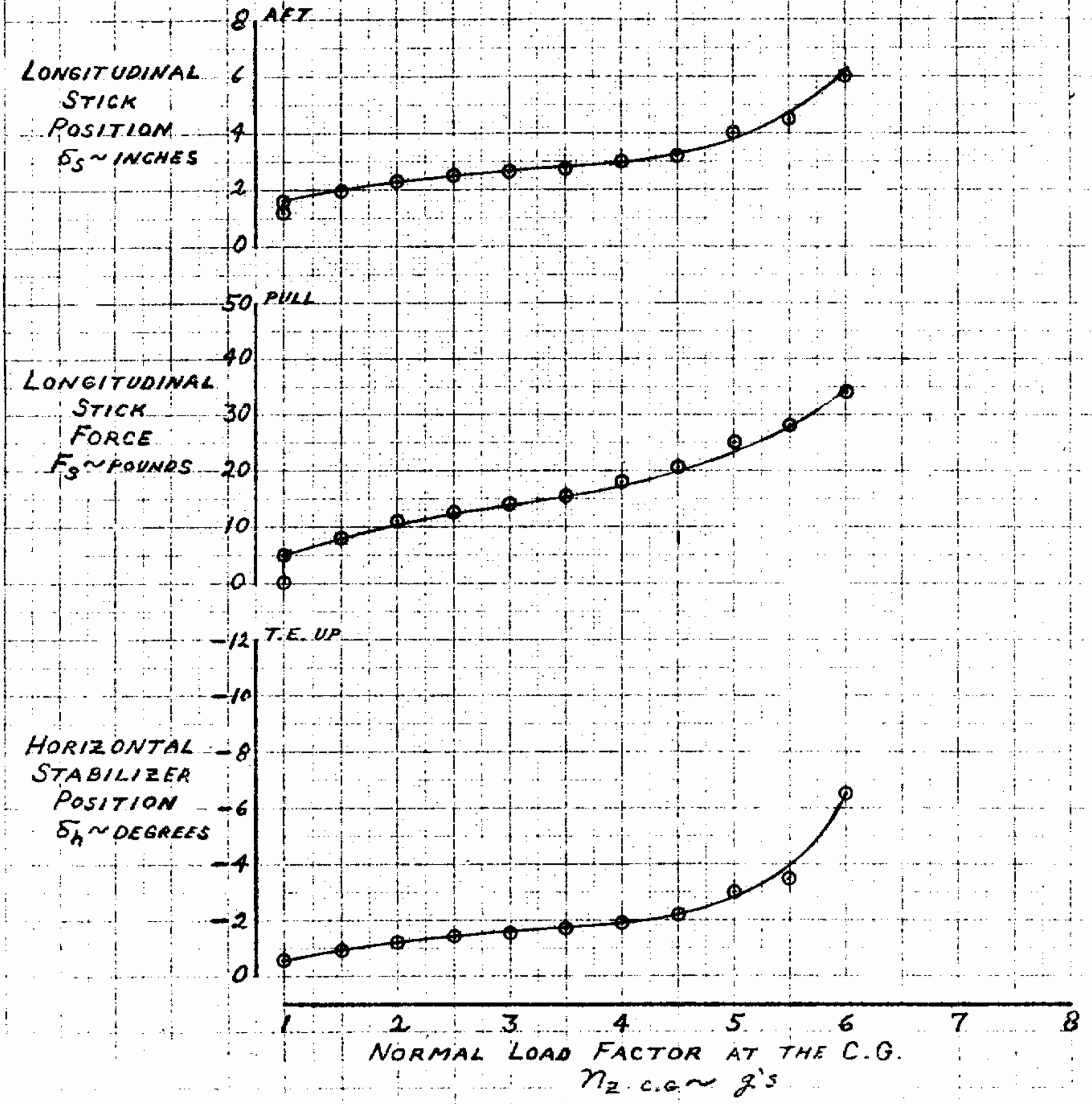
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 9 (3.2.2.2)

F-5 FLIGHT TEST DATA
WIND-UP TURN MANEUVER

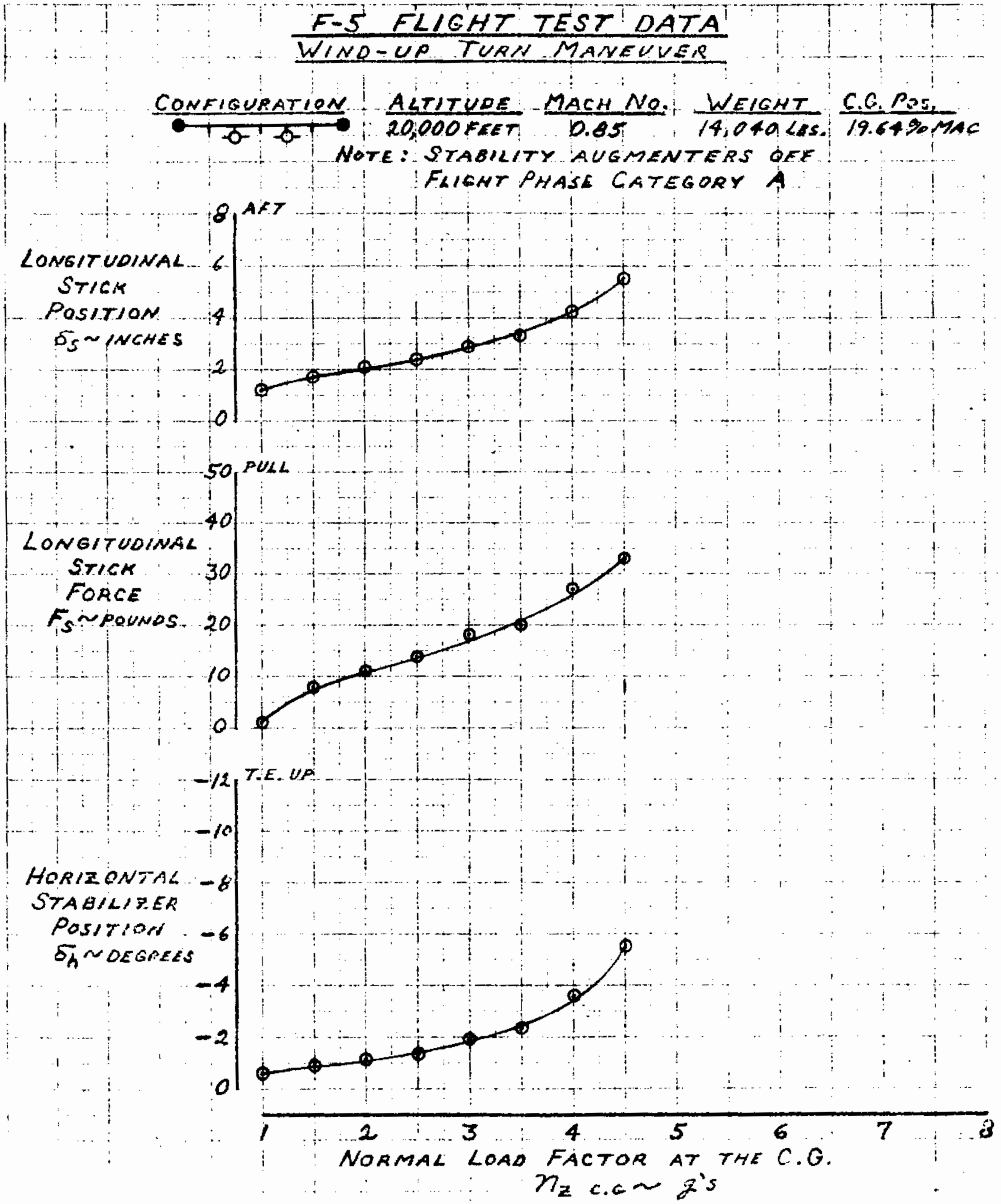
CONFIGURATION	ALTITUDE	MACH NO.	WEIGHT	C.G. POS.
	10,000 FEET	0.8	14130 LBS.	19.45% MAC

NOTE: STABILITY AUGMENTERS OFF
FLIGHT PHASE CATEGORY A



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 10 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

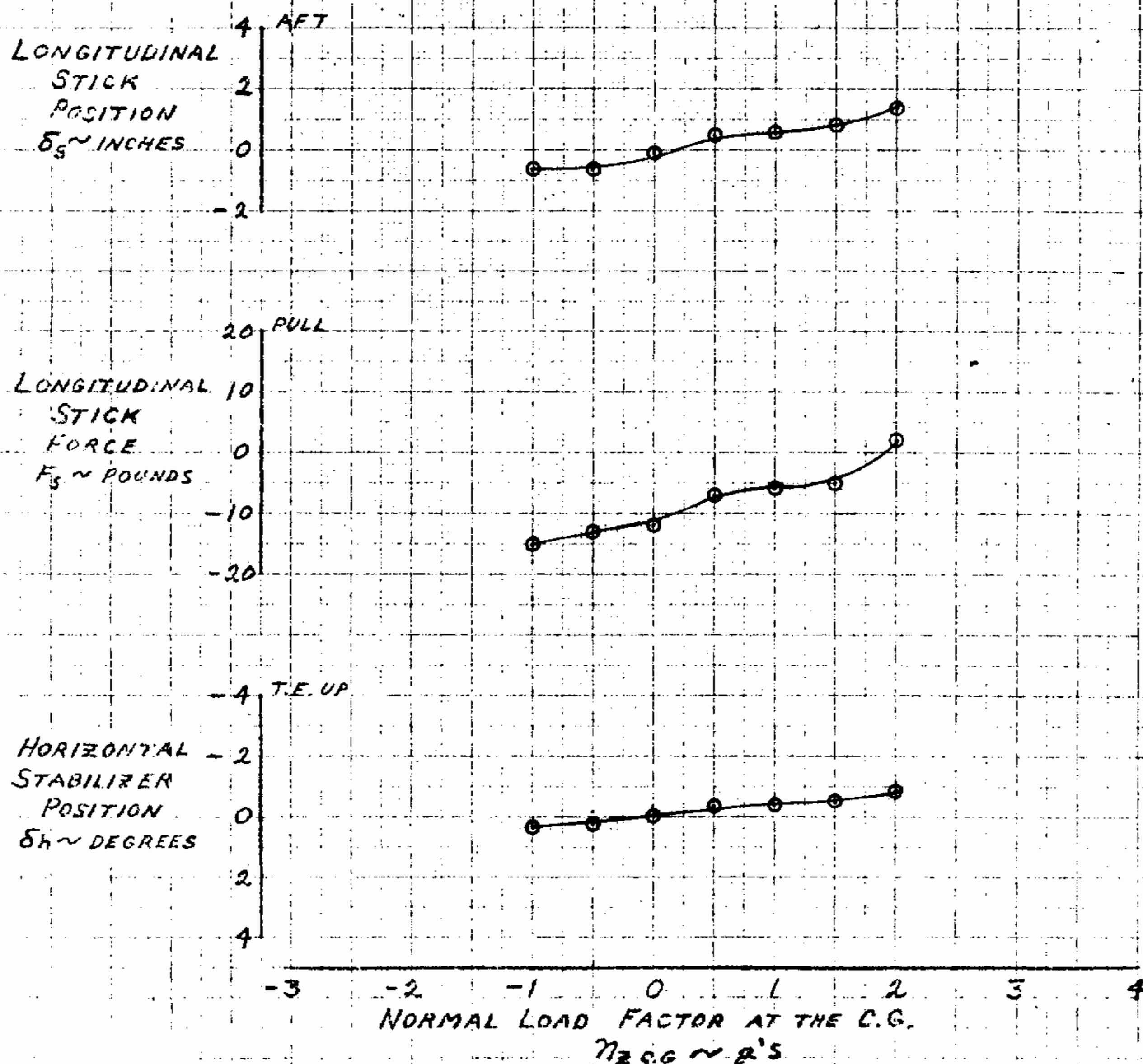
FIGURE 11 (3.2.2.2)

F-5 FLIGHT TEST DATA

BRITISH PUSHOVER MANEUVER

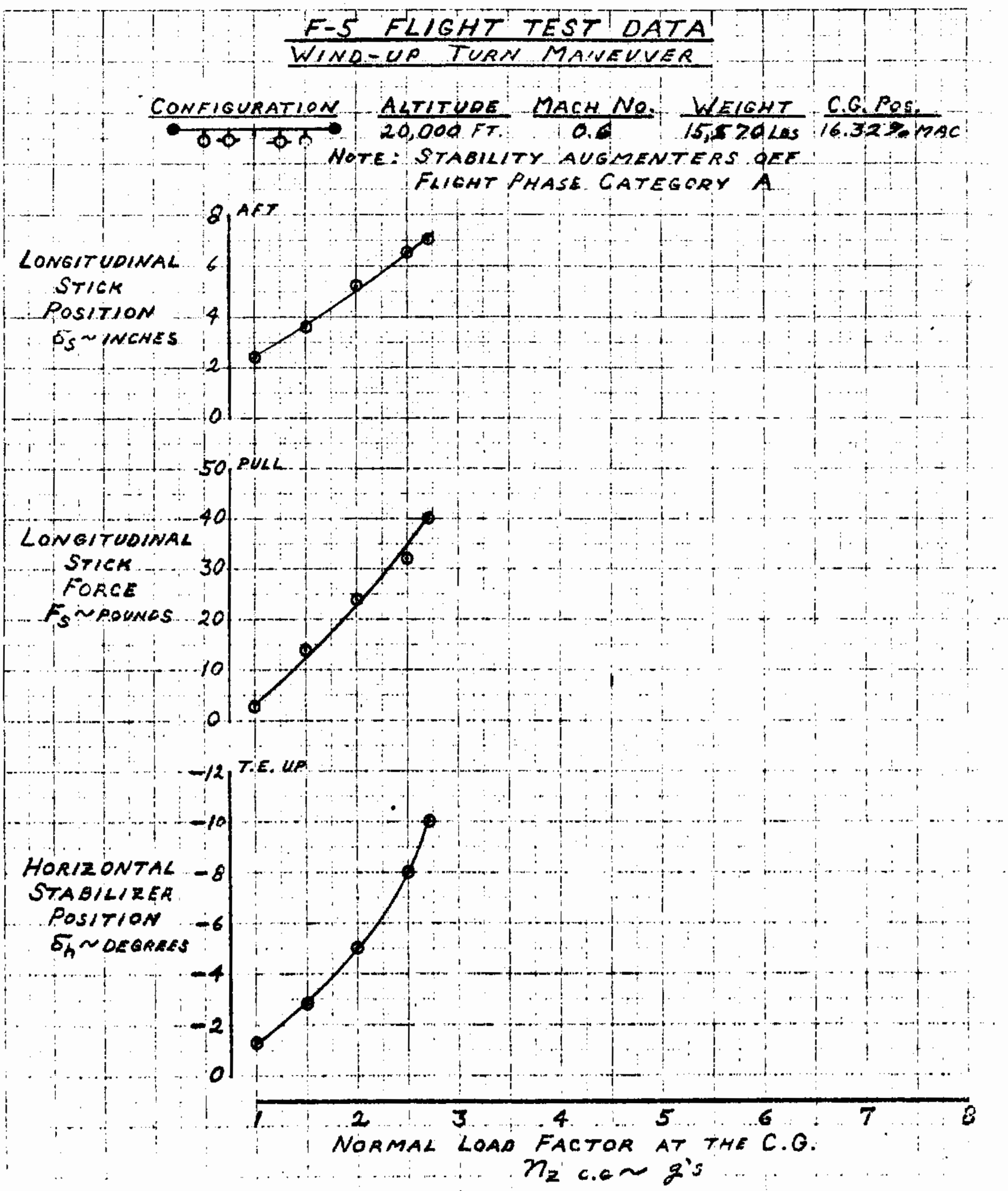
CONFIGURATION	ALTITUDE	MACH NO.	WEIGHT	C.G. POS.
	20,000 FEET	0.8	16,220 LBS.	16.45% MAC

NOTE 1
STABILITY AUGMENTERS OFF
FLIGHT PHASE CATEGORY A



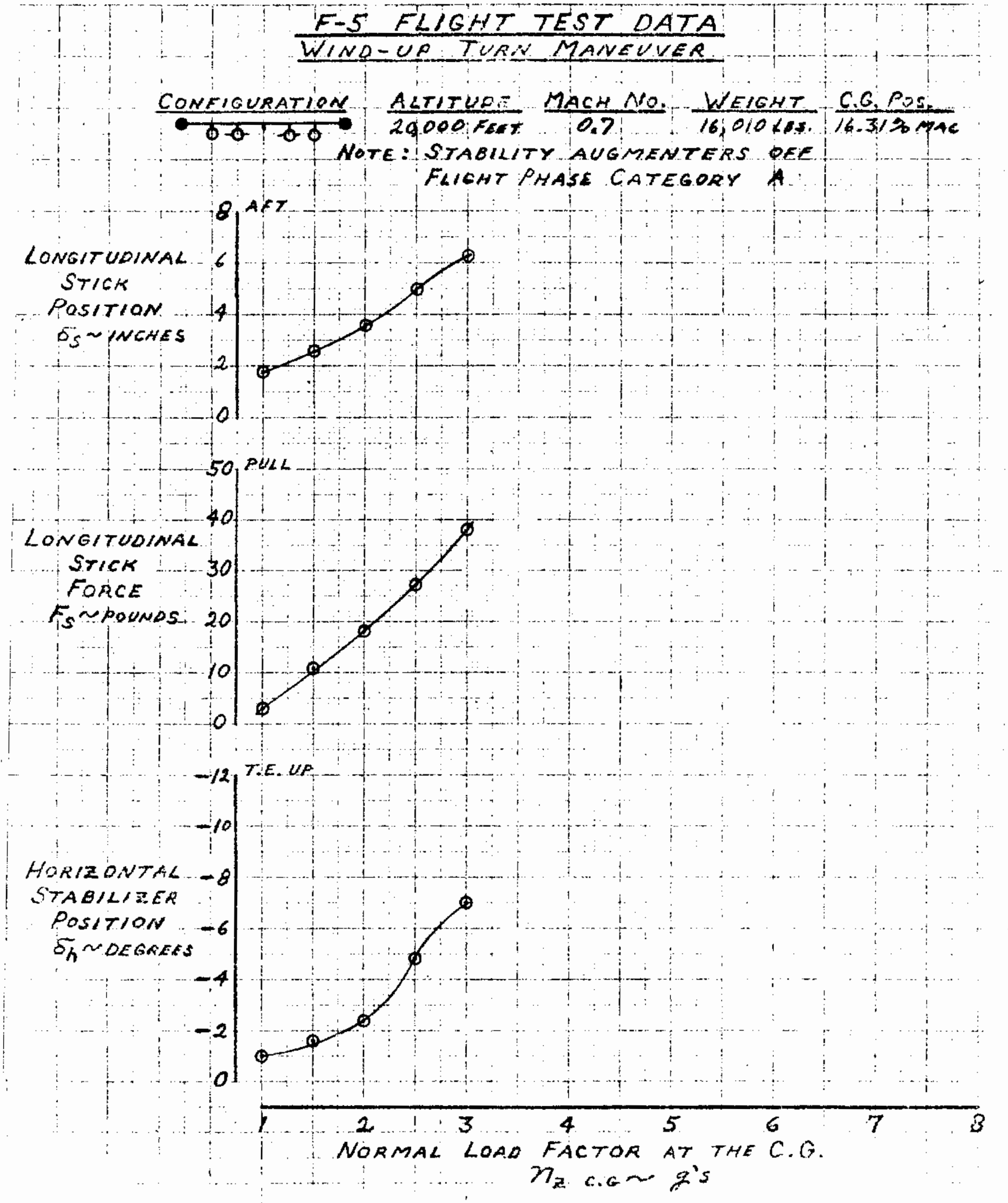
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 12 (3.2.2.2)



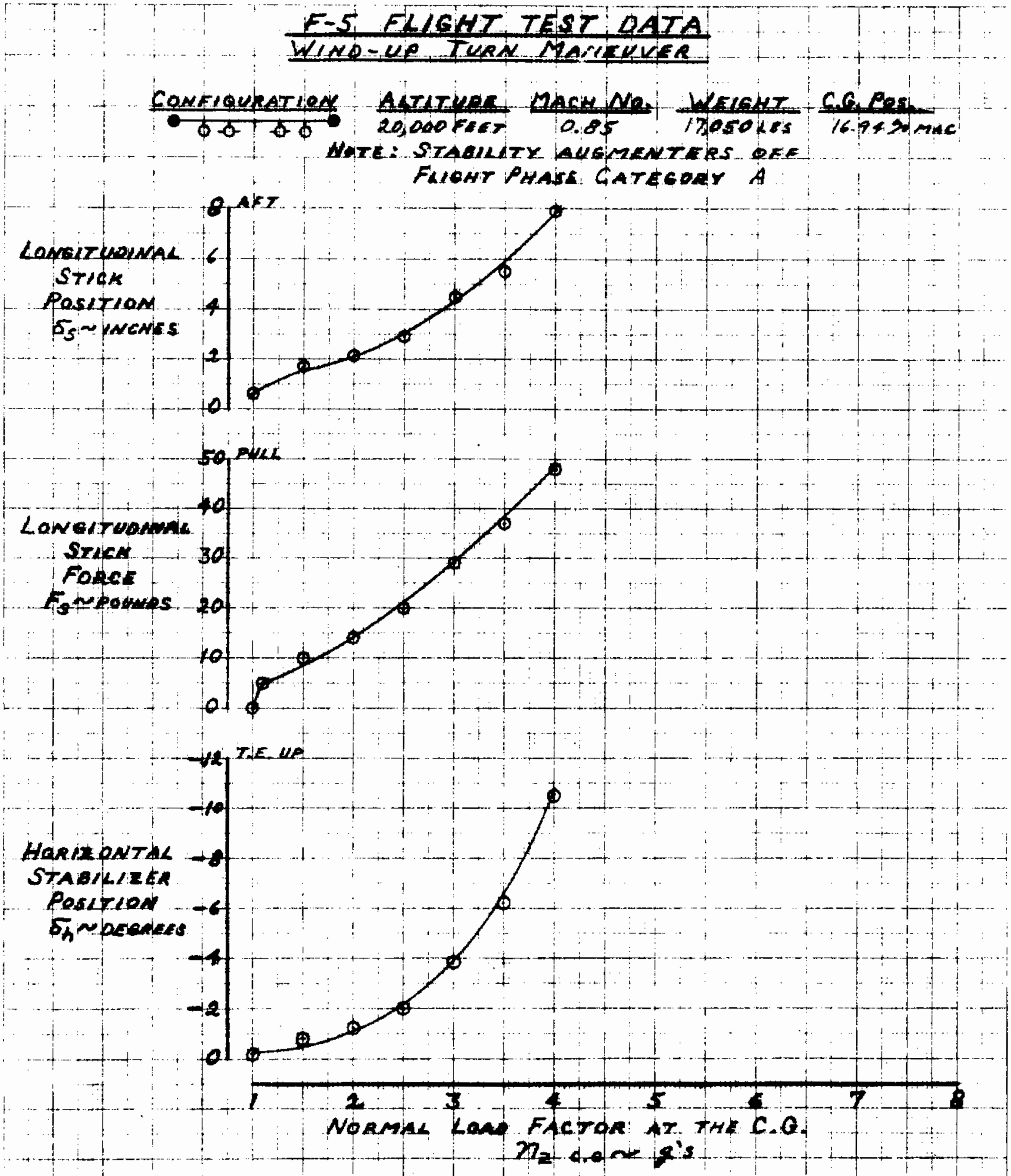
CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 13 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

FIGURE 14 (3.2.2.2)



CONTROL FEEL AND STABILITY IN MANEUVERING FLIGHT

Figure 15 (3.2.2.2)

Requirement

Paragraph 3.2.2.2.1 Control forces in maneuvering flight. At constant speed in steady turning flight, pullups, and pushovers, the variations in elevator-control force with steady-state normal acceleration shall be approximately linear. In general, a departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50 percent is considered excessive. All local force gradients shall be within the limits of table V. In addition, whenever the short-period frequency is near the upper boundaries of figure 1, F_s/n should be near the Level 1 upper boundaries of table V. This may be necessary to avoid abrupt response, sensitivity, or tendencies toward pilot-induced oscillations. The term gradient does not include that portion of the force versus n curve within the preloaded breakout force or friction band.

Comparison

Flight test wind-up turn F-5 data used for comparison with the requirements of this paragraph were selected from the data presented in Figures 1 (3.2.2.2) through 15 (3.2.2.2). These data, reduced in the form of F_s/n versus n/α where F_s/n is the average force gradient, are presented in Figures 1 (3.2.2.2.1) through 3 (3.2.2.2.1). The results exhibit favorable comparison with the requirements of this paragraph.

Resolution

Average gradient is not defined. In order to avoid inconsistency, with possible consequences of false compliance or non-compliance, a mathematical method of defining average should be made part of the specification.

Recommendation

It is recommended that the following definition of average gradient be made part of the specification.

$$(F_s/n)_{average} = \frac{F_{s(1)} - F_{s(2)}}{n_{(1)} - 1}$$

where,

$F_{s_{(1)}}$ = stick force at the lower n of:

(1) n at 85% of $C_{L_{max}}$

or

(2) n at 85% of n_L

$F_{s(2)}$ = break-out force

$n_{(1)}$ = normal load factor at the condition of $F_{s(1)}$.

TABLE V. Elevator Maneuvering Force Gradient Limits

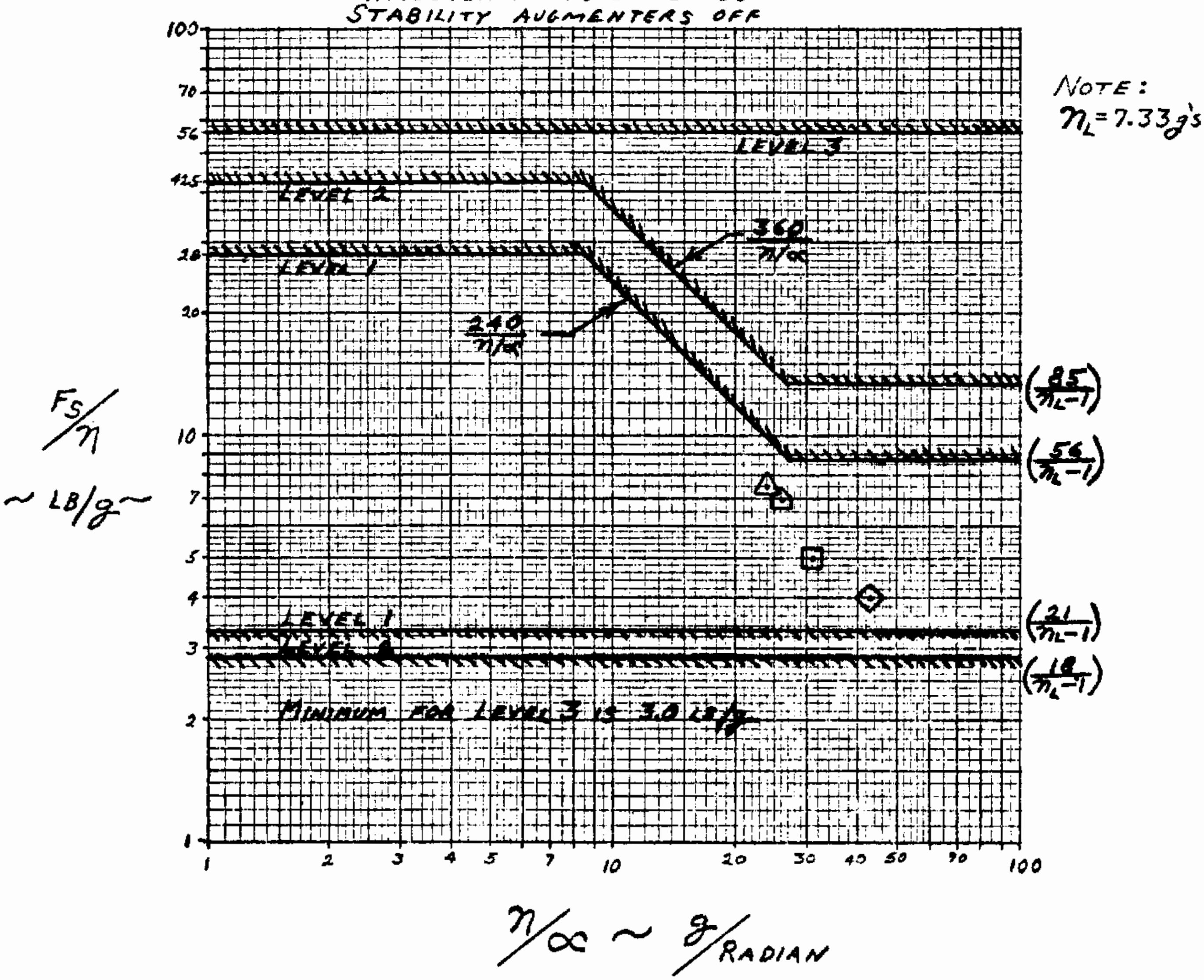
Center Stick Controllers		
Level	Maximum Gradient, $(F_s/n)_{max}$, pounds per g	Minimum Gradient, $(F_s/n)_{min}$, pounds per g
1	$\frac{240}{n/\alpha}$ but not more than 28.0 nor less than $\frac{56}{n_L-1}$ *	The higher of $\frac{21}{n_L-1}$ and 3.0
2	$\frac{360}{n/\alpha}$ but not more than 42.5 nor less than $\frac{85}{n_L-1}$ *	The higher of $\frac{18}{n_L-1}$ and 3.0
3	56.0	3.0
*For $n_L < 3$, $(F_s/n)_{max}$ is 28.0 for Level 1, 42.5 for Level 2.		

Wheel Controllers		
Level	Maximum Gradient, $(F_s/n)_{max}$, pounds per g	Minimum Gradient, $(F_s/n)_{min}$, pounds per g
1	$\frac{500}{n/\alpha}$ but not more than 120.0 nor less than $\frac{120}{n_L-1}$	The higher of $\frac{45}{n_L-1}$ and 6.0
2	$\frac{775}{n/\alpha}$ but not more than 182.0 nor less than $\frac{182}{n_L-1}$	The higher of $\frac{38}{n_L-1}$ and 6.0
3	240.0	6.0

F-5
FLIGHT TEST DATA

CONFIGURATION	ALTITUDE	MN	MANEUVER	SYMBOL
o———o	10,000 FEET	0.6	WIND-UP TURN	△
"	" "	0.7	" " "	□
"	" "	0.8	" " "	◇
"	30,000 FEET	0.95	" " "	⌂

WEIGHT RANGE—12,000 TO 13,500 LBS. C.G. RANGE — 22.4% TO 22.8% MAC
MANEUVER FLAPS DEFLECTED
STABILITY AUGMENTERS OFF



CONTROL FORCES IN MANEUVERING FLIGHT

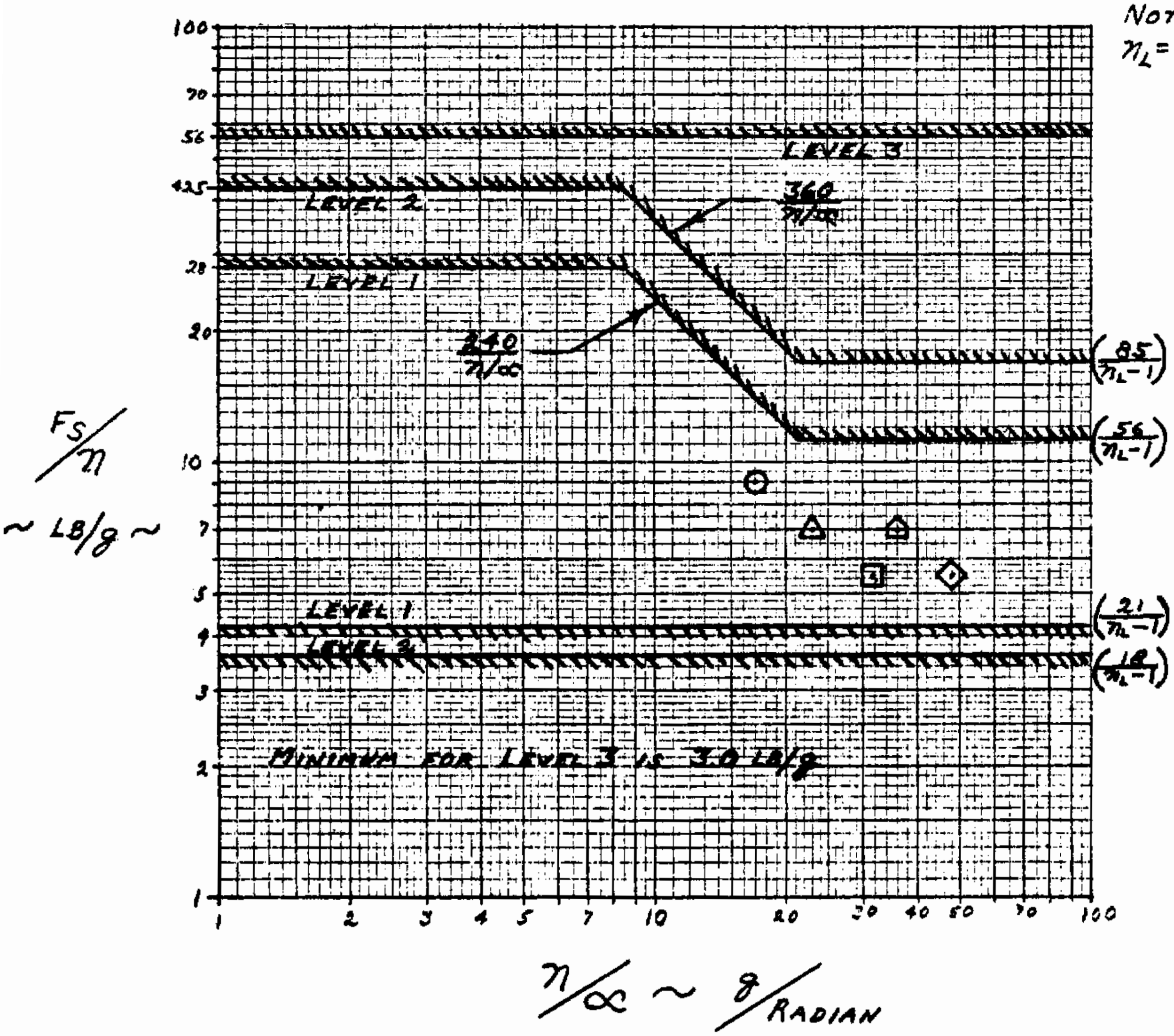
FIGURE 1 (3.2.2.2.1)

F-5
FLIGHT TEST DATA

CONFIGURATION	ALTITUDE	MN	MANEUVER	SYMBOL
	10,000 FEET	0.5	WIND-UP TURN	○
"	" "	0.6	" " "	△
"	" "	0.7	" " "	□
"	" "	0.8	" " "	◇
"	" "	0.85	" " "	⌂

WEIGHT RANGE — 12,500 TO 14,100 LBS. C.G. RANGE — 19.6 % TO 20.9 % MAC
STABILITY AUGMENTERS OFF

NOTE : $n_L = 6.0$ g's



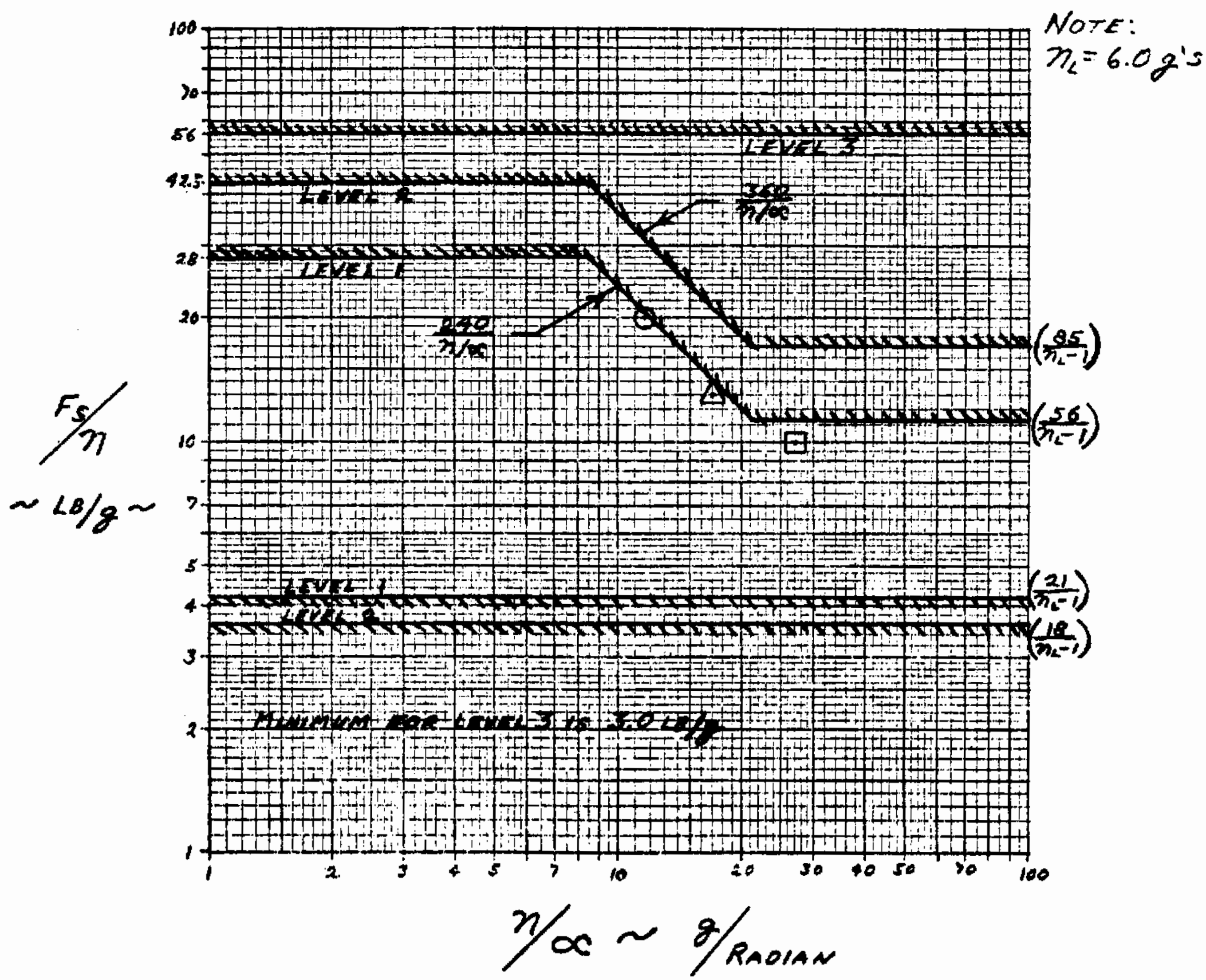
CONTROL FORCES IN MANEUVERING FLIGHT

FIGURE 2 (3.2.2.2.1)

F-5
FLIGHT TEST DATA

CONFIGURATION	ALTITUDE	MN	MANEUVER	SYMBOL
	20,000 FEET	0.6	WIND-UP TURN	○
"	" "	0.7	" " "	△
"	" "	0.85	" " "	□

WEIGHT RANGE – 15,570 TO 17,000 LBS. C.G. RANGE – 16.3% TO 16.9% MAC
STABILITY AUGMENTERS OFF



CONTROL FORCES IN MANEUVERING FLIGHT

FIGURE 3 (3.2.2.2.1)

Requirement

Paragraph 3.2.2.2.2 Control motions in maneuvering flight. The elevator-control motions in maneuvering flight shall not be so large or so small as to be objectionable. For Category A Flight Phases, the average gradient of elevator-control force per inch of elevator-control deflection at constant speed shall be not less than 5 pounds per inch for Levels 1 and 2.

Comparison

The F-5 longitudinal basic control system is a tri-gradient design of elevator-control force to elevator-control deflection, pounds per inch. These gradients are 5.363 pounds per inch, 6.680 pounds per inch, and 14.124 pounds per inch. Agreement with the minimum requirement of 5 pounds per inch is obtained.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.2.3 Longitudinal pilot-induced oscillations. There shall be no tendency for pilot-induced oscillations, that is, sustained or uncontrollable oscillations resulting from the efforts of the pilot to control the airplane.

Comparison

The following is a descriptive background of the work conducted on the T-38 and F-5 aircraft to check and correct any tendency for pilot-induced oscillations (PIO's).

The capability of predicting pilot-closed-loop dynamic stability such that satisfactory total system stability is assured prior to flight is of paramount importance for both safety and cost effectiveness (keeping design changes to minimum). Prediction methods and practical solutions applied to the T-38 and F-5 aircraft have been developed to such a degree that on the T-38, the longitudinal pitch damper could be removed.

T-38 Analysis. A P.I.O. occurred in an early version of the T-38 (since modified to completely eliminate the possibility from all T-38/F-5 aircraft) following the shutoff of a failed pitch damper in a mistrimmed position. This occurrence initiated a large analytic and flight test program to cure the problem.

A computer analysis determined the vehicle aerodynamic relationship that led to the P.I.O. encountered. Pilot-in-the-loop system analysis techniques of evaluating P.I.O. tendencies were developed. Use of a flight simulator verified fixes prior to flight. Criteria were established to evaluate future Northrop T-38/F-5 aircraft versions with respect to P.I.O.

Flight tests were accomplished incorporating the recommended hardware changes and the aircraft satisfactorily met all program objectives.

F-5 Analysis. It was established during the T-38 studies that important factors in determining P.I.O. sensitivity are transient stick force per g, bobweight effects, control system dynamics, spring force gradients and stick-to-surface deflection gearing curves.

The F-5A, having both (CO) and (GA) Flight Phase capabilities, can be flown in many configurations. All configurations that exhibit low stability (low F_s/n_z and short-period damping) or large trim changes due to c.g. movement caused by fuel consumption or store jettison (stick-to-surface gearing changes) have been analytically investigated. Two criteria have been determined.

The first criterion was established from T-38 and early F-5A flight tests. It relates transient stick forces to phasing of load factor by utilizing the gain margin concept. The loop for which the relative stability is evaluated consists of the pilot, controls, airframe and augmentation system. The stick-free mode is the inner loop of the multiloop system, Figure 1 (3.2.2.3). The complete controls-airframe-augmentation must be evaluated to define sufficiently the relative system dynamics with which the pilot will be required to cope.

This analysis is most readily accomplished by either the frequency response or root locus techniques. However, since the sinusoid is readily adaptable to flight test techniques, the frequency response technique is used. This allows the results to be presented directly in the log-magnitude/phase angle relationship. It has been substantiated that the stick free mode (n_z/F_S) gain margin greater than 15 db will provide a sufficient P.I.O. free margin, Figure 2 (3.2.2.3).

The second criterion was obtained by modifying an existing criterion to include the aircraft control system-short period mode interaction effects. It uses the ratio of the stick free damping and frequency to the dimensional derivative of normal force due to alpha to establish boundaries as a function of stick force per g, Figure 3 (3.2.2.3).

The boundaries were established by researching data from numerous aircraft P.I.O. studies noting P.I.O., possible P.I.O., and P.I.O.-free values. References 3 through 6 may be consulted for additional information relative to P.I.O. and the above discussion.

F-5 Flight Test. The importance of having P.I.O.-free flight makes it imperative for each F-5 configuration, which is flight tested to determine aft c.g. limit, also to be P.I.O. checked by performing sinusoidal stick motions about trim at given frequencies. The pilot follows an oscillating tone to fix the frequency, and both qualitative and quantitative data are used to analyze the results. Figure 4 (3.2.2.3) presents flight test results indicating accord between the first criterion and flight test results, and exhibiting no tendency for pilot-induced oscillations.

Resolution

A qualitative requirement to ensure no tendency for pilot-induced oscillations is not considered sufficient as a specification.

Recommendation

In the comparison part, it was shown how criteria using quantitative values can be specified as minimum requirement. It is recommended that research be conducted to establish a requirement with quantitative values and be so specified.

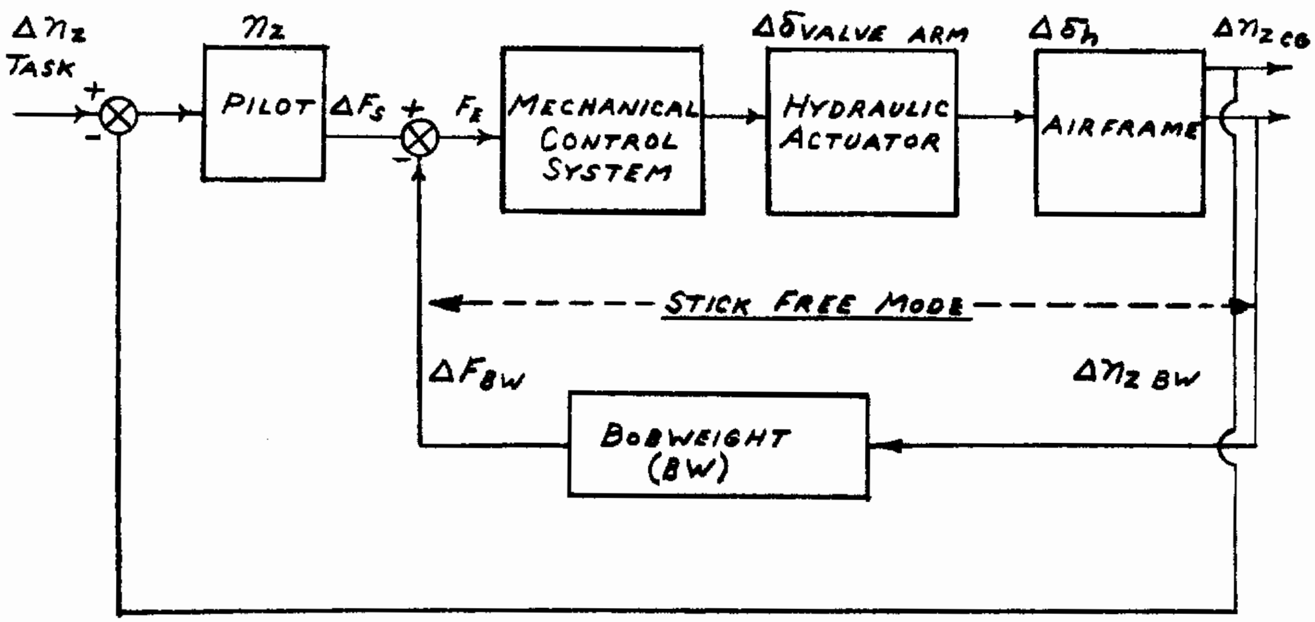
The following enumerates the tasks that should be conducted in the process of establishing a quantitative requirement for this paragraph :

1. Evaluate the criteria discussed in the comparison through application to other airplanes.

2. Compare the results with pilot comments and establish the P.I.O.-prone and non-prone regions.

3. Establish an optimum quantitative requirement based on all airplanes.

F-5
PILOT TRACKING MODE LOOP

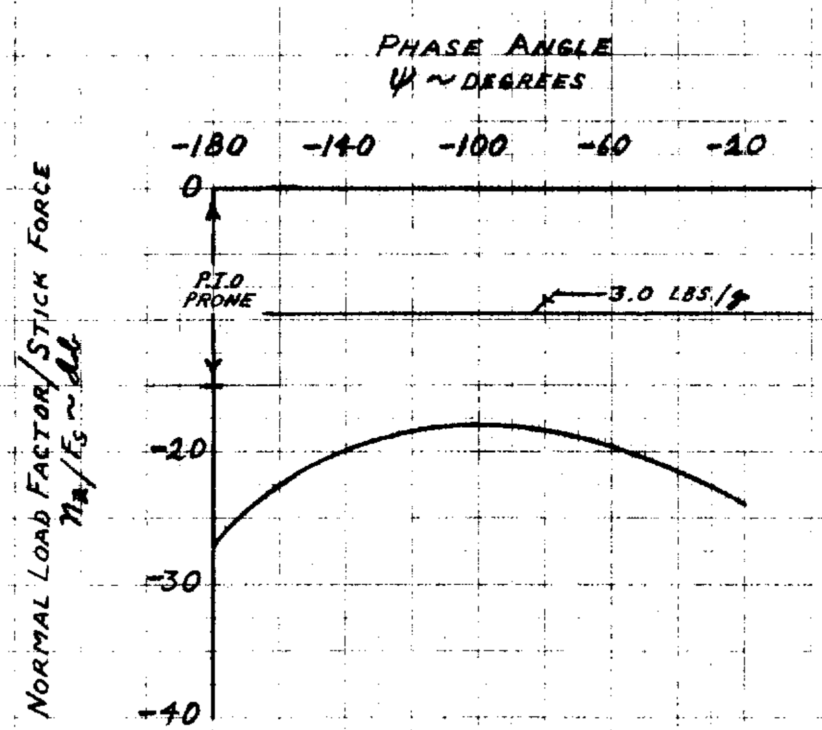


LONGITUDINAL PILOT INDUCED OSCILLATIONS

FIGURE 1 (3.2.2.3)

P.I.O. CRITERIA
F-5 ANALYTICAL DATA

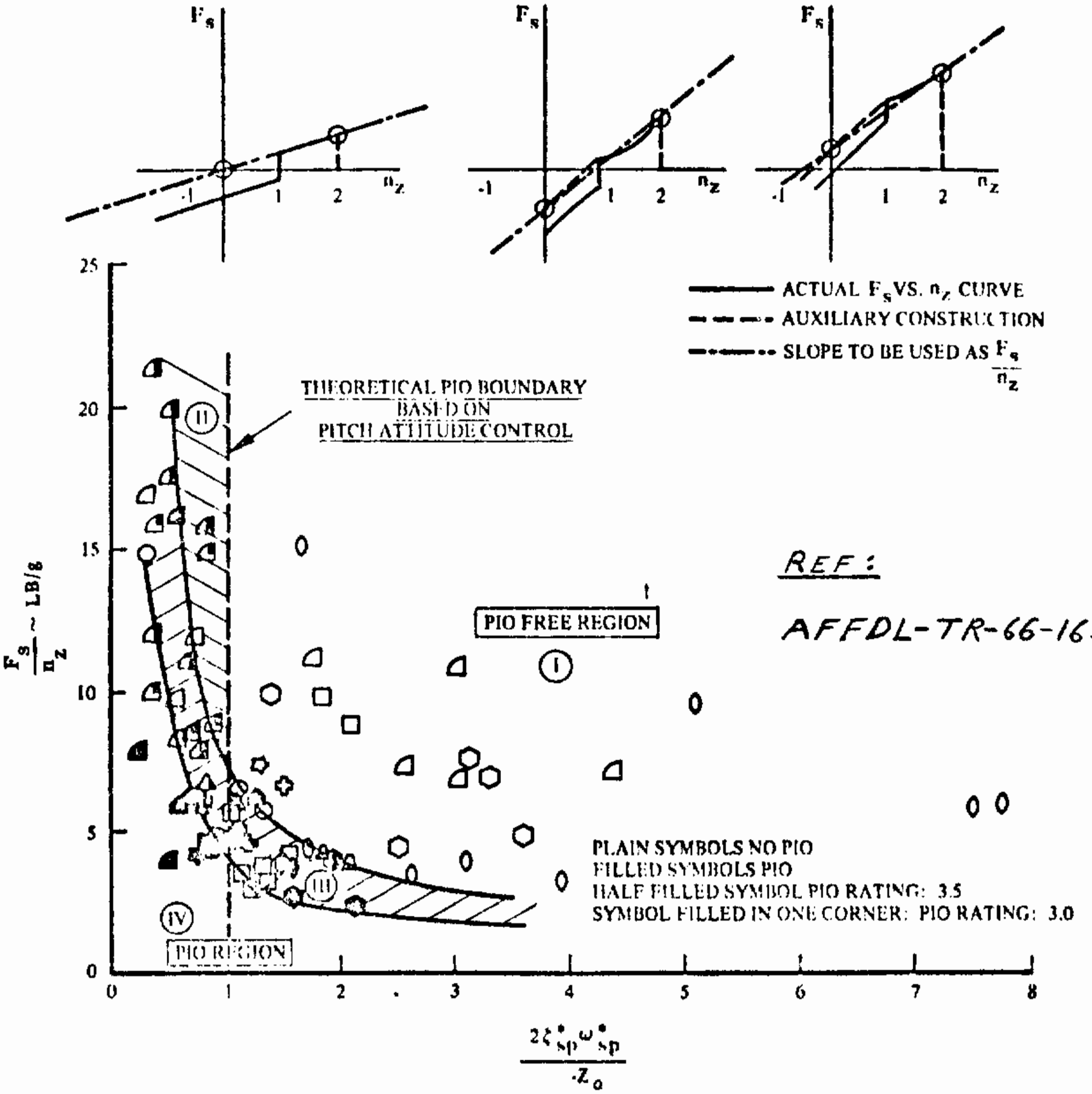
CONFIGURATION	ALTITUDE	MACH NO.
	10,000 FEET	0.875



LONGITUDINAL PILOT INDUCED OSCILLATIONS

FIGURE 2 (3.2.2.3)

P.I.O. BOUNDARIES



REF:

AFFDL-TR-66-163

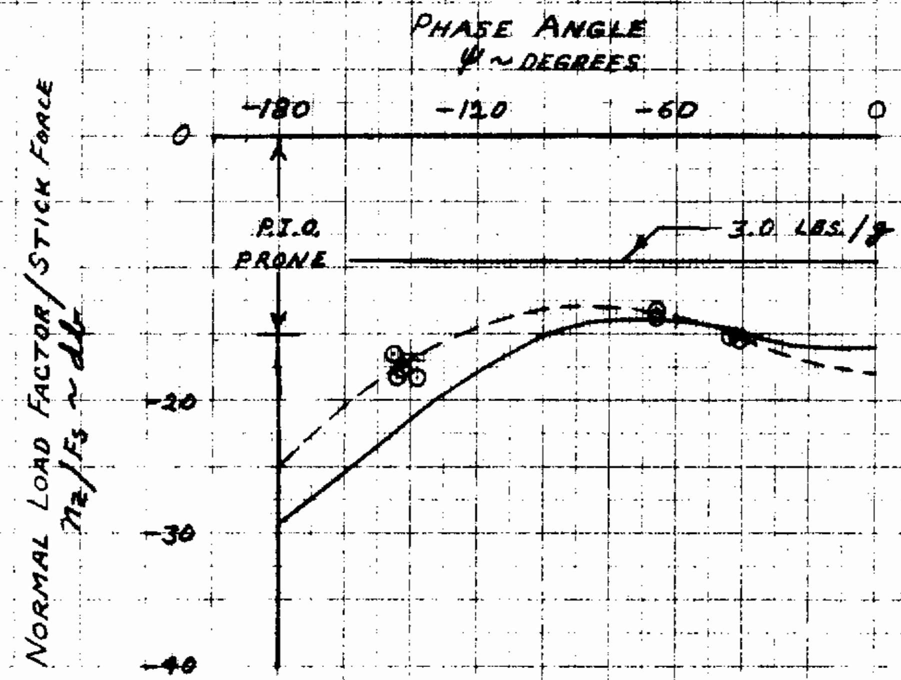
LONGITUDINAL PILOT INDUCED OSCILLATIONS

FIGURE 3 (3.2.2.3)

F-5 AMPLITUDE – PHASE ANGLE RELATIONSHIP FOR P.I.O. SUSCEPTIBILITY

CONFIGURATION	WEIGHT	C.G. POSITION	SYMBOL	MACH NO.	ALTITUDE
o———o	12,333 LBS.	26.0% MAC	———	0.80	10,000 FT.
			- - - - -	0.95	20,000 FT.
			⊙	0.95	20,000 FT.

NOTE:
⊙ ~ FLIGHT TEST DATA
- - - - - - ~ PREDICTED DATA
MANEUVERING FLAPS DOWN, AUGMENTERS OFF



LONGITUDINAL PILOT INDUCED OSCILLATIONS

FIGURE 4 (3.2.2.3)

Requirement

Paragraph 3.2.2.3.1 Transient control forces. The peak elevator-control forces developed during abrupt maneuvers shall not be objectionably light, and the buildup of control force during the maneuver entry shall lead the buildup of normal acceleration. Specifically, the following requirement shall be met when the elevator control is pumped sinusoidally. For all input frequencies, the ratio of the peak force amplitude to the peak normal load factor amplitude at the c.g. measured from the steady oscillation, shall be greater than:

Center-Stick Controllers ----- 3.0 pounds per g

Wheel Controllers ----------- 6.0 pounds per g

Comparison

Flight test data from F-5 sinusoidal maneuvers, performed with input frequencies of 0.22 Hz to 0.75 Hz were used to compare the F-5 characteristics with the requirements specified within this paragraph. Figure 1 (3.2.2.3.1) presents the data obtained and demonstrates agreement with the specified requirements.

Resolution

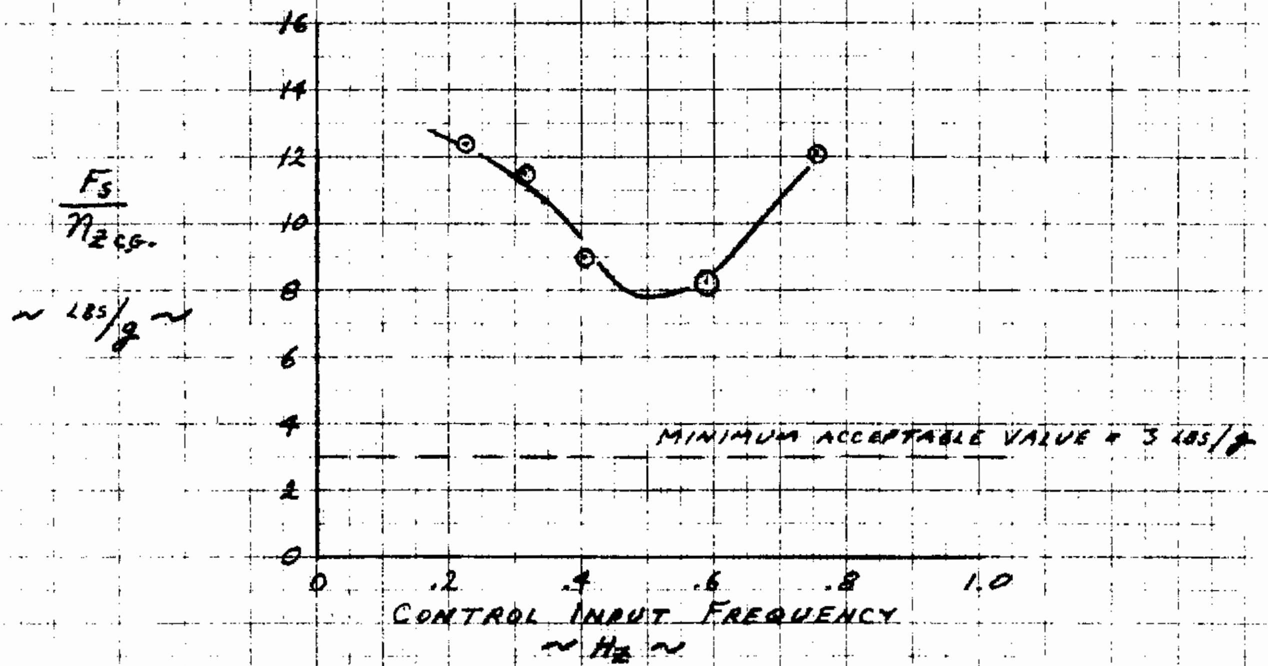
None

Recommendation

None

F-5 FLIGHT TEST DATA

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
o———o	23,000 FEET	0.95	12,800 LBS.	23.5 % MAC



TRANSIENT CONTROL FORCES

FIGURE 7 (3.2.2.3.1)

Requirement

Paragraph 3.2.3 Longitudinal control

Paragraph 3.2.3.1 Longitudinal control in unaccelerated flight. In erect unaccelerated flight at all service altitudes, the attainment of all speeds between V_S and V_{max} shall not be limited by the effectiveness of the longitudinal control, or controls.

Comparison

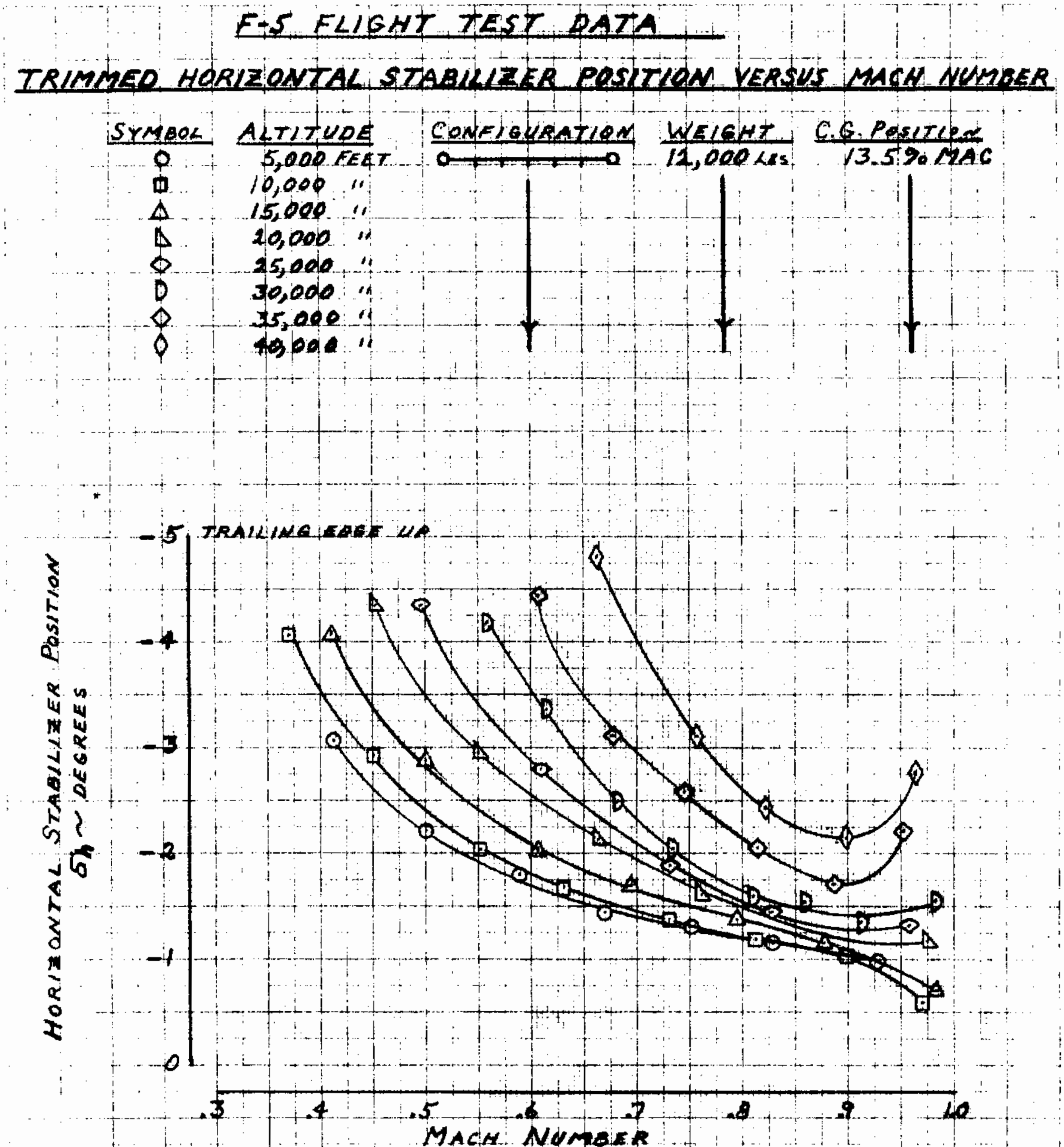
Data obtained, Figure 1 (3.2.3.1), from a series of steady state trimmed level flight runs were used to compare the F-5 longitudinal control effectiveness with the requirements of this paragraph. The horizontal stabilizer has a maximum travel of 5.5 degrees airplane-nose-down deflection to -17.0 degrees airplane-nose-up deflection. Although supersonic data have not been reduced in flight tests, performance flights to V_{max} indicated that the attainment of all speeds to V_{max} are possible with the available travel of the F-5 horizontal stabilizer.

Resolution

None

Recommendation

None



LONGITUDINAL CONTROL EFFECTIVENESS

FIGURE I (3.2.3.1)

Requirement

Paragraph 3.2.3.2 Longitudinal control in maneuvering flight. Within the Operational Flight Envelope, it shall be possible to develop, by use of the elevator control alone, the following range of load factors:

Levels 1 and 2 ----- n_o (-) to n_o (+)

Level 3 ------------ $n = 0.5g$ to the lower of:

a) n_o (+)

b) $n = \begin{cases} 2.0 \text{ for } n_o (+) \leq 3g \\ 0.5 \; [n_o (+) + 1] \text{ for } n_o(+) > 3 \text{ g.} \end{cases}$

This maneuvering capability is required at the 1g trim speed and, with trim and throttle settings not changed by the crew, over a range about the trim speed the lesser of ±15 percent or ±50 knots equivalent airspeed (except where limited by the boundaries of the Operational Flight Envelope). Within the Service and Permissible Flight Envelopes, the dive-recovery requirements of 3.2.3.5 and 3.2.3.6, respectively, shall be met.

Comparison

It is possible for the F-5 with the available horizontal tail to develop, within the Operational Flight Envelope, n_o (-) to n_o (+) as presented in Figures 3 (3.1.7) and 4 (3.1..7), respectively, for the clean loading and external stores configurations.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.3.3 Longitudinal control in takeoff. The effectiveness of the elevator control shall not restrict the takeoff performance of the airplane and shall be sufficient to prevent over-rotation to undesirable attitudes during takeoffs. Satisfactory takeoffs shall not be dependent upon use of the trimmer control during takeoff or on complicated control manipulation by the pilot. For nose-wheel airplanes it shall be possible to obtain, at 0.9 V_{min}, the pitch attitude which will result in takeoff at V_{min}. For tail-wheel airplanes, it shall be possible to maintain any pitch attitude up to that for a level thrust-line at 0.5 V_S for Class I airplanes and at V_S for Class II, III and IV airplanes. These requirements shall be met on hard-surfaced runways. In the event that an airplane has a mission requirement for operation from unprepared fields, these requirements shall be met on such fields.

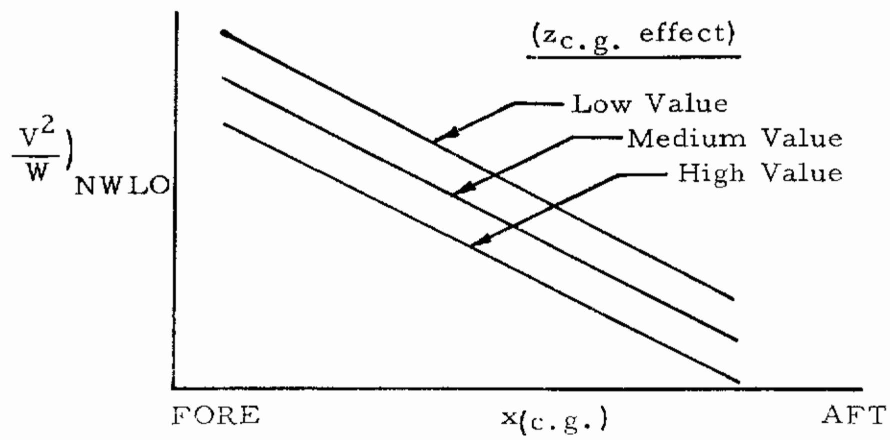
Comparison

The nosewheel liftoff speeds as determined from flight test for the F-5 in many external loadings are presented in Figure 1 (3.2.3.3). Note that the presentation format is V^2_{NWLO}/WEIGHT versus c.g. position. It can be shown that the complete nosewheel liftoff speed equation may be formated as:

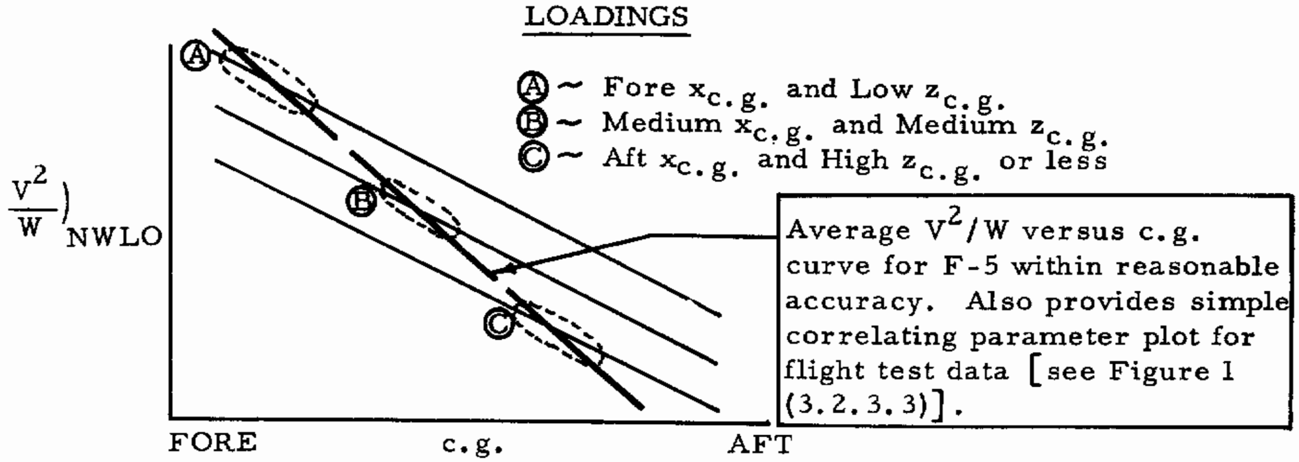
$$\frac{V^2}{W}_{NWLO} = f\ (\text{Aero moment/lift, thrust, } x_{c.g.},\ z_{c.g.})$$

where $x_{c.g.}$ and $z_{c.g.}$ are respectively the fwd/aft and vertical distances between the aircraft c.g. and the main gear pivot axis.

The calculated (predicted) effect of $x_{c.g.}$ and $z_{c.g.}$ on nosewheel liftoff speeds appears as shown below (with negligible errors due to thrust/weight changes when weight is changed at constant thrust):



The F-5 stores configurations exhibit c.g. variations as depicted below:



The NWLO flight data from Plot I, Figure 1 (3.2.3.3) are compared to the specification requirement in Plot II. V_{min} on Plot II is defined as 1.1 V_S (power off) as previously described; where V_S corresponds to $C_{L_{max}}$ as a function of c.g. V_{min} has been converted to $(V^2/W)_{min}$ for comparison with $(V^2/W)_{NWLO}$.

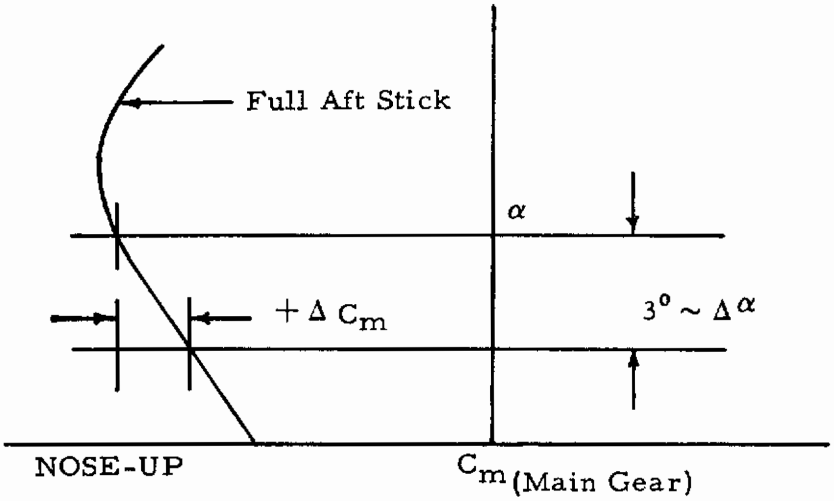
Takeoff speed of the F-5 is approximately 1.08 V_S when not nosewheel liftoff limited and about 5 knots above NWLO speed when the aircraft is NWLO limited (to account for rotation to takeoff attitude). These speeds were determined from both Contractor Cat I and USAF Cat II flight tests to be the recommended takeoff speeds.

As shown in Plot II the F-5 is nosewheel liftoff limited in terms of not being able to takeoff at V_{min} for c.g.'s ahead of 13.5% MAC (Point A on Plot II). Rotation capability at .9 V_{min} is not possible at c.g.'s forward of 21.5% MAC (Point B on Plot II).

Because a number of F-5 configurations have takeoff c.g.'s ahead of 13% MAC in heavy weight conditions the NWLO limited takeoff ground runs were improved on the CF-5A (Canadian) aircraft by physically lengthening the nose gear. This results in a 3° static pitch attitude with normal nosewheel oleo compression (about a 3° increase from the F-5A) and a 5° attitude at the oleo extended nosewheel liftoff condition.

With the extended nose gear installed the NWLO speeds determined from flight test are as shown in Plot III.

In general, the effect of extending the nose gear reduces the NWLO speed by a (V^2/W) of .3 to .4 due to the increase in lift and nose up pitching moment as depicted below.



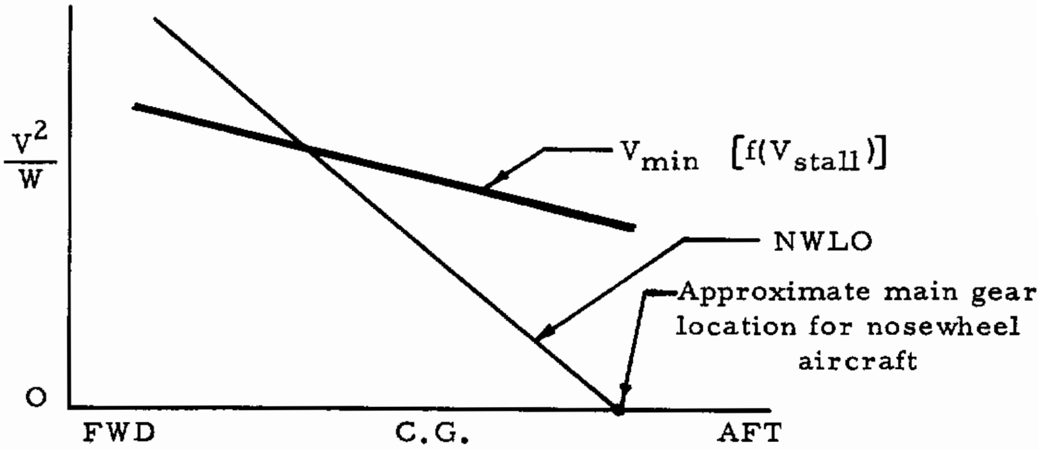
The specification requirements are compared in Plot IV. Note that V_{min} and .9 V_{min} in Plot IV are the same as those in Plot II while the takeoff speeds versus c.g. line from Plot II has been extended from 14.2% MAC to 6% MAC (Point A) before the actual takeoff becomes nosewheel liftoff limited. Although the CF-5 takes off at or less than V_{min}, in agreement with the specification, rotation to the pitch attitude is not possible at .9 V_{min} for c.g.'s ahead of about 12.5% MAC (Point B), in partial disagreement with the requirement.

Resolution

The F-5 with the extended nose gear (CF-5) demonstrates compliance with the requirement to takeoff at V_{min} while not complying with the requirement to rotate at .9 V_{min} at forward c.g. positions.

Pilot assessment of the takeoff characteristics of the CF-5 (extended nose gear) relative to the F-5 (standard nose gear) indicate that the F-5 was undesirable at forward c.g. positions while the CF-5 has quite acceptable characteristics.

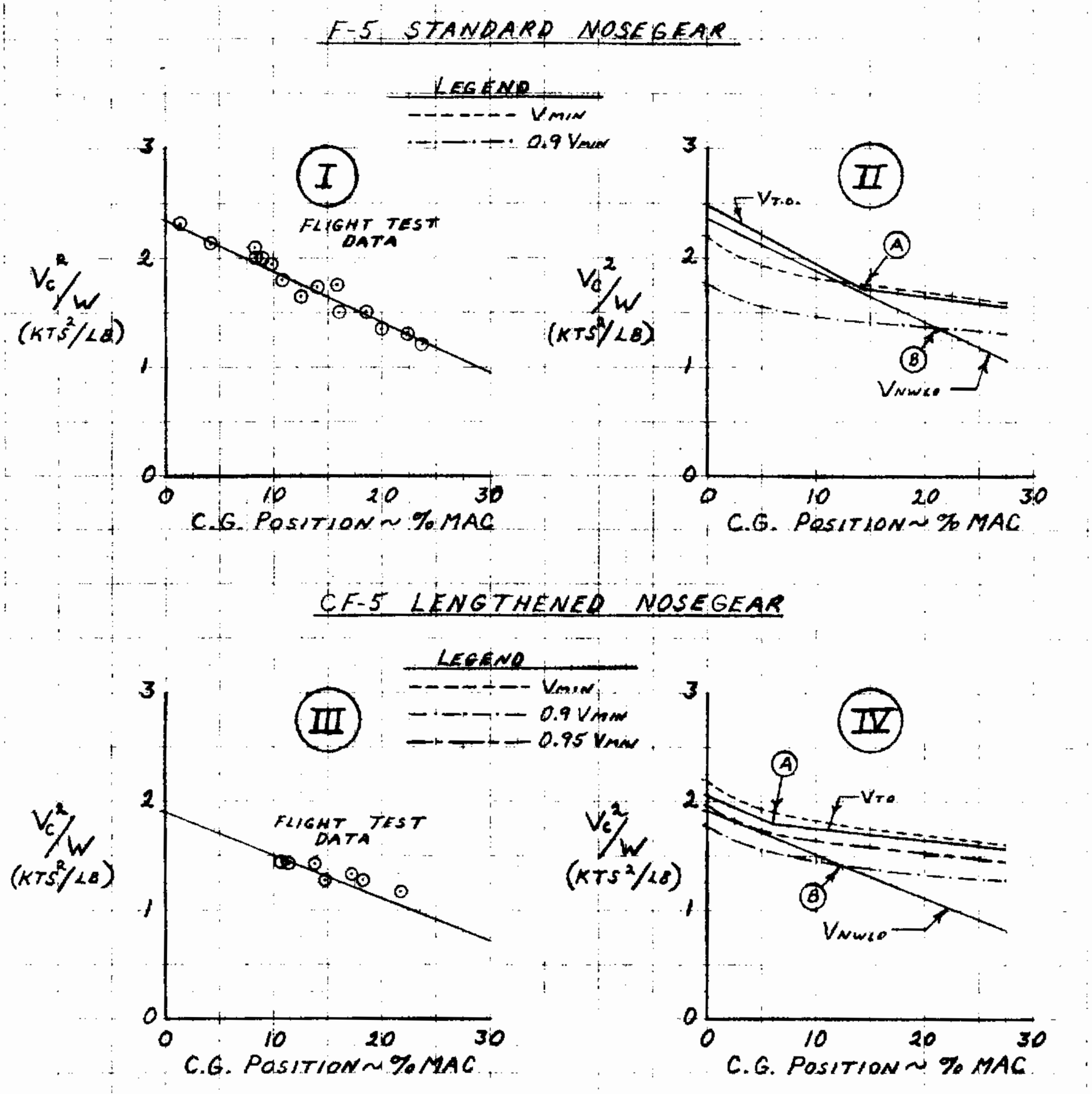
For airplanes with large c.g. travel such as the F-5, the percentage of V_{min} required to attain takeoff attitude is difficult to establish. The functional relationships between $\left(\frac{V^2}{W}\right)_{V_{min}}$ versus c.g. and $\left(\frac{V^2}{W}\right)_{NWLO}$ capability of any aircraft are completely different as shown below:



If the aircraft has a forward c.g. loading that is a primary mission loading then any requirement must pertain to that loading; however, if this loading were not a "primary mission" loading, then having the takeoff for this configuration limited because of a lack of longitudinal control would not be unreasonable. As shown in Plot IV, the F-5 takeoffs are not limited when the capability to rotate is present at .95 V_{min}. The "capability" is here defined as that corresponding to full aft stick with the knowledge that on high thrust/weight aircraft the rapid forward acceleration during takeoff will require the pilot to initiate his aft stick at a speed less than nosewheel liftoff speed in order to lift off at V_{min}.

Recommendation

It is recommended that the minimum speed, to initiate pitch attitude which will result in takeoff at V_{min} for nosewheel airplanes, be changed to .95 V_{min} for a forward c.g. primary operational mission configuration established in conjunction with the procuring activity.



LONGITUDINAL CONTROL IN TAKEOFF

FIGURE 1 (3.2.3.3)

Requirement

Paragraph 3.2.3.3.1 Longitudinal control in catapult takeoff. On airplanes designed for catapult takeoff, the effectiveness of the elevator control shall be sufficient to prevent the airplane from pitching up or down to undesirable attitudes in catapult takeoffs at speeds ranging from the minimum safe launching speed to a launching speed 30 knots higher than the minimum. Satisfactory catapult takeoffs shall not depend upon complicated control manipulations by the pilot.

Comparison

None. The F-5 airplane is not designed or equipped for catapulting.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.3.3.2 Longitudinal control force and travel in takeoff. With the trim setting optional but fixed, the elevator-control forces required during all types of takeoffs for which the airplane is designed, including short-field takeoffs and assisted takeoffs such as catapult or rocket-augmented, shall be within the following limits:

Nose-wheel and bicycle-gear airplanes

Classes I, IV-C -------	20 pounds pull to 10 pounds push
Classes II-C, IV-L ----	30 pounds pull to 10 pounds push
Classes II-L, III -------	50 pounds pull to 20 pounds push

Tail-wheel airplanes

Classes I, II-C, IV	20 pounds push to 10 pounds pull
Classes II-L, III	35 pounds push to 15 pounds pull

The elevator-control travel during these takeoffs shall not exceed 75 percent of the total travel, stop-to-stop. For purposes of this requirement, the term takeoff includes the ground run, rotation and lift-off, the ensuing acceleration to V_{max} (TO), and the transient caused by assist cessation. Takeoff power shall be maintained until V_{max} (TO) is reached, with the landing gear and high lift devices retracted in the normal manner at speeds from $V_{o_{min}}$ (TO) to V_{max} (TO).

Comparison

Figure 1 (3.2.3.3.2) presents the longitudinal stick forces required to obtain the horizontal tail deflection necessary for takeoff. The stick forces required for takeoff are less than 30 pounds throughout the c.g. range of the F-5 with the assigned takeoff trim increments. The trim increments range is from 7 units to 10 units, and the maximum elevator control travel necessary to obtain takeoff δ_h is less than 55% of the total elevator control travel, stop to stop. Agreement of the F-5 takeoff characteristics with the requirements of this paragraph is exhibited.

Resolution

None

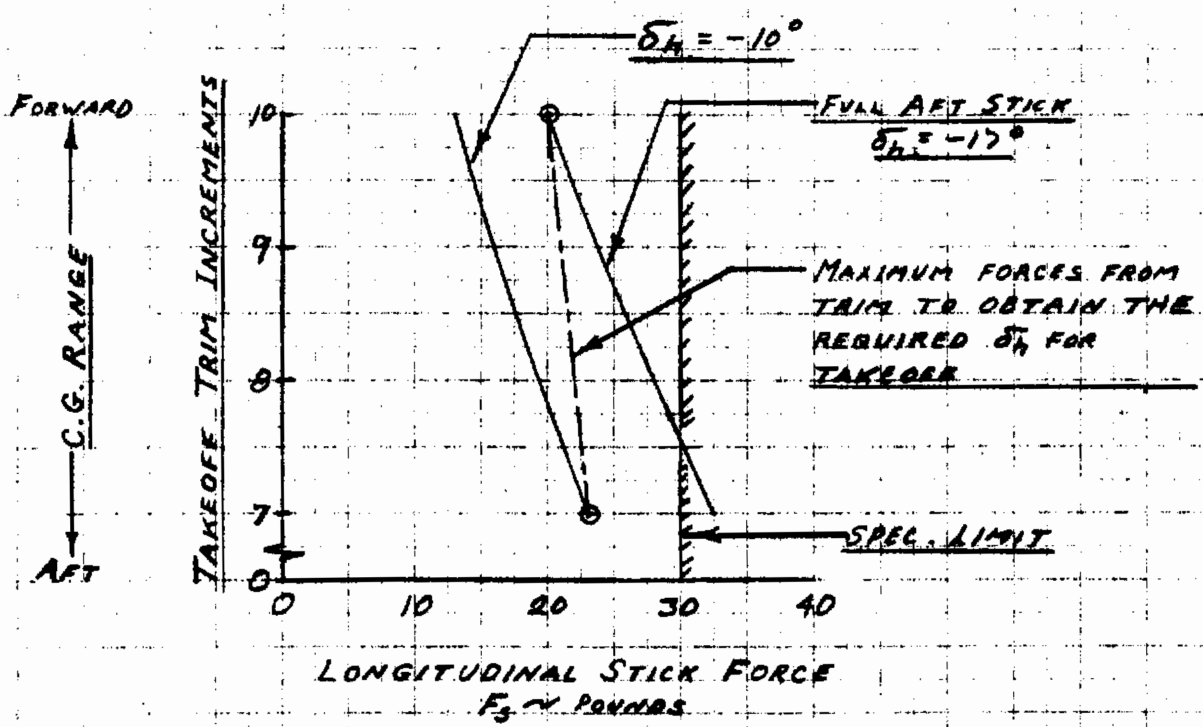
Recommendation

None

F-5

TAKEOFF δ_h REQUIRED

BASED ON FLIGHT TEST



LONGITUDINAL CONTROL FORCE IN TAKEOFF

FIGURE 1 (3.2.3.3.2)

Requirement

Paragraph 3.2.3.4 Longitudinal control in landing. The elevator control shall be sufficiently effective in the landing Flight Phase in close proximity to the ground, that:

a. The geometry-limited touchdown attitude can be maintained in the level flight, or
b. The lower of V_S (L) or the guaranteed landing speed can be obtained.

This requirement shall be met with the airplane trimmed for the approach Flight Phase at the recommended approach speed. The requirements of 3.2.3.4 and 3.2.3.4.1 define Levels 1 and 2. For Level 3, it shall be possible to execute safe approaches and landings in the presence of atmospheric disturbances.

Comparison

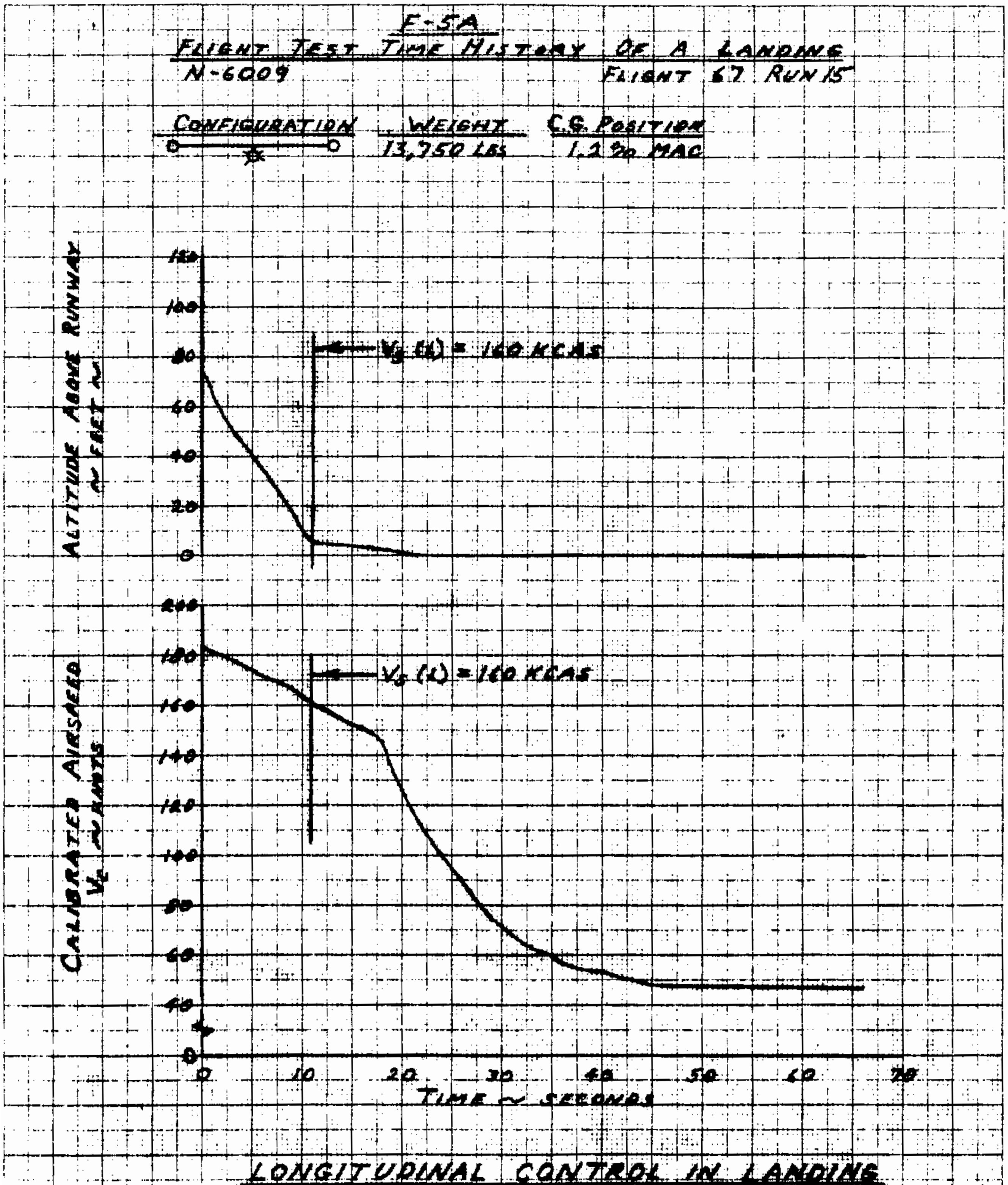
F-5 flight test time history of a landing with a representative critical c.g. is presented in Figure 1a (3.2.3.4) through Figure 1c (3.2.3.4). The data exhibit that with a critical forward c.g., the effectiveness of the elevator control is such that V_S (L) can be obtained in close proximity to the ground, thereby comparing favorably with the requirements of this paragraph.

Resolution

None

Recommendation

None



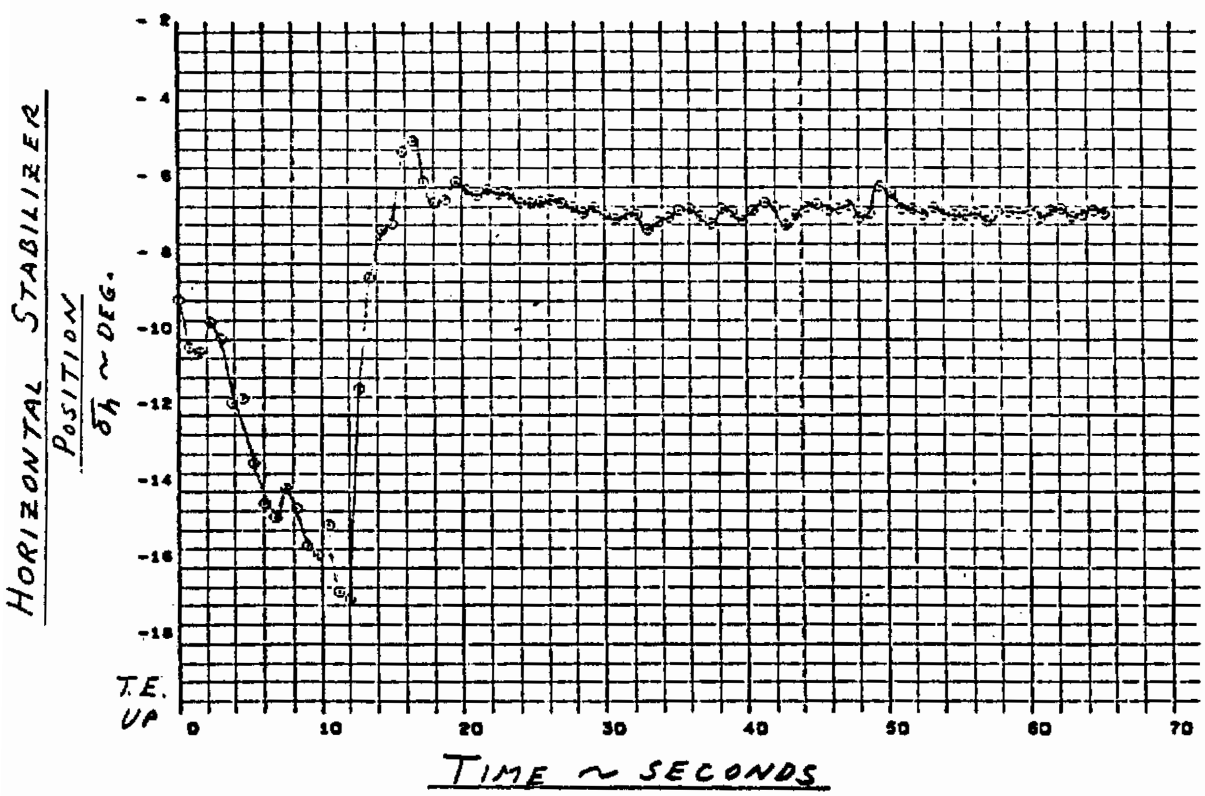
LONGITUDINAL CONTROL IN LANDING

FIGURE 16 (3.2.3.4)

F-5A

FLIGHT TEST TIME HISTORY OF A LANDING

N-6009 FLIGHT 67 RUN 15



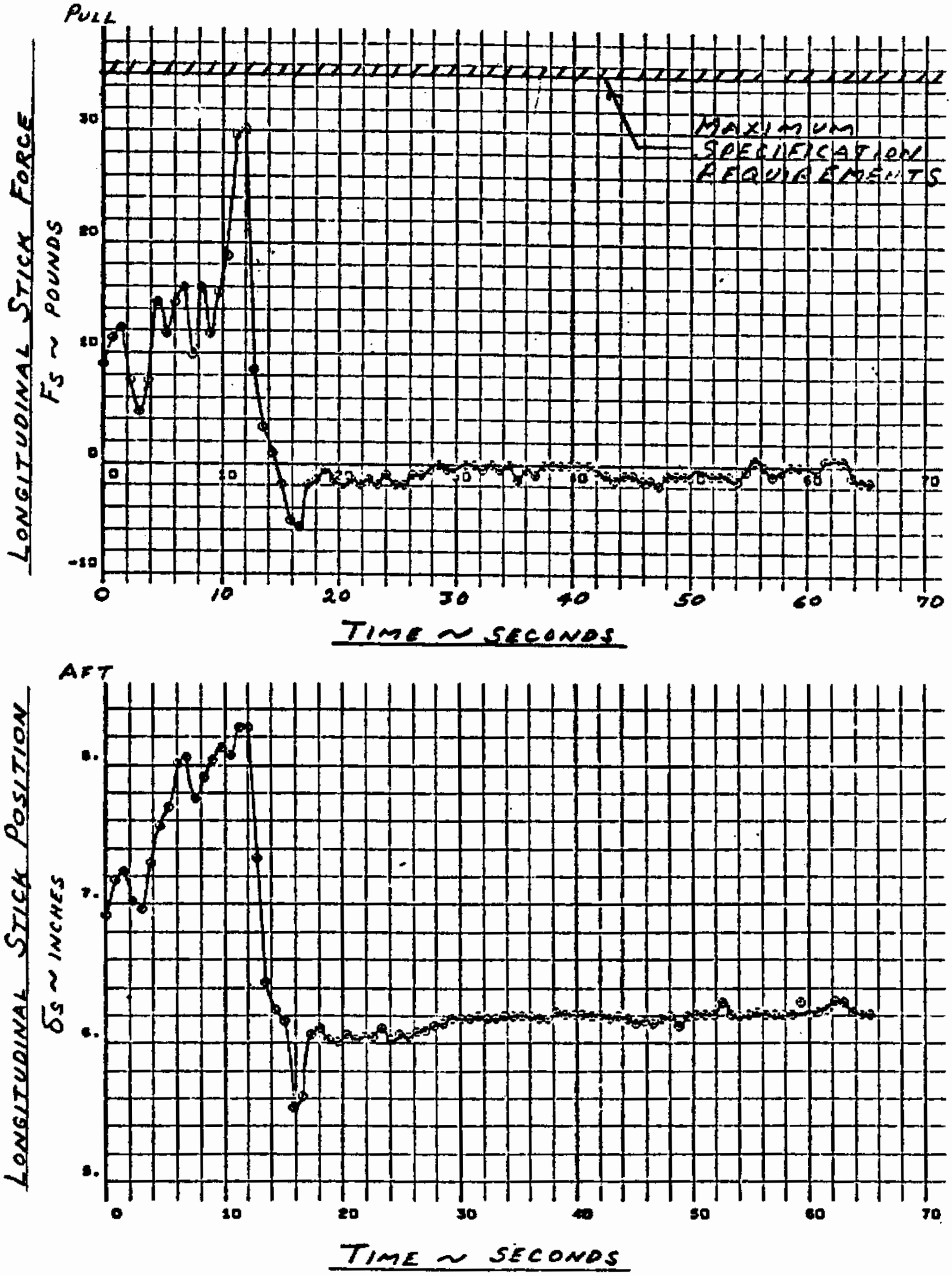
LONGITUDINAL CONTROL IN LANDING

FIGURE 1b (3.2.3.4)

F-5A

FLIGHT TEST TIME HISTORY OF A LANDING

N6009 FLIGHT 67 RUN 15



LONGITUDINAL CONTROL IN LANDING

FIGURE 1c (3.2.3.4)

Requirement

Paragraph 3.2.3.4.1 Longitudinal control forces in landing. The elevator-control forces required to meet the requirements of 3.2.3.4 shall be pull forces and shall not exceed:

Classes I, II-C, IV ---- 35 pounds

Classes II-L, III ------ 50 pounds

Comparison

Figure 1c (3.2.3.4) presents an F-5 flight test time history of a landing at a critical c.g. The data show that the maximum force was pull force of 30 pounds, well within the requirement of 35 pounds maximum.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.2.3.5 Longitudinal control forces in dives - Service Flight Envelope. With the airplane trimmed for level flight at speeds throughout the Service Flight Envelope, the elevator control forces in dives to all attainable speeds within the Service Flight Envelope shall not exceed 50 pounds push or 10 pounds pull for airplanes with center-stick controllers, nor 75 pounds push or 15 pounds pull for airplanes with wheel controllers. In similar dives, but with trim optional following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within the limits of 10 pounds push or pull for airplanes with center-stick controllers, and 20 pounds push or pull for airplanes with wheel controllers. The forces required for recovery from these dives shall be in accordance with the gradients specified in 3.2.2.2.1 although speed may vary during the pullout.

Comparison

Flight test data, from a dive starting at 48,000 feet with pullout initiated at 26,500 feet, were used to compare F-5 characteristics with the requirements of this paragraph. This comparison is exhibited in Figure 1 (3.2.3.5).

The F-5 control system is such that with control forces trimmed to zero and longitudinal stick position at 4.6 inches aft, the flight test data in Figure 1 (3.2.3.5) show that the maximum longitudinal control force required to full forward stick is approximately 35 pounds push with no retrim. This is well within the 50 pounds limit specified.

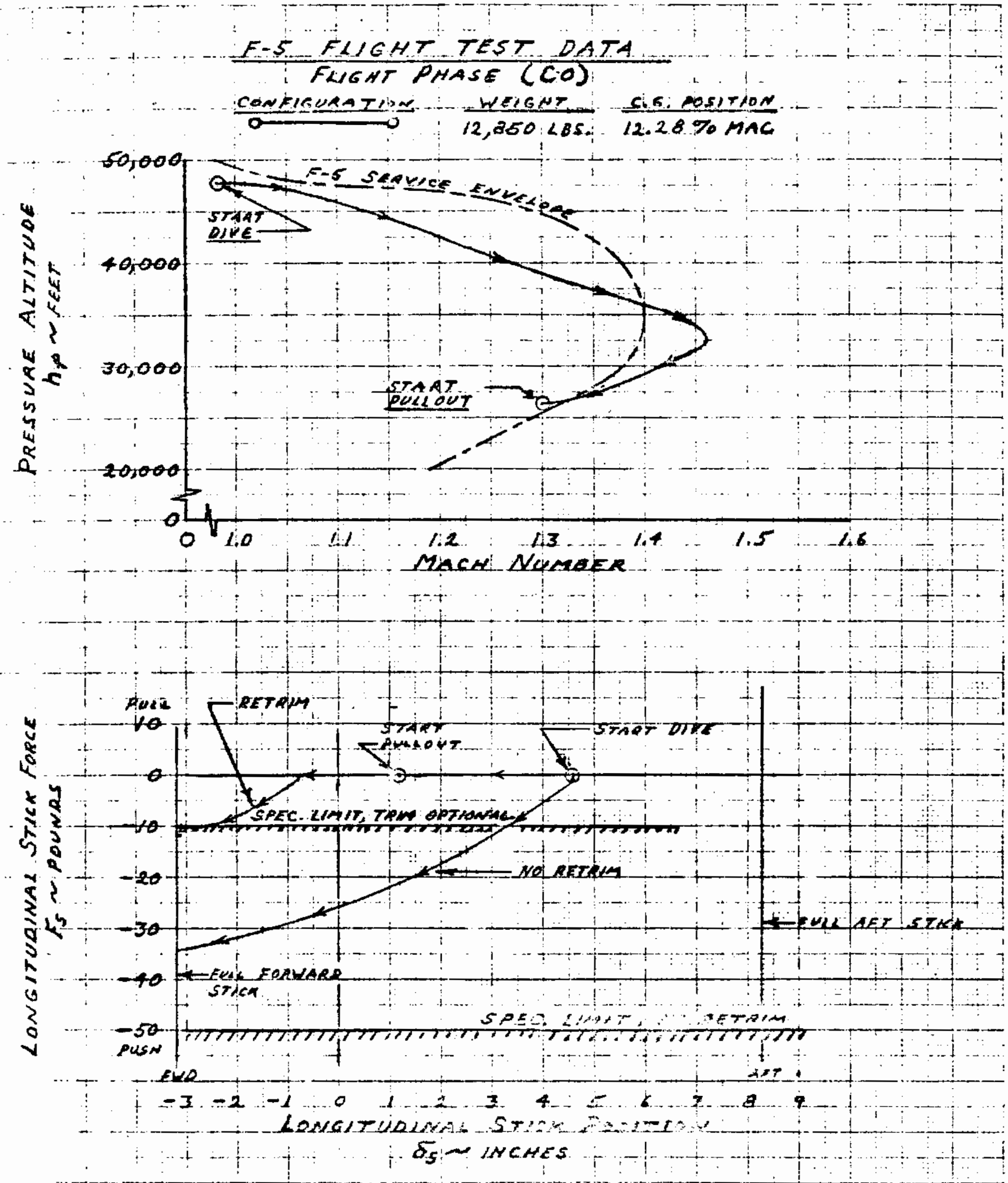
With trim optional, the maximum push force required to full forward stick is approximately 10 pounds. However, the maximum stick movement in the dive reached a stick position of 1.2 inches aft, well within the available control system trim, holding forces to zero. During the recovery phase of the dive, the control force gradients were within the limits specified in Paragraph 3.2.2.2.1.

Resolution

None

Recommendation

None



LONGITUDINAL CONTROL FORCES IN DIVES

SERVICE FLIGHT

FIGURE 1 (3.2.3.5)

Requirement

Paragraph 3.2.3.6 Longitudinal control forces in dives - Permissible Flight Envelope. With the airplane trimmed for level flight at V_{MAT} but with trim optional in the dive, it shall be possible to maintain the elevator control force within the limits of 50 pounds push or 35 pounds pull in dives to all attainable speeds within the Permissible Flight Envelope. The force required for recovery from these dives shall not exceed 120 pounds. Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

Comparison

Flight test data, from a dive starting at 48,000 feet with pullout initiated at 20,500 feet, were used to compare F-5 characteristics with the requirements of this paragraph. This comparison is exhibited in Figure 1 (3.2.3.6).

The F-5 control system is such that with control forces trimmed to zero and longitudinal stick position at approximately 4 inches aft, the maximum longitudinal control force required to full forward stick is approximately 32 pounds push, with no retrim, however, the pilot required only 20 pounds push as shown in the flight test data. With trim optional, the maximum push force required to full forward stick is approximately 10 pounds.

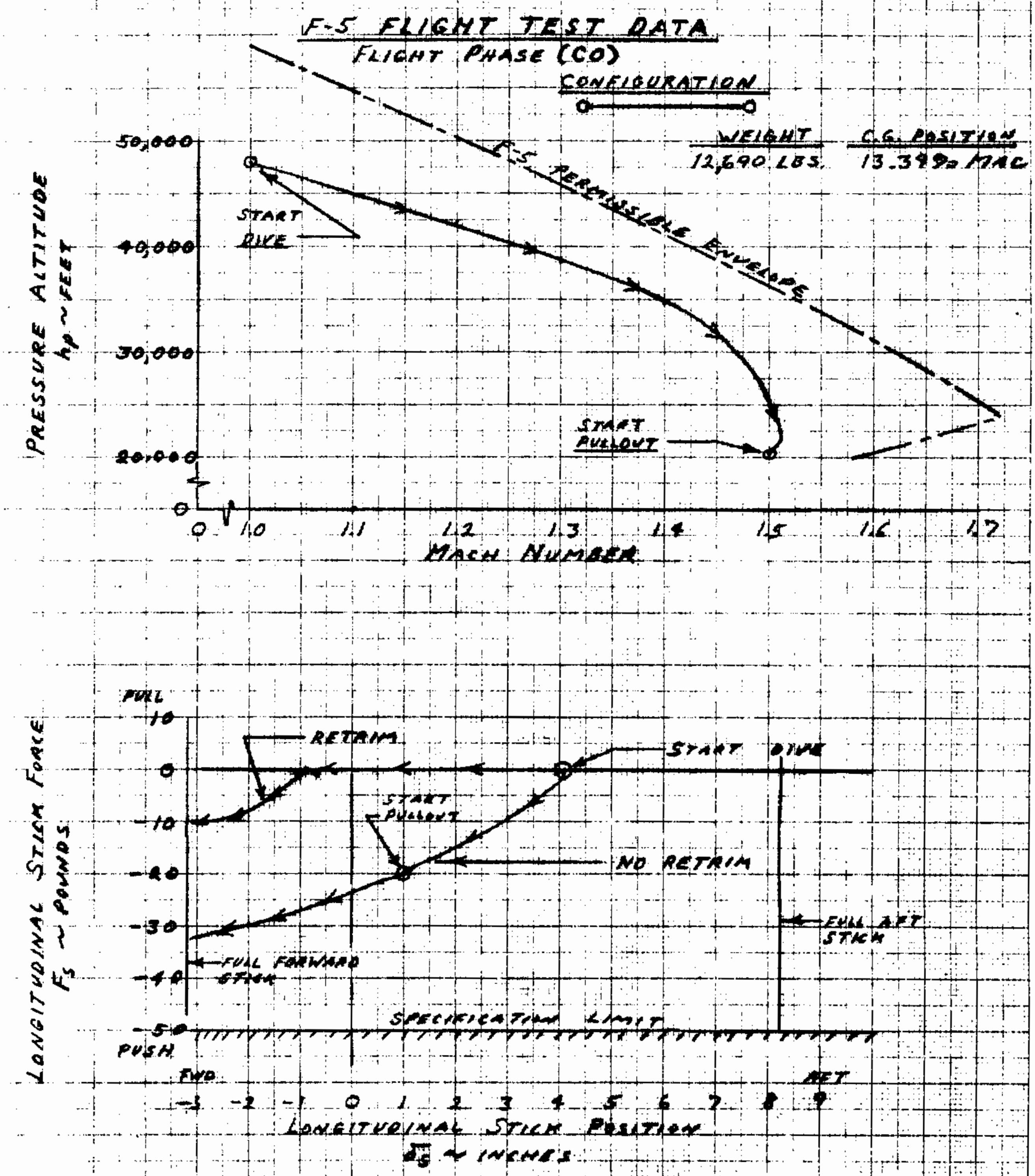
These values are well within the 50 pounds limit specified. The maximum stick force required for recovery from the dive was approximately 50 pounds well below the 120 pounds allowable.

Resolution

None

Recommendation

None



LONGITUDINAL CONTROL FORCES IN DIVES
PERMISSIBLE FLIGHT ENVELOPE

FIGURE 1 (3.2.3.6)

Requirement

Paragraph 3.2.3.7 Longitudinal control in sideslips. With the airplane trimmed for straight, level flight with zero sideslip, the elevator-control force required to maintain constant speed in steady sideslips with up to 50 pounds of rudder pedal force in either direction shall not exceed the elevator-control force that would result in a 1g change in normal acceleration. In no case, however, shall the elevator-control force exceed:

Center-stick controllers -------- 10 pounds pull to 3 pounds push

Wheel controllers -------------- 15 pounds pull to 10 pounds push

If a variation of elevator-control force with sideslip does exist, it is preferred that increasing pull force accompany increasing sideslip, and that the magnitude and direction of the force change be similar for right and left sideslips. These requirements define Levels 1 and 2. For Level 3, there shall be no uncontrollable pitching motions associated with the sideslips discussed above.

Comparison

Flight test data used to compare F-5 characteristics with the requirements of this paragraph are presented in Table 1 (3.2.3.7). Favorable comparison is exhibited in all except in two of the flight conditions. These conditions were for a five store configuration at 20,000 feet and M= .55, where the stick forces were slightly higher than the maximum allowed. However, no adverse pilot comments were reported.

Resolution

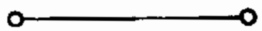
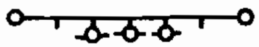
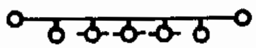
Partial disagreement between the F-5 known flight test results and requirements of this paragraph exists. Nevertheless, the specification is considered reasonable and need not be changed at this time due to insufficient amount of data to substantiate new specification values.

Recommendation

None

Contrails

FLIGHT TEST DATA
NF-5A 1001

CONFIGURATION	FLIGHT	RUN	WEIGHT (POUNDS)	C.G. POS. (% M.A.C.)	ALTITUDE (FEET)	M.N.	LONGITUDINAL STICK FORCE AT 50 LB RUDDER PEDAL FORCE (POUNDS)
	201	17	12,140	13.85	20,000	.44	10.0 pull
	201	18	12,100	13.90		.45	7.0 pull
	201	12	12,520	13.60		.79	5.5 pull
	183	8	15,550	9.46		.50	7.5 pull
	182	13	14,570	9.06		.70	3.5 pull
	187	9	16,400	17.50		.55	12.0 pull
	185	34	14,900	16.31		.79	7.0 pull
	185	30	15,700	15.85		.85	4.0 pull
	187	12	16,000	17.50		.54	13.5 pull
	185	32	15,220	16.05		.78	5.5 pull
	185	28	16,300	15.61		.84	8.0 pull

LONGITUDINAL CONTROL IN SIDESLIPS

TABLE 1 (3.2.3.7)

Requirement

Paragraph 3.3 Lateral-directional flying qualities

Paragraph 3.3.1 Lateral-directional mode characteristics

Paragraph 3.3.1.1 Lateral-directional oscillations (Dutch roll). The frequency, ω_{nd}, and damping ratio, ζ_d, of the lateral-directional oscillations following a rudder disturbance input shall exceed the minimums in table VI. The requirements shall be met with cockpit controls fixed and with them free, in oscillations of any magnitude that might be experienced in operational use. If the oscillation is nonlinear with amplitude, the requirement shall apply to each cycle of the oscillation. Residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance. For Category A Flight Phases, angular deviations shall be less than ±3 mils. With the control surfaces fixed, ω_{nd} shall always be greater than zero.

TABLE VI. Minimum Dutch Roll Frequency and Damping

Level	Flight Phase Category	Class	Min ζ_d *	Min $\zeta_d \omega_{n_d}$ *, rad/sec.	Min ω_{n_d}, rad/sec.
1	A	I, IV	0.19	0.35	1.0
		II, III	0.19	0.35	0.4**
	B	All	0.08	0.15	0.4**
	C	I, II-C, IV	0.08	0.15	1.0
		II-L, III	0.08	0.15	0.4**
2	All	All	0.02	0.05	0.4**
3	All	All	0.02	-	0.4**

*The governing damping requirement is that yielding the larger value of ζ_d.

**Class III airplanes may be excepted from the minimum ω_{n_d} requirement, subject to approval by the procuring activity, if the requirements of 3.3.2 through 3.3.2.4.1, 3.3.5 and 3.3.9.4 are met.

When $\omega_{nd}^2 \left|\phi/\beta\right|_d$ is greater than 20 $(\text{rad/sec})^2$, the minimum $\zeta_d\, \omega_{nd}$ shall be increased above the $\zeta_d\, \omega_{nd}$ minimums listed above by:

Level 1 - $\Delta\zeta_d \quad \omega_{n_d} = .014\,(\omega_{n_d}^2 \mid \phi/\beta \mid_d - 20)$

Level 2 - $\Delta\zeta_d \quad \omega_{n_d} = .009\,(\omega_{n_d}^2 \mid \phi/\beta \mid_d - 20)$

Level 3 - $\Delta\zeta_d \quad \omega_{n_d} = .005\,(\omega_{n_d}^2 \mid \phi/\beta \mid_d - 20)$

with ω_{n_d} in rad/sec.

Comparison

Flight test data from rudder kick maneuvers were used for comparison of F-5 lateral-directional oscillation characteristics with the requirements of this paragraph. These data are presented in graphical form for Flight Phase Category A in Figures 1 (3.3.1.1) through 3 (3.3.1.1) for a clean, a three-store, and a five-store configuration, respectively. Figures 4 (3.3.1.1) and 5 (3.3.1.1) present Flight Phase Category C data for a two-store and single-store configuration. In cases where $\omega_{n_d}^2 |\phi/\beta|_d$ exceeded 20 (rad/sec)^2, consequently increasing the minimum allowable value of $\zeta_d\ \omega_{n_d}$, the value of $\zeta_d\ \omega_{n_d}$ corresponding to the largest value of $\omega_{n_d}^2\ |\phi/\beta|_d$ was used in the figures. This was done to preclude presenting $\zeta_d\,\omega_{n_d}$ boundaries for each data point where $\omega_{n_d}^2\ |\phi/\beta|_d > 20$ (rad/sec)^2; however the most stringent boundary is presented.

The data in the figures demonstrate that these F-5 lateral-directional oscillation characteristics compare favorably with the requirements of this paragraph. During the F-5 flight test program, angle of sideslip was obtained from a vane mounted on the pitot-static boom at the airplane nose. Because boom dynamics and bending were superimposed in the data, compliance with the requirement that residual oscillations be within ±3 mils could not be determined. For F-5A compliance demonstration with this portion of the requirement, special instrumentation and flight test data reduction techniques would be necessary.

Resolution

None

Recommendation

None

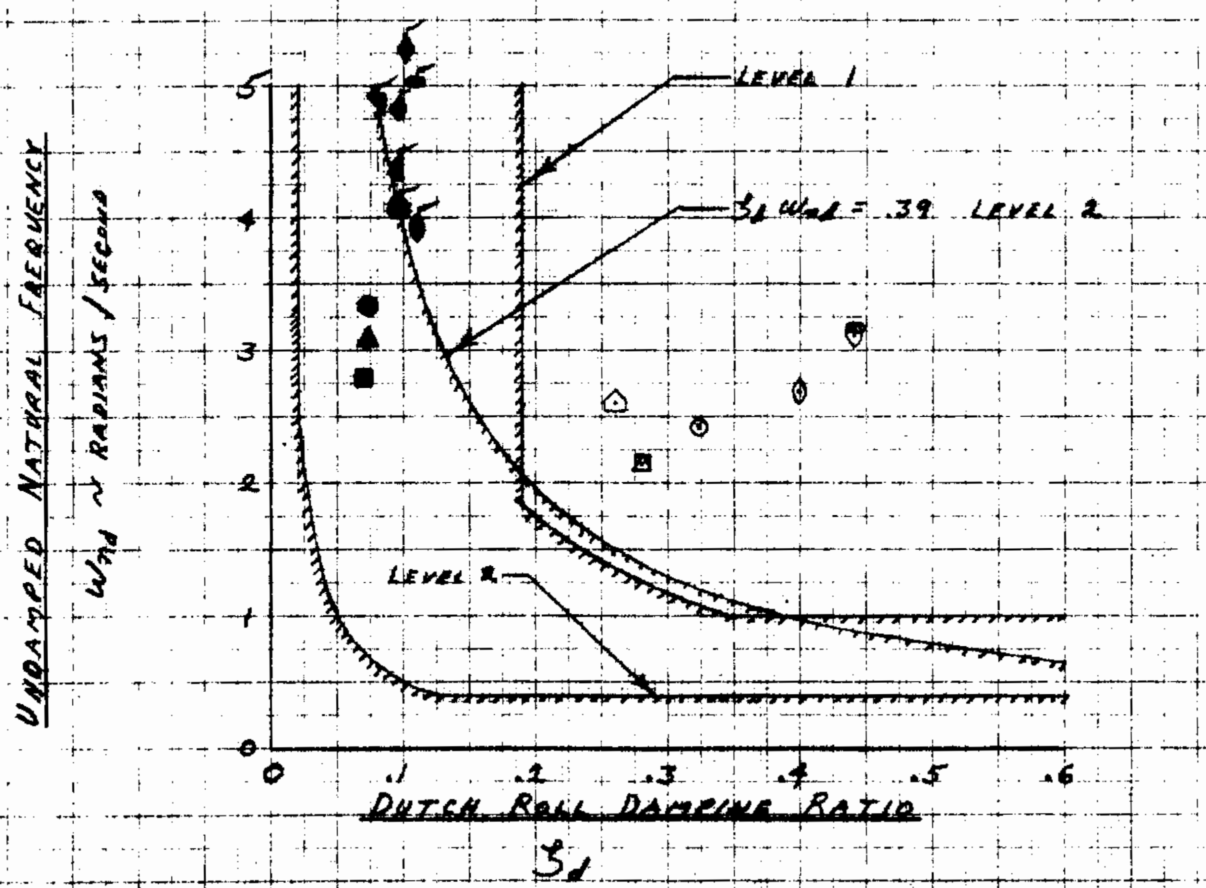
F-5 FLIGHT TEST DATA

CONFIGURATION

SYM.	ALT.	M.N.	SYM.	ALT.	MN
○	40,000 FT	.96	(flagged)	20,000 FT.	.85
△	"	.90	(flagged)	"	.80
□	"	.86	(flagged)	10,000 FT.	.79
◇	20,000 FT	.95	(flagged)	"	.80
▽	"	.90	(flagged)	"	.70

NOTE:

1. FILLED SYMBOLS INDICATE STABILITY AUGMENTERS OFF
2. FLAGGED SYMBOLS INDICATE $\omega_{n_d}^2 \, |\phi/\beta|_d > 20 \, (RAD^2/SEC)^2$

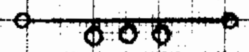


LATERAL-DIRECTIONAL OSCILLATIONS

FIGURE 1 (3.3.1.1)

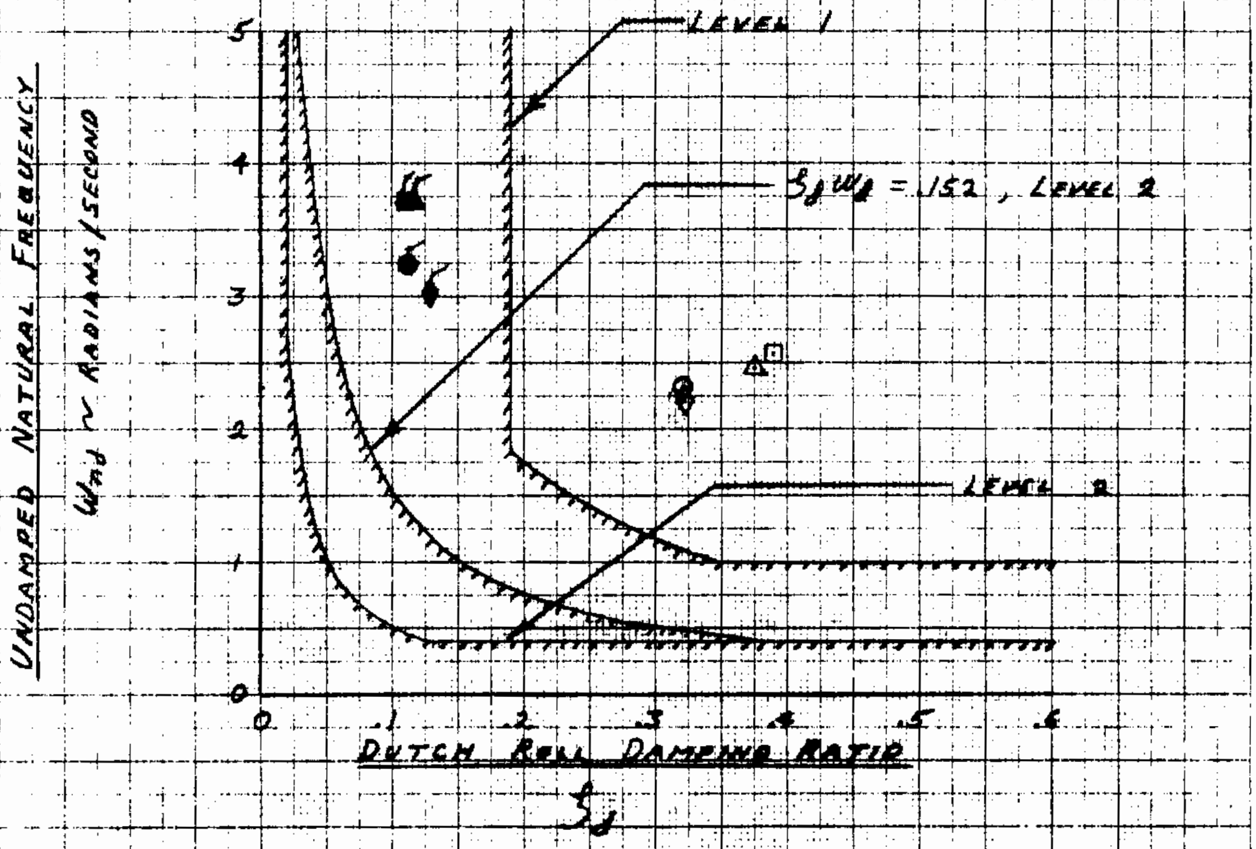
F-5 FLIGHT TEST DATA

CONFIGURATION



SYMBOL	ALTITUDE	MN
○	20,000 FT.	.84
△	15,000 FT.	.84
□	10,000 FT.	.80
◊	10,000 FT.	.70

NOTE:
1. FILLED SYMBOLS INDICATE STABILITY AUGMENTERS OFF.
2. FLAGGED SYMBOLS INDICATE $\omega_{n_d}^2 \, |\phi/\beta|_d > 20 \, (RAD^2/SEC)^2$

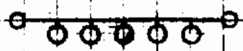


LATERAL-DIRECTIONAL OSCILLATIONS

FIGURE 2 (3.3.1.1)

F-5 FLIGHT TEST DATA

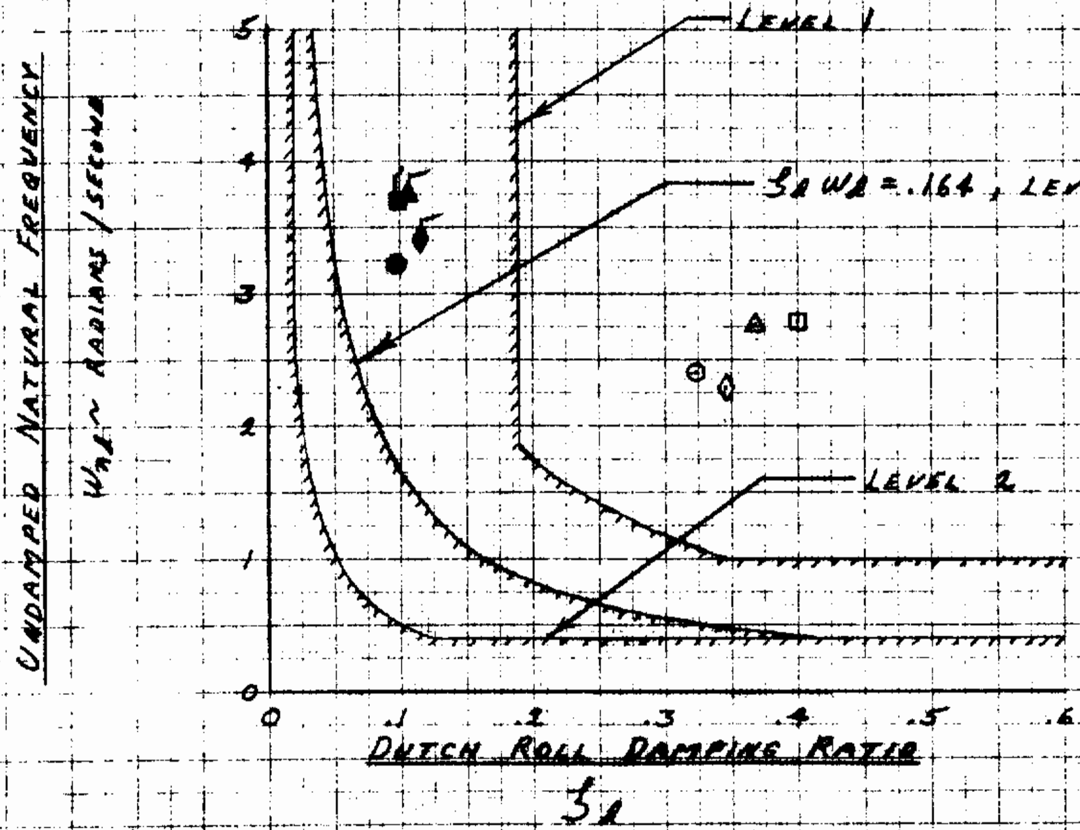
CONFIGURATION



SYMBOL	ALTITUDE	MN
○	10,000 FT.	.70
△	13,000 FT.	.83
□	10,000 FT.	.78
◇	10,000 FT.	.68

NOTE:

1. FILLED SYMBOLS INDICATE STABILITY AUGMENTERS OFF
2. FLAGGED SYMBOLS INDICATE $\omega_{n_d}^2 |\phi/\beta| > 20 (\text{RAD/SEC})^2$



LATERAL-DIRECTIONAL OSCILLATIONS

FIGURE 3 (3.3.1.1)

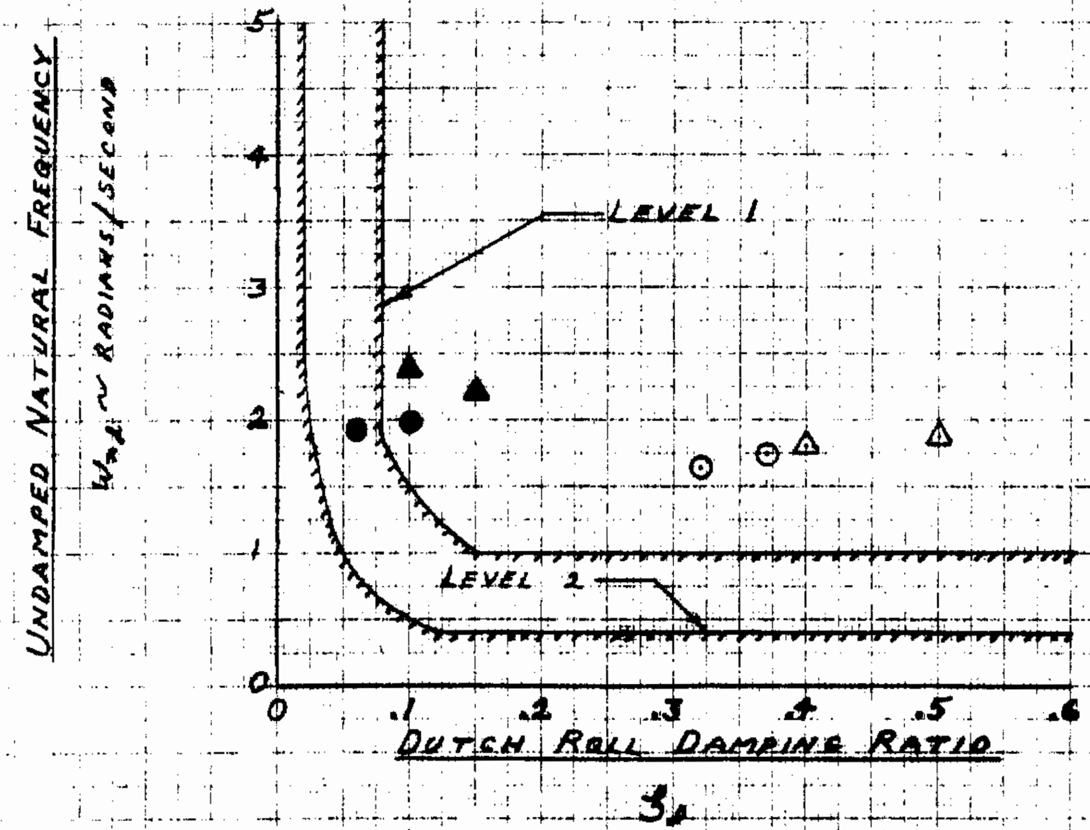
F-5 FLIGHT TEST DATA

CONFIGURATION

FLIGHT PHASE CATEGORY C

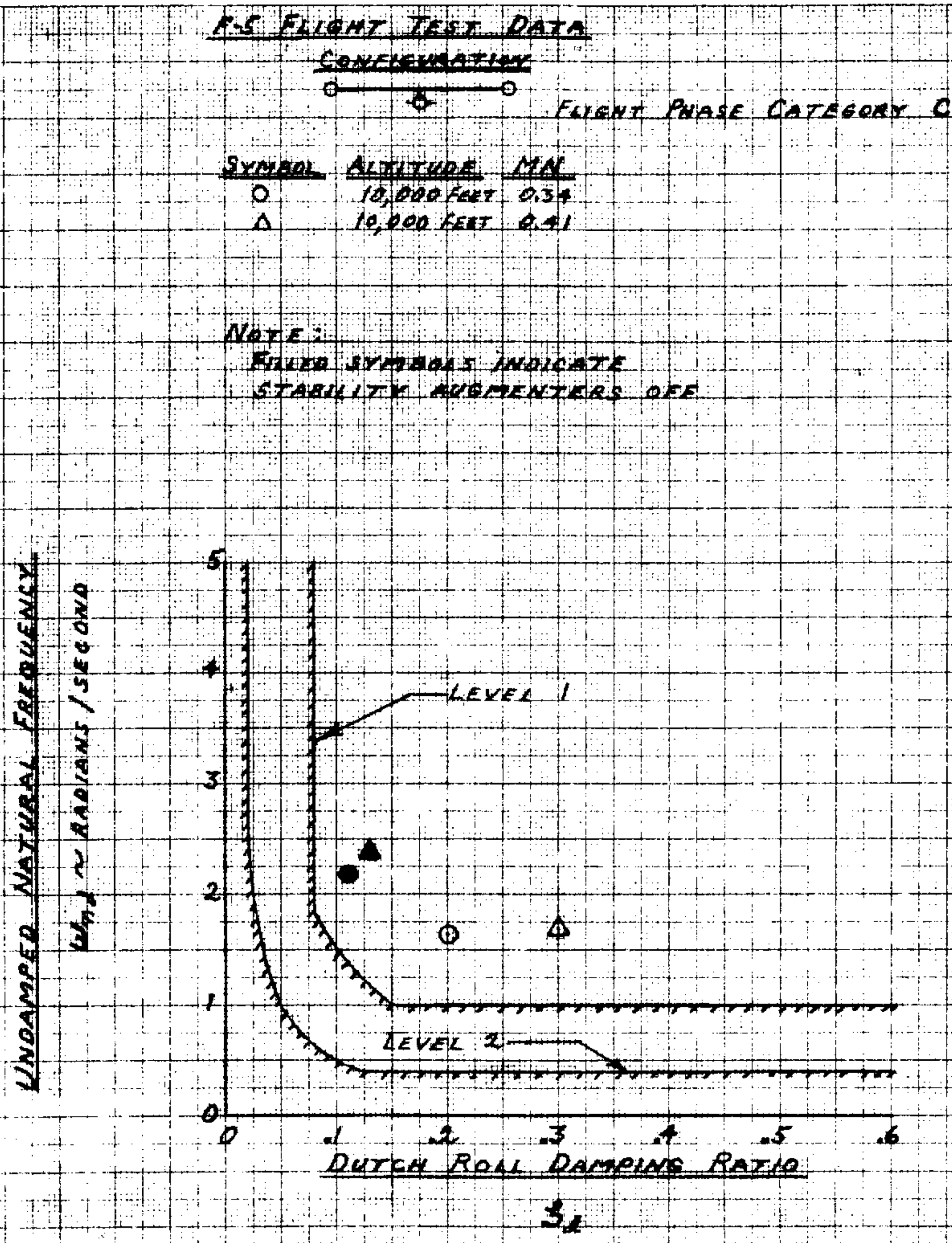
SYMBOL	ALTITUDE	MN
○	10,000 FEET	0.34
△	10,000 FEET	0.40

NOTE:
FILLED SYMBOLS INDICATE
STABILITY AUGMENTATION OFF



LATERAL-DIRECTIONAL OSCILLATIONS

FIGURE 4 (3.3.1.1)



LATERAL-DIRECTIONAL OSCILLATIONS

FIGURE 5 (3.3.1.1)

Requirement

Paragraph 3.3.1.2 Roll mode. The roll-mode time constant, τ_R, shall be no greater than the appropriate value in table VII.

TABLE VII. Maximum Roll-Mode Time Constant

Flight Phase Category	Class	Level		
		1	2	3
A	I, IV	1.0	1.4	
	II, III	1.4	3.0	
B	ALL	1.4	3.0	10
C	I, II-C, IV	1.0	1.4	
	II-L, III	1.4	3.0	

Comparison

Values of the roll mode time constant, τ_R, were calculated in order to compare F-5 characteristics with the requirements of this paragraph. These values were calculated as follows:

$$\tau_R \cong \frac{-4Ix}{\rho S V b^2 Clp} \quad \text{in seconds}$$

where,

Ix is the roll moment of inertia in slugs-ft^2, ρ is the air density in slugs per $feet^3$, S is wing area in $feet^2$, V is true airspeed in ft/sec, b^2 is wing span squared in ft^2 and Clp is roll damping in per radian. Figures 1 (3.3.1.2) and 2 (3.3.1.2) present the τ_R calculated values as a function of Mach number for clean, two-stores, and five-stores configurations with altitudes ranging from 10,000 feet to 30,000 feet.

It is evident from these two figures that the F-5 in a clean configuration (CO Flight Phase) demonstrates a favorable comparison with this paragraph. However, as wing stores are added (GA Flight Phase) and Ix is increased, τ_R increases. Consequently, the F-5 roll mode in the GA Flight Phase disagrees with the paragraph requirements.

Resolution

The τ_R is primarily a function of Ix with all other terms constant for a given Mach number and altitude. When an airplane is designed, the roll capability or the roll mode time constant, τ_R, is designed either for a CO or GA Flight Phase, depending on design mission requirements.

In the case of the F-5, the airplane was primarily designed for a CO Flight Phase, although it also possesses GA capability. Therefore, the roll power design was based primarily on a clean wing; consequently, the GA configuration roll power is insufficient to meet requirements equivalent to those for a CO configuration.

The τ_R is definitely a function of the basic design mission of the airplane. It is not considered technically or economically feasible to design roll power for a GA Flight Phase when the primary mission is the CO Flight Phase. However, the roll mode time constant, τ_R, could be specified as a function of Ix, rendering the design technically qualified and economically feasible, thus producing an airplane that is acceptable for the dual role of CO and GA Flight Phases.

The data presented in Figure 3 (3.3.1.2) exhibit the mean rate of change of τ_R with Ix. For the F-5, this value is 1.3 seconds per 10,000 slugs-ft^2 averaged for a range of Mach numbers and altitudes.

Recommendation

The most critical phases of the ground attack maneuver relevant to τ_R characteristics consist of (1) rolling on target, (2) target acquisition, (3) pickle and (4) breakaway.

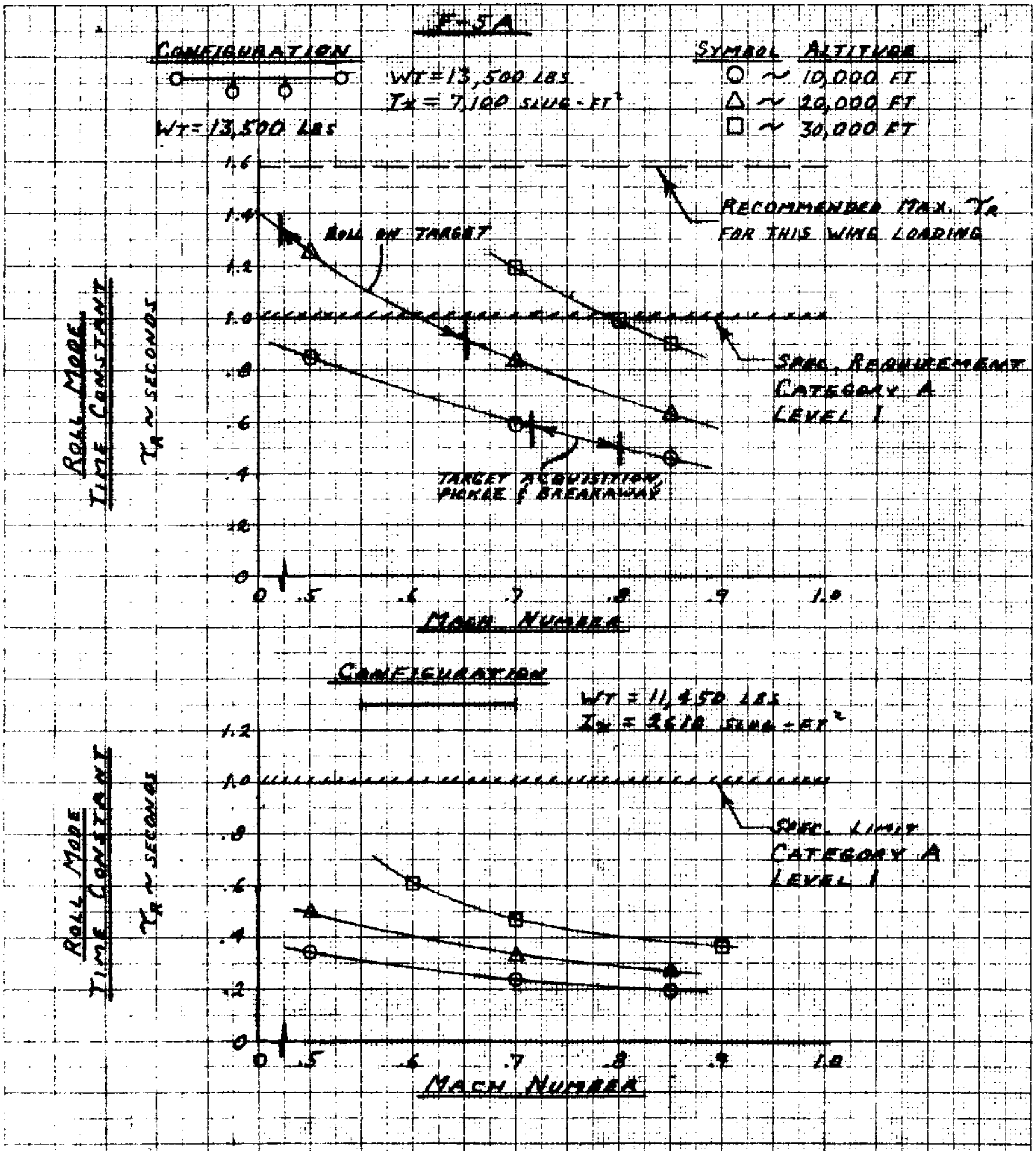
The τ_R of the F-5, shown in Figures 1 (3.3.1.2) and 2 (3.3.1.2) at these phases, exhibit values relatively commensurate with the mean value shown in Figure 3 (3.3.1.2). Because the F-5 excelled in various air-to-ground delivery competitions, such as the "Sparrow Hawk", it is considered that its characteristics as exhibited by the mean value of τ_R in Figure 3 (3.3.1.2), establish a good basis for the following recommendation:

$$\tau_{R(GA)} = \tau_{R(CO)} + \frac{\tau_R}{Ix} \quad (\Delta Ix)$$

where

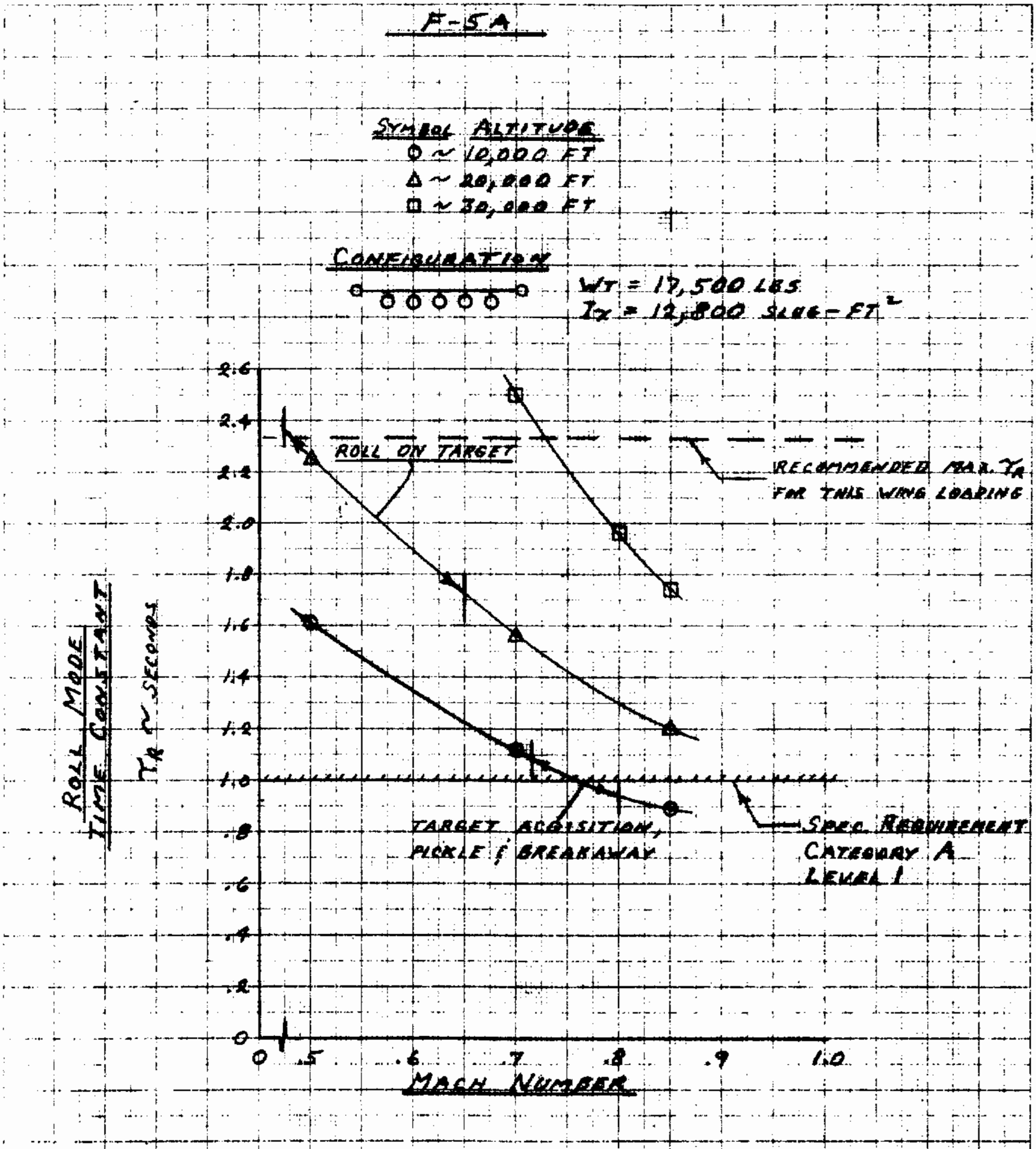
$$\frac{\tau_R}{Ix} = 1.3 \text{ sec}/10,000 \text{ slugs-ft}^2 \text{ (based on F-5 mean value)}$$

$$\Delta Ix = I_{x(GA)} - I_{x(CO)}$$



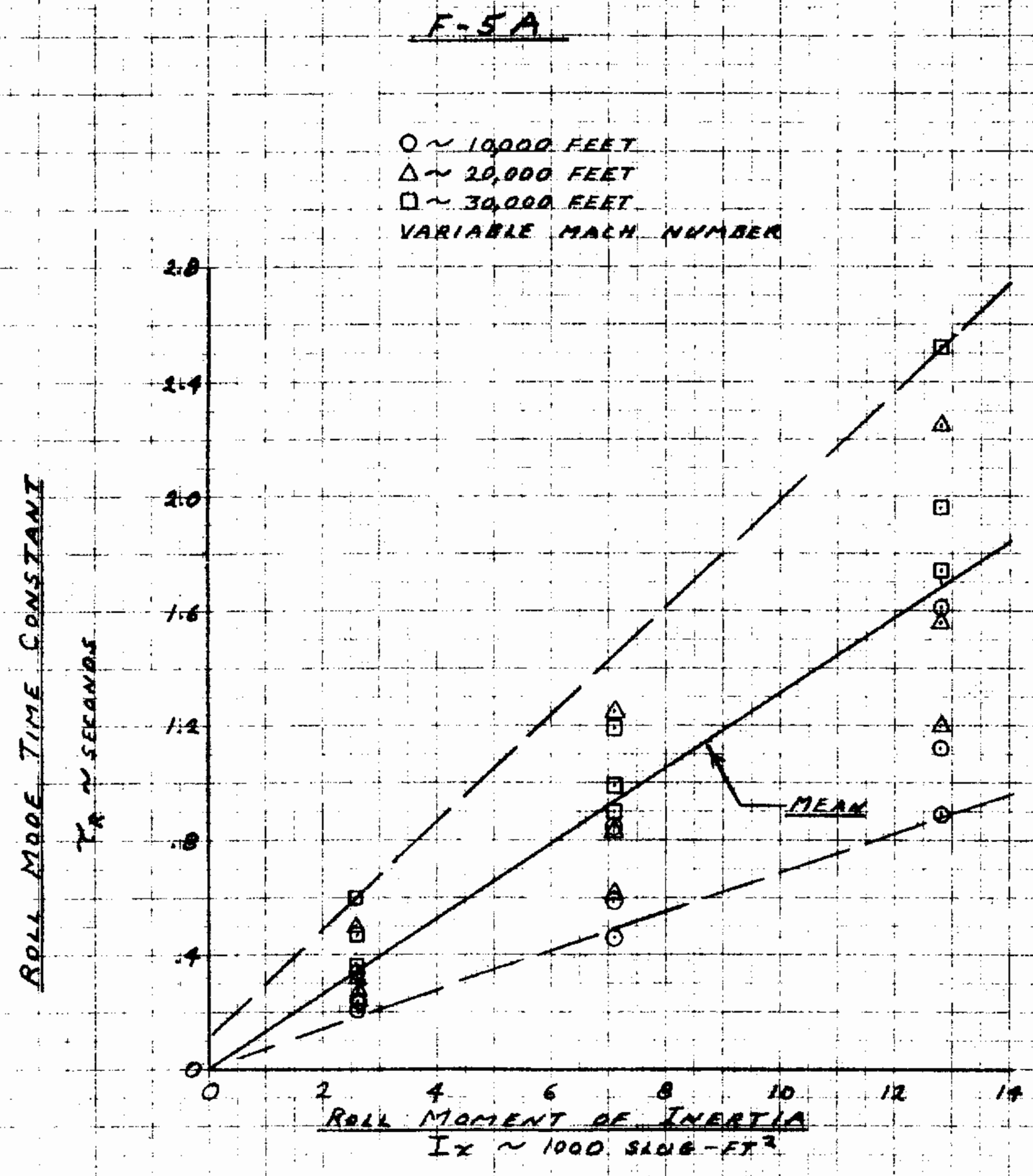
ROLL MODE

FIGURE 1 (3.3.1.2)



ROLL MODE

FIGURE 2 (3.3.1.2)



ROLL MODE

FIGURE 3 (3.3.1.2)

Requirement

Paragraph 3.3.1.3 Spiral stability. The combined effects of spiral stability, flight-control-system characteristics, and trim change with speed shall be such that following a disturbance in bank of up to 20 degrees, the time for the bank angle to double will be greater than the values in table VIII. This requirement shall be met with the airplane trimmed for wings-level, zero-yaw-rate flight with the cockpit controls free.

TABLE VIII. Spiral Stability - Minimum Time to Double Amplitude

Class	Flight Phase Category	Level 1	Level 2	Level 3
I & IV	A	12 sec	12 sec	4 sec
	B & C	20 sec	12 sec	4 sec
II & III	All	20 sec	12 sec	4 sec

Comparison

To validate the requirement specified in this paragraph, an analytical evaluation was performed, using F-5 basic aerodynamic data. The data presented in the form of a bank angle time history, Figure 1 (3.3.1.3), were obtained from a 6-degree-of-freedom computer program which comprises 6 nonlinear differential equations of motion (3 force and 3 moment) solved simultaneously.

Results were obtained with normal load factor initially trimmed at 1.0 g, for 2 airspeeds, to evaluate the effect of C_L on spiral stability.

The F-5 spiral mode is convergent and the convergence rate is decreased as C_L is decreased. Figure 2 (3.3.1.3) presents time to half amplitude as a function of Mach number for three configurations and three altitudes.

Time to half amplitude was obtained from the following approximation which is basically E/D of the lateral-directional quartic equation.

$$\tau_S \simeq -\left[\frac{V}{g} \left(\frac{C_{\ell_p}}{C_{n_r} \frac{C_{\ell_\beta}}{C_{n_\beta}} - C_{\ell_r}} \right) \right]$$

$$T_{\frac{1}{2}} = .693\ \tau_S$$

where,

$V = \sqrt{\frac{2W}{SC_L \rho}}$ true airspeed in feet per sec

$g \sim 32.2$ ft/sec^2

$n_z \sim$ normal load factor, g's

$C_{\ell_p} \sim$ roll damping derivative, per rad

$Cn_r \sim$ yaw damping derivative, per rad

$C_{\ell_\beta} \sim$ effective dihedral, per rad

$Cn_\beta \sim$ directional stability, per rad

$C_{\ell_r} \sim$ cross rotary derivative, roll due to yaw, per rad

$T_{\frac{1}{2}} \sim$ time to half amplitude, sec

The above is an algebraic simplification of the equation given on Page 31 of Reference 7. The accuracy of the results yielded by this approximation is substantiated through corroboration with the results of the complete 6-degree-of-freedom nonlinear solution as exhibited by the half-closed symbol in the upper plot of Figure 2 (3.3.1.3).

The ratio of effective dihedral to directional stability is the primary contributor to the convergence characteristics of the F-5 spiral mode.

Resolution

None

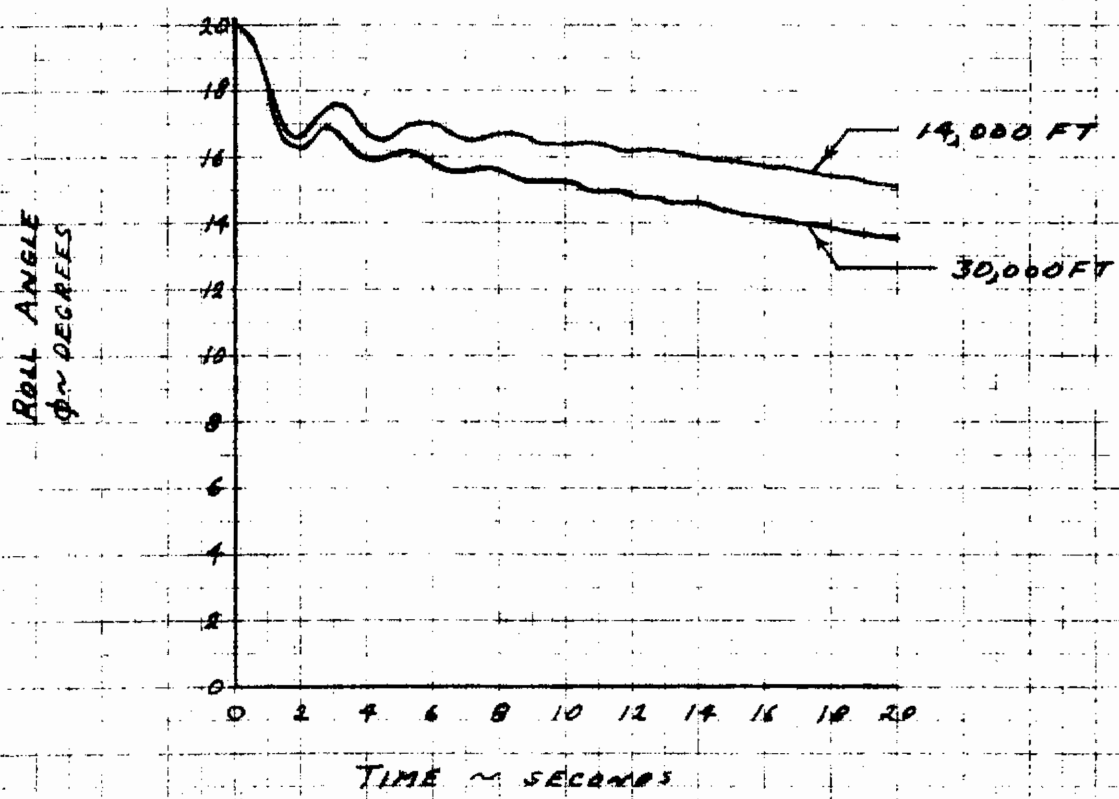
Recommendation

None

F-5A

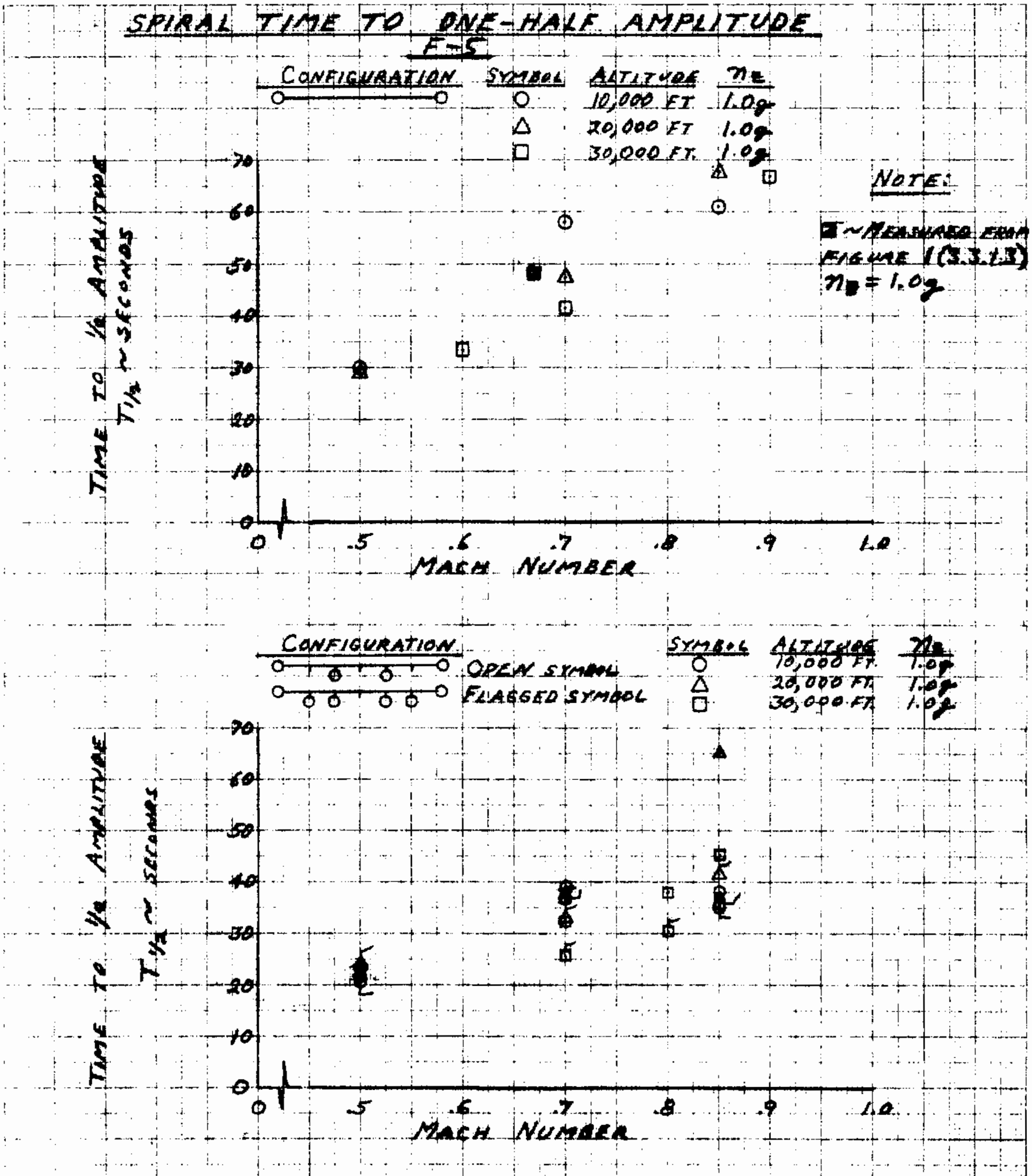
TIME HISTORY OF A COMPUTED SPIRAL STABILITY RUN

CONFIGURATION	ALTITUDE	MACH NO.	WEIGHT	LOAD FACTOR
o———o	30,000 FT.	0.67	13,180 LBS.	1.0g
	14,000 FT.	0.67	13,180 LBS.	1.0g



SPIRAL STABILITY

FIGURE 1 (3.3.1.3)



SPIRAL STABILITY

FIGURE 2 (3.3.1.3)

Requirement

Paragraph 3.3.1.4 Coupled roll-spiral oscillation. A coupled roll-spiral mode will not be permitted.

Comparison

The limiting condition for a coupled roll-spiral mode requires that the roll mode time constant, τ_R, equal the spiral mode time constant, τ_S. This condition produces a coupled roll-spiral mode which is critically damped.

$$\left(s + \frac{1}{\tau_R}\right)\left(s + \frac{1}{\tau_S}\right) = s^2 + 2\zeta\omega_n s + \omega_n^2$$

where,

$$\omega_n^2 = \frac{1}{\tau_R \tau_S}$$

and,

$$\zeta = \frac{(2\zeta\omega_n)}{2\omega_n} = \frac{1}{2}\left(\frac{1}{\tau_R} + \frac{1}{\tau_S}\right)\sqrt{\tau_R \tau_S} = \frac{(\tau_R + \tau_S)}{2\sqrt{\tau_R \tau_S}}$$

If $\tau_R = \tau_S$

then, $\zeta = 1.0$

Roll and spiral mode time constant data were presented in the discussion of paragraphs 3.3.1.2 and 3.3.1.3, respectively. The τ_R data presented in Figures 1 (3.3.1.2) and 2 (3.3.1.2) range from 0.2 to 2.5 seconds. The τ_S can be calculated from Figure 2 (3.3.1.3). The τ_S values range from 29 to 98 seconds. Because the τ_R and τ_S values are never equal for the F-5 airplane, it is concluded that a coupled roll-spiral mode will not be encountered.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.2 Lateral-directional dynamic response characteristics. Lateral-directional dynamic response characteristics are stated in terms of response to atmospheric disturbances and in terms of allowable roll rate and bank oscillations, sideslip excursions, aileron stick or wheel forces, and rudder pedal forces that occur during specified rolling and turning maneuvers. The requirements of 3.3.2.2, 3.3.2.3, and 3.3.2.4 apply for both right and left aileron commands of all magnitudes up to the magnitude required to meet the roll performance requirements of 3.3.4 and 3.3.4.1.

Comparison

The appropriate succeeding paragraphs will present the F-5 lateral-directional characteristics comparison data and discussion.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.2.1 Lateral-directional response to atmospheric disturbances. Although no numerical requirements are specified, the combined effect of ω_{n_d}, ζ_d, τ_R, * p/β, $|\phi/\beta|_d$, gust sensitivity, and flight-control-system nonlinearities shall be such that the airplane will have acceptable response and controllability characteristics in atmospheric disturbances. In particular, the roll acceleration, rate, and displacement responses to side gusts shall be investigated for airplanes with large rolling moment due to sideslip.

Comparison

None

Resolution

None

Recommendation

This paragraph should specify that during the airplane design stage, the contractor shall submit an analysis which indicates that the airplane has acceptable response characteristics in atmospheric disturbances. One such analytical procedure is presented and recommended in paragraph 3.7.5.

Requirement

Paragraph 3.3.2.2 Roll rate oscillations. Following a rudder-pedals-free step aileron control command, the roll rate at the first minimum following the first peak shall be of the same sign and not less than the following percentage of the roll rate at the first peak:

Level	Flight Phase Category	Percent
1	A & C	60
	B	25
2	A & C	25
	B	0

For all Levels, the change in bank angle shall always be in the direction of the aileron control command. The aileron command shall be held fixed until the bank angle has changed at least 90 degrees.

Comparison

Flight test data from roll maneuvers were reduced in tabular form and are presented in Table 1 (3.3.2.2) for comparison of F-5 roll rate oscillation characteristics with the requirements of this paragraph. The data indicate that the ratio of the first minimum roll rate (P_2) to the first peak roll rate (P_1) is well above the 60 percent minimum requirement.

Resolution

None

Recommendation

None

F-5A FLIGHT TEST DATA
LEVEL 1, CATEGORY A

Configuration	Entry n_z ~g's~	Altitude ~Feet~	Mach Number	Roll Rate At First Peak P_1 ~ Deg/Sec	Roll Rate At First Min. P_2 ~ Deg/Sec	(P_2/P_1) X 100 ~%~	Minimum Spec. Req. (P_2/P_1) X 100 ~%~
	1.0	35,000	.61	145	120	82.7	60
	2.5	10,000	.70	180	165	91.8	
	3.5	10,000	.82	185	176	95.1	
	3.0	20,000	.80	190	160	84.2	
	3.0	20,000	.90	195	180	92.4	
	4.0	10,000	.82	160	155	97.0	
	3.2	20,000	.80	160	130	81.2	
	4.0	20,000	.82	160	130	81.2	
	4.0	20,000	.92	175	170	97.1	
	2.7	9,900	.79	180	155	86.2	
	1.0	10,000	.80	190	190	100.0	

ROLL RATE OSCILLATIONS
TABLE 1 (3.3.2.2)

Requirement

Paragraph 3.3.2.2.1 Additional roll rate requirement for small inputs. The value of the parameter p_{OSC}/p_{AV} following a rudder-pedals-free step aileron command shall be within the limits shown on figure 4 for Levels 1 and 2. This requirement applies for step aileron control commands up to the magnitude which causes a 60 degree bank angle change in $1.7T_d$ seconds.

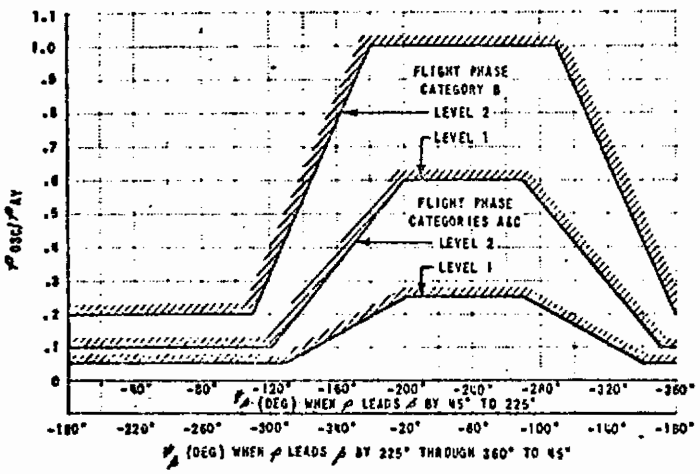


FIGURE 4. Roll Rate Oscillation Limitations

Comparison

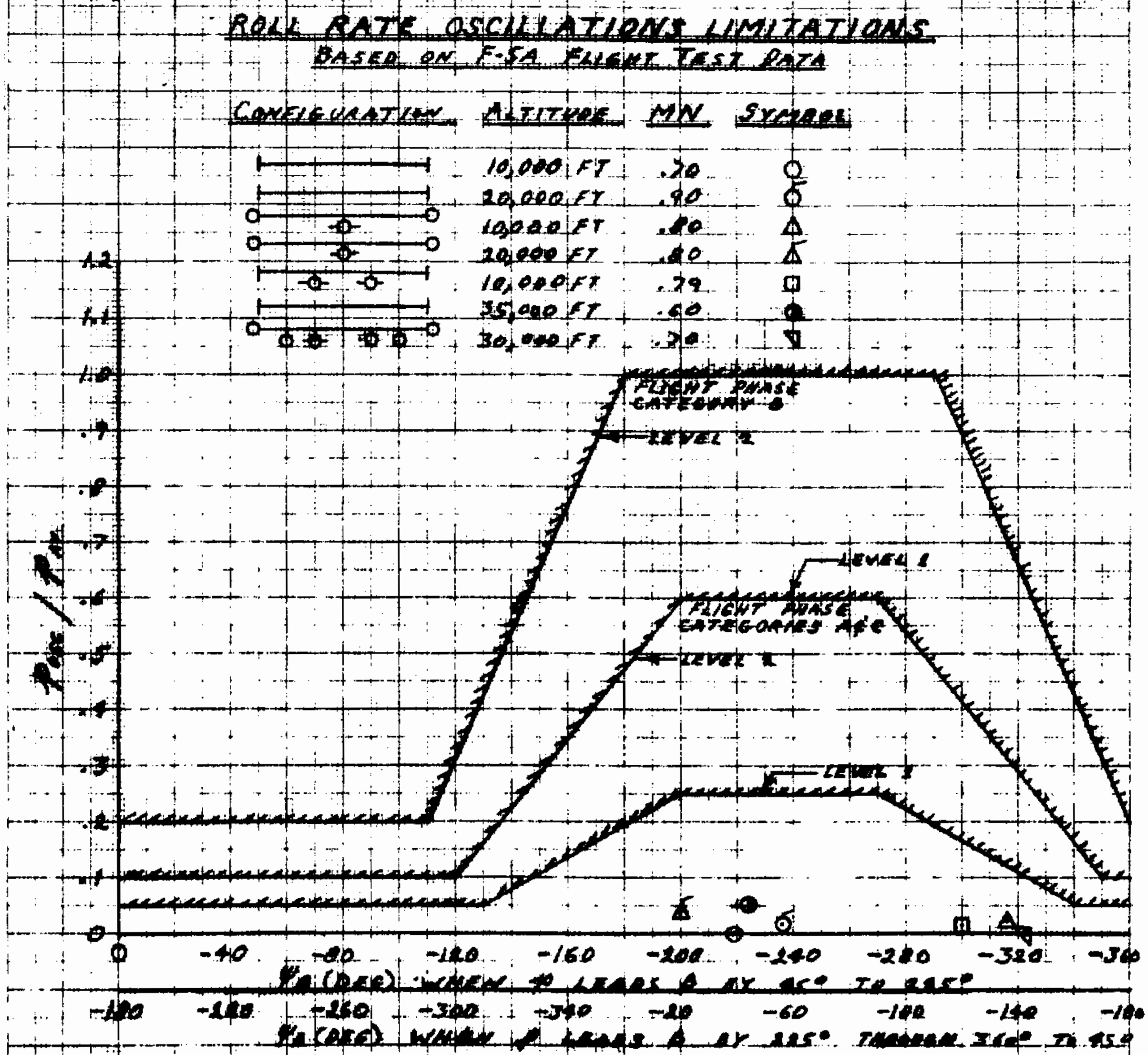
A favorable comparison with the paragraph requirements is exhibited, based on the data presented in Figure 1 (3.3.2.2.1) which show the highly damped F-5 roll mode characteristics.

Resolution

None

Recommendation

None



ADDITIONAL ROLL RATE REQUIREMENT FOR SMALL INPUTS

FIGURE 1 (3.3.2.2.1)

Requirement

Paragraph 3.3.2.3 Bank angle oscillations. The value of the parameter ϕ_{OSC}/ϕ_{AV} following a rudder-pedals-free impulse aileron control command shall be within the limits in figure 5 for Levels 1 and 2. The impulse shall be as abrupt as practical within the strength limits of the pilot and the rate limits of the aileron control system.

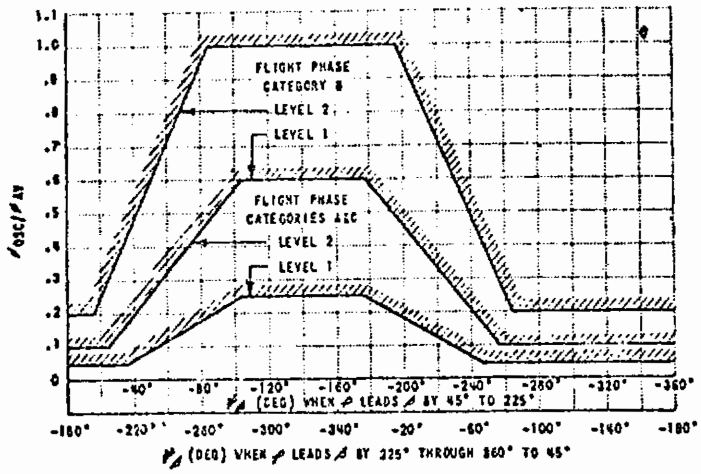


FIGURE 5. Bank Angle Oscillation Limitations

Comparison

MIL-F-8785 specification compliance flight tests for the F-5 were neither required nor conducted for aileron control impulses. The F-5 roll response is so highly damped that the difference in the roll angle oscillatory responses to an aileron step input and to an aileron impulse is considered to be negligible.

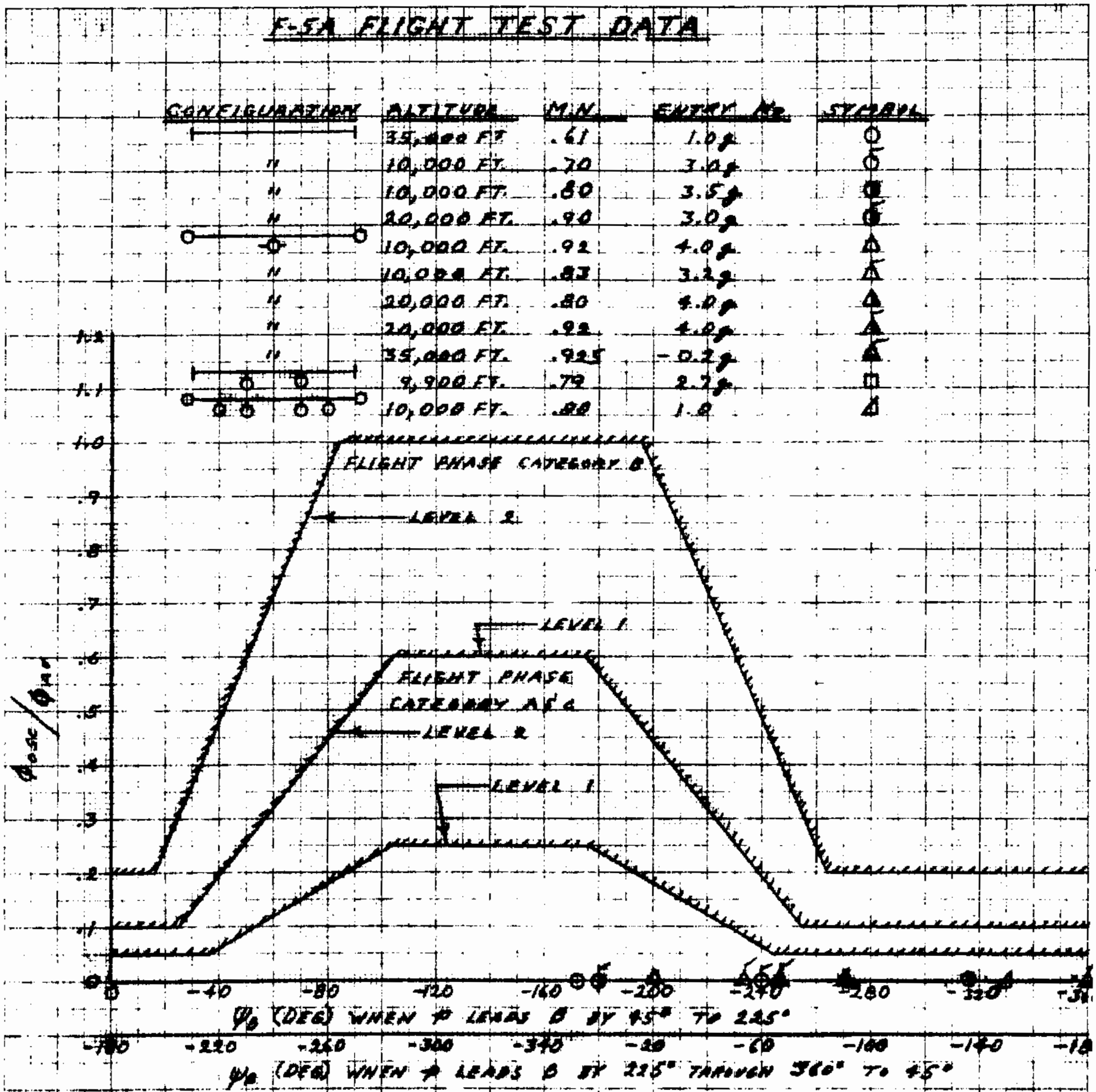
The flight test data presented in Figure 1 (3.3.2.3) are extracted from aileron rolls using maximum roll control system ramp aileron inputs. All the data are well within the limitations imposed by the requirements of this paragraph and a favorable comparison is demonstrated.

Resolution

None

Recommendation

None



BANK ANGLE OSCILLATIONS

FIGURE 1 (3.3.2.3)

Requirement

Paragraph 3.3.2.4 Sideslip excursions. Following a rudder-pedals-free step aileron control command, the ratio of the sideslip increment, $\Delta\beta$, to the parameter k (6.2.6) shall be less than the values specified herein. The aileron command shall be held fixed until the bank angle has changed at least 90 degrees.

Level	Flight Phase Category	Adverse Sideslip (Right roll command causes right sideslip)	Proverse Sideslip (Right roll command causes left sideslip)
1	A	6 degrees	2 degrees
	B & C	10 degrees	3 degrees
2	All	15 degrees	4 degrees

Comparison

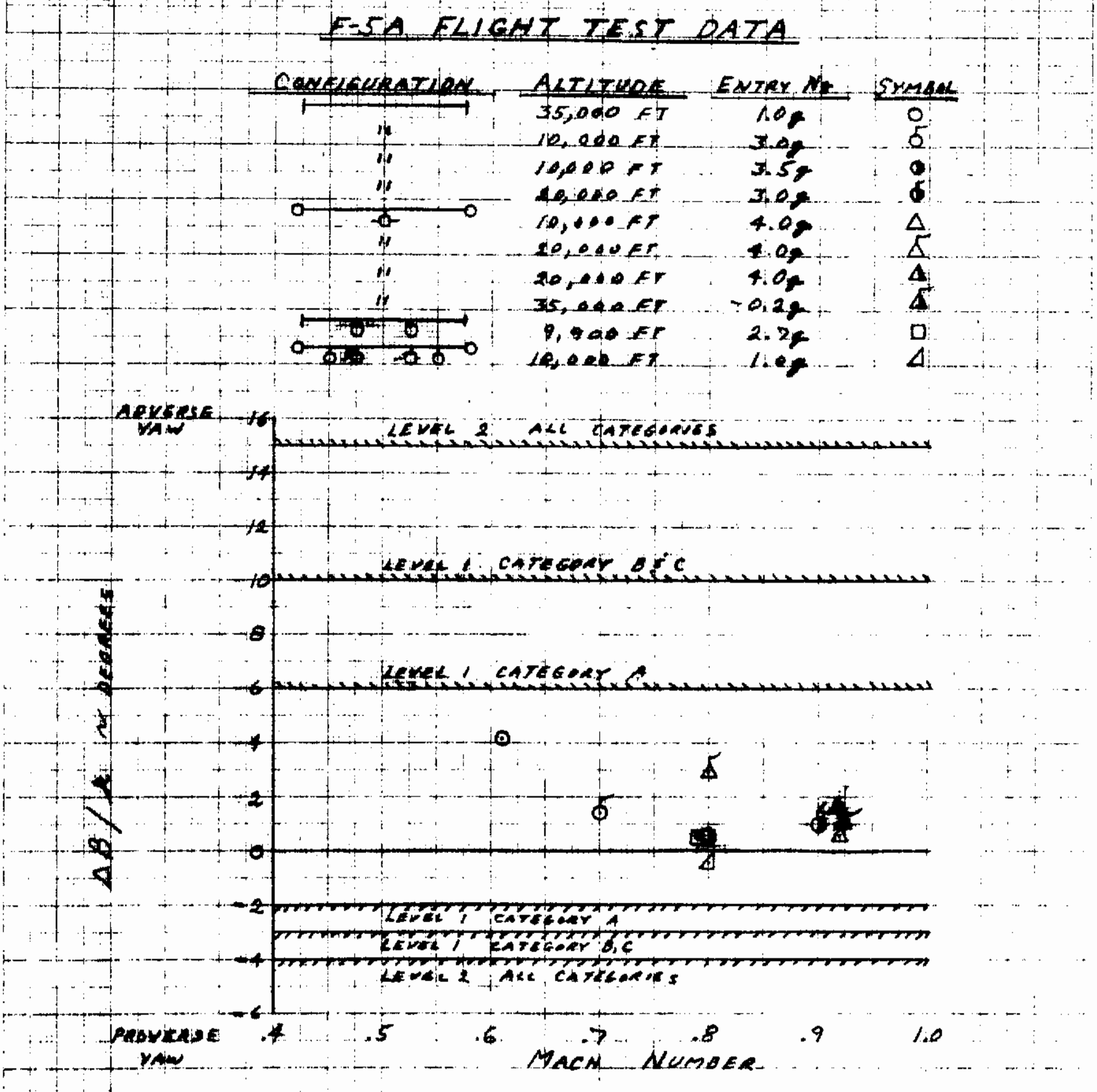
Flight test data from F-5 roll time histories were used to compare sideslip excursion characteristics with the requirements specified in this paragraph. Figure 1 (3.3.2.4) presents these data in the form of $\Delta\beta/k$ (degrees) as a function of Mach number for clean, single store, two store and four store configurations through an altitude range from 10,000 feet to 35,000 feet. Roll entry normal load factors ranged from -0.2 g to 4.0 g. Agreement of F-5 characteristics with the specified requirements has been exhibited.

Resolution

None

Recommendation

None



SIDESLIP EXCURSIONS

FIGURE 1 (3.3.2.4)

Requirement

Paragraph 3.3.2.4.1 Additional sideslip requirement for small inputs. The amount of sideslip following a rudder-pedals-free step aileron control command shall be within the limits shown on figure 6 for Levels 1 and 2. This requirement shall apply for step aileron control commands up to the magnitude which causes a 60-degree bank angle change within T_d or 2 seconds, whichever is longer.

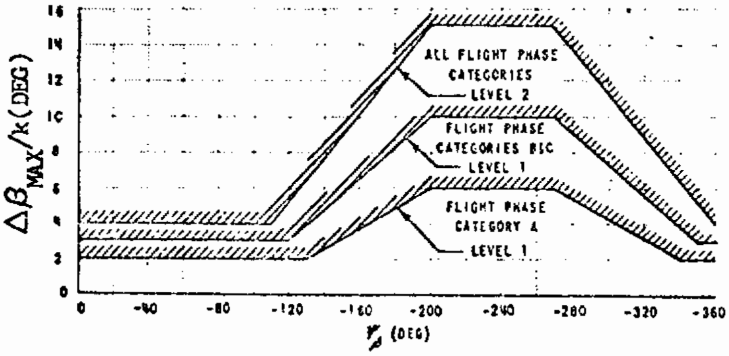


FIGURE 6. Sideslip Excursion Limitations

Comparison

The same F-5 flight test data used to validate 3.3.2.4 were used to validate this paragraph. These data were plotted as $\Delta\beta_{max}/k$ versus ψ_β as shown in Figure 1 (3.3.2.4.1). Agreement with the requirements of this paragraph has been exhibited.

Resolution

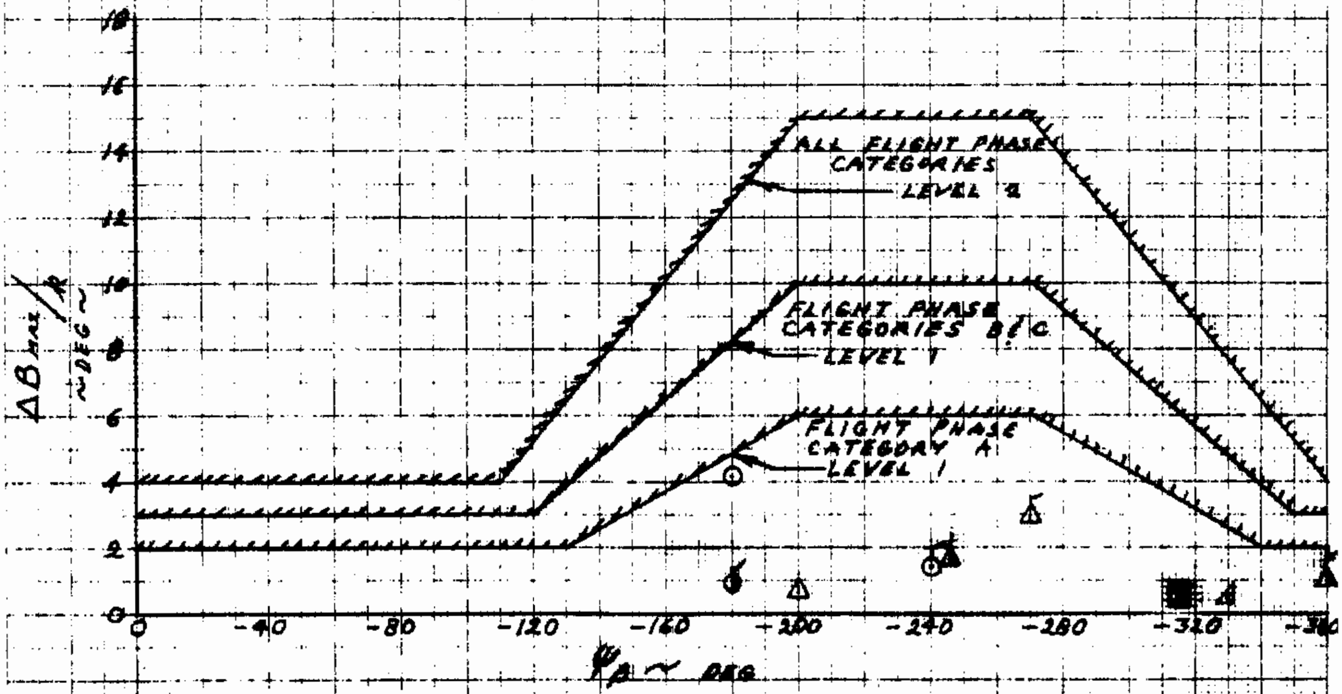
None

Recommendation

None

F-5A FLIGHT TEST DATA

CONFIGURATION	ALTITUDE	MN	ENTRY Nz	SYMBOL
	35,000 FT	.61	1.0 g	○
"	10,000 FT	.70	3.0 g	
"	10,000 FT	.80	3.5 g	
"	20,000 FT	.90	3.0 g	
	10,000 FT	.92	4.0 g	△
"	20,000 FT	.80	4.0 g	
"	20,000 FT	.92	4.0 g	▲
"	35,000 FT	.925	+0.2 g	
	9,900 FT	.79	2.7 g	□
	10,000 FT	.80	1.0	



ADDITIONAL SIDESLIP REQUIREMENT FOR SMALL INPUTS

FIGURE 1 (3.3.2.4.1)

Requirement

Paragraph 3.3.2.5 Control of sideslip in rolls. In the rolling maneuvers described in 3.3.4, but with the rudder pedals used for coordination for all Classes, directional-control effectiveness shall be adequate to maintain zero sideslip with a rudder pedal force not greater than 50 pounds for Class IV airplanes in Flight Phase Category A, Level 1, and 100 pounds for all other combinations of Class, Flight Phase Category and Level.

Comparison

Rolling maneuvers with rudder used for coordination were not conducted in flight test for the F-5. In order to compare the F-5 coordinated rolling characteristics with the requirements of this paragraph, flight test time histories of rolling maneuvers for two extreme external loadings, clean and five-store configurations, were reduced and analyzed for 2 altitudes and a range of Mach numbers to determine the rudder pedal force, Figure 1 (3.3.2.5), required to maintain zero sideslip for coordination.

The following describes the procedure used.

1. Time histories of uncoordinated rolling maneuvers were examined for the maximum sideslip angle, β.

2. The change of rudder position, δ_r, with sideslip angle, β, was determined from flight test data of steady state sideslip maneuvers. Plots of δ_r versus β were constructed.

3. The rudder positions, δ_r, required to maintain zero sideslip angle, β, were obtained from the plots of δ_r versus β, by reading the δ_r required to produce β equivalent to the maximum β of Item 1 above.

4. The rudder pedal forces required to maintain zero sideslip angle during coordinated rolls were then obtained by examining the control system curve of rudder pedal force versus rudder position for the force values commensurate with the rudder positions for Item 3 above.

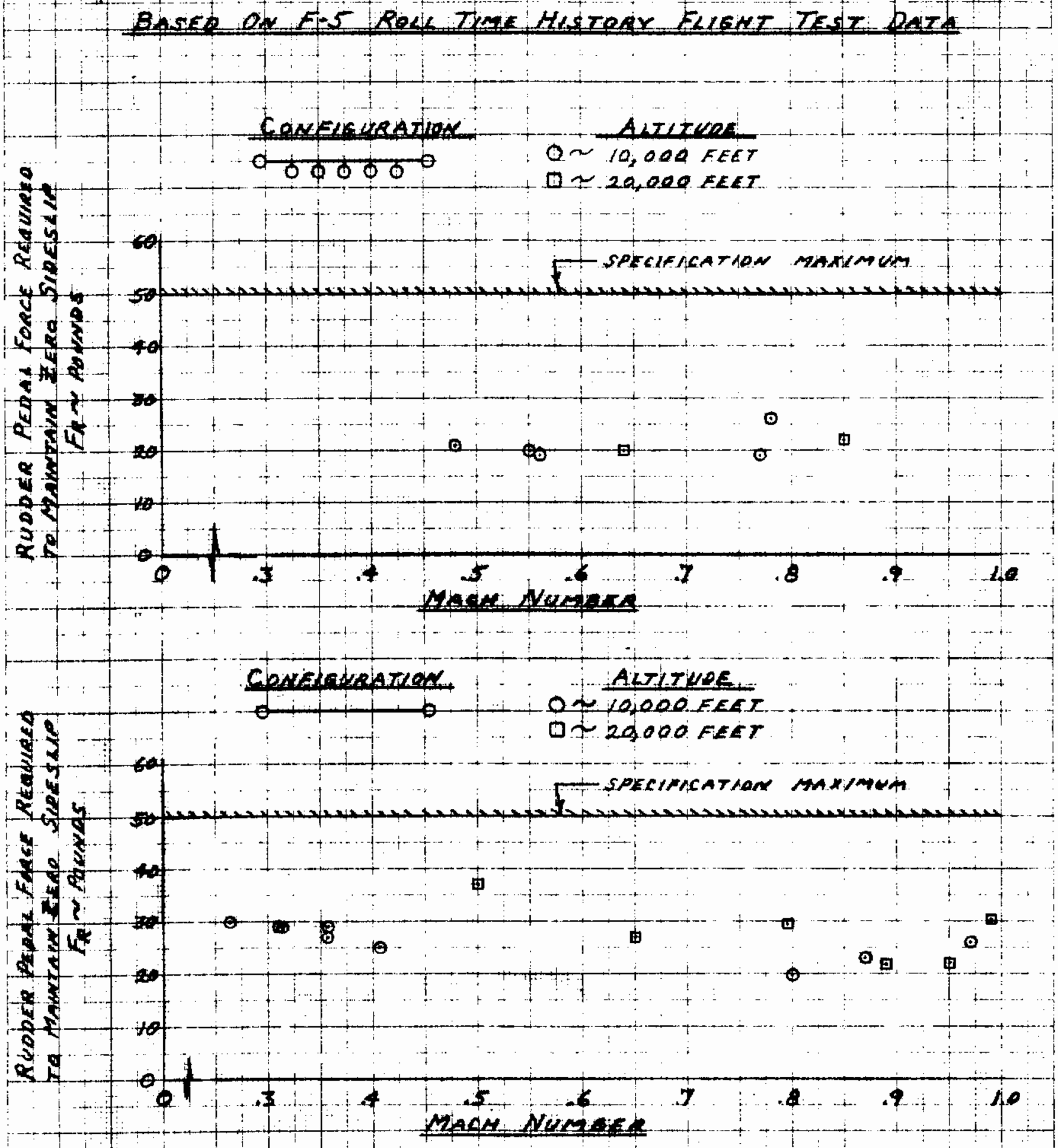
As shown in Figure 1 (3.3.2.5), agreement between the F-5 characteristics and this paragraph is demonstrated.

Resolution

None

Recommendation

None



CONTROL OF SIDESLIP IN ROLLS

FIGURE 1 (3.3.2.5)

Requirement

Paragraph 3.3.2.6 Turn coordination. It shall be possible to maintain steady coordinated turns in either direction, using 60 degrees of bank for Class IV airplanes, 45 degrees of bank for Class I and II airplanes, and 30 degrees of bank for Class III airplanes, with a rudder pedal force not exceeding 40 pounds. It shall be possible to perform steady turns at the same bank angles with rudder pedals free, with an aileron stick force not exceeding 5 pounds or an aileron wheel force not exceeding 10 pounds. These requirements constitute Levels 1 and 2 with the airplane trimmed for wings-level straight flight.

Comparison

The purpose of this requirement is to ensure that a steady coordinated turn can be maintained and that the control forces to overcome the spiral divergence or convergence in constant-bank turns does not exceed the specified values. Since no flight tests were conducted specifically to demonstrate this capability, the following rationale was implemented to reduce available F-5 flight test data and compare with this paragraph.

1. Examine wind-up turn data for the aileron stick force required to obtain 60-degree bank angle. Rudder pedals were free.

2. The 60-degree bank angle was achieved at a rate much faster than the convergence rate of the F-5 spiral mode. Consequently, it is considered that the rate of aileron control is more than sufficient to maintain constant bank angle required to arrest the spiral convergence.

3. Assume that the aileron control force to obtain the 60-degree bank angle is more than ample to maintain and hold the bank angle.

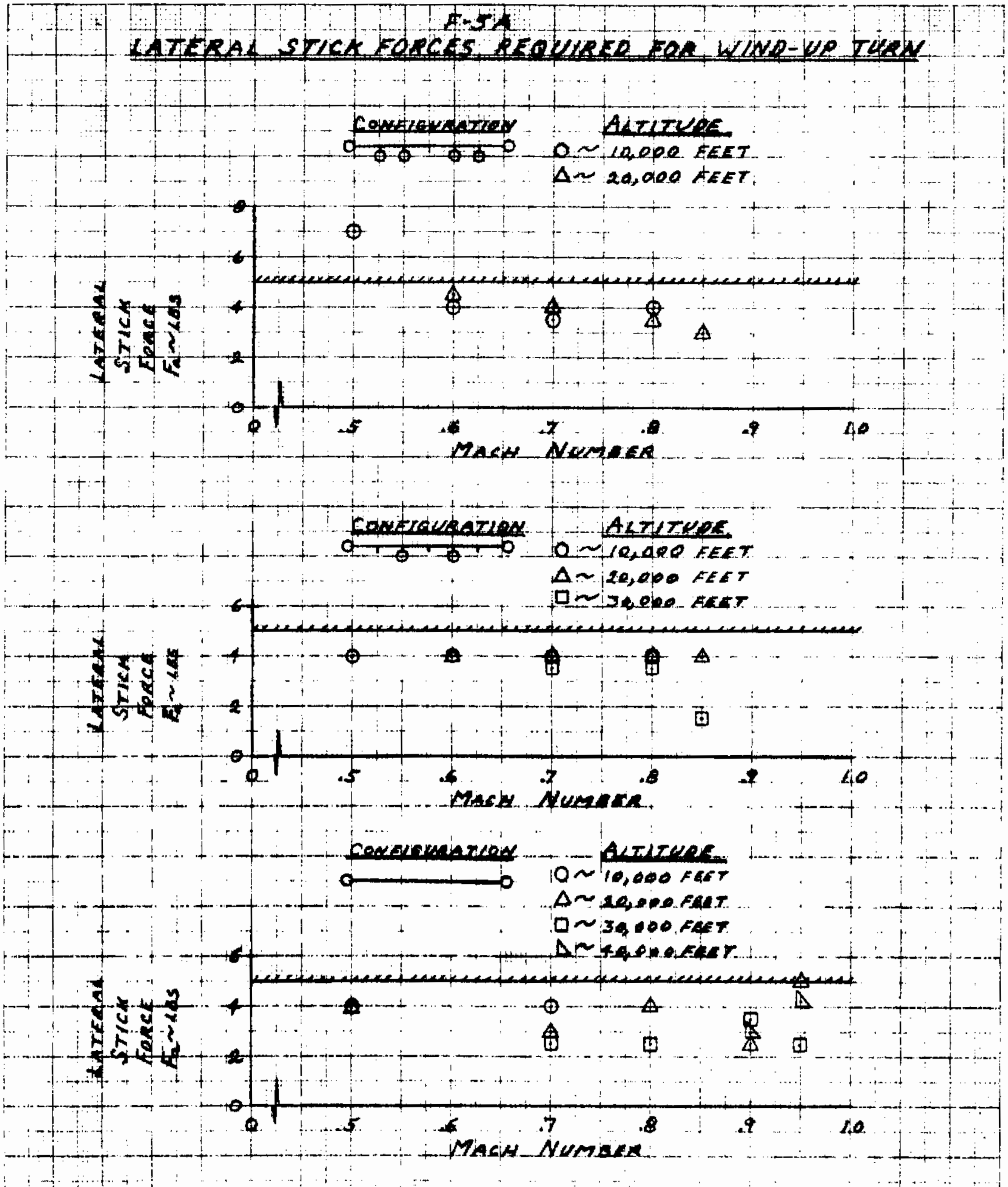
Based on the above and the flight test data presented in Figure 1 (3.3.2.6), it is considered that the F-5 turn coordination characteristics basically agreed with the requirements of this paragraph.

Resolution

None

Recommendation

None



TURN COORDINATION

FIGURE 1 (3.3.2.6)

Requirement

Paragraph 3.3.3 Pilot-induced oscillations. There shall be no tendency for sustained or uncontrollable lateral-directional oscillations resulting from efforts of the pilot to control the airplane.

Comparison

No tendency for sustained or uncontrollable lateral-directional oscillations resulting from efforts of the pilot to control the airplane has been reported either by test pilots during flight testing or by service pilots in the field. Although specific testing for this has not been conducted, it is considered, based on service life of the F-5, that no tendency for lateral P.I.O. exists.

Resolution

A qualitative requirement to ensure no tendency for pilot-induced oscillations is not considered sufficient as a specification.

Recommendation

Although little or no work has been conducted in the industry to establish criteria to evaluate lateral P.I.O. quantitatively, it is considered essential that a method be derived and quantitative evaluation be specified. It is recommended that research work be conducted in this field to establish a quantitative specification. This work should parallel the work for paragraph 3.2.2.3.

Requirement

Paragraph 3.3.4 Roll control effectiveness. Roll performance in terms of bank angle change in a given time, ϕ_t, is specified in table IX and in 3.3.4.1. Aileron control commands shall be initiated from zero roll rate in the form of abrupt inputs, with time measured from the initiation of control-force application . Rudder pedals shall remain free for Class IV airplanes for Level 1, and for all carrier-based airplanes in Category C Flight Phases for Levels 1 and 2; but otherwise, rudder pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip that augments roll rate) if rudder pedal inputs are simple, easily coordinated with aileron-control inputs, and consistent with piloting techniques for the airplane Class and mission. Roll control shall be sufficiently effective to balance the airplane in roll throughout the Service Flight Envelope in the atmospheric disturbances of 3.7.3 and 3.7.4.

Comparison

Flight test data were used to compare F-5 characteristics with the requirements of this paragraph. These data are presented in Figures 1 (3.3.4) for Level 1, Flight Phase Category A and B, and in Figure 2 (3.3.4) for Level 1, Flight Phase Category C.

The data in Figure 1 (3.3.4) were reduced from rolling maneuvers with roll entry normal load factors, n_z, ranging from 1.0 g to 5.0 g. The single store configuration demonstrates agreement with the paragraph requirements. However, as additional stores are added, increasing the rolling moment of inertia, I_x, disagreement between the F-5 and the requirements of this paragraph becomes apparent.

Figure 2 (3.3.4) presents data for six different store loadings for Flight Phases L, PA and TO. Partial disagreement is exhibited in that certain store loading configurations with high rolling moments of inertia do not meet the requirements of Table IX.

Significant pilot comments regarding the magnitude of acceptable roll performance with respect to time to bank were not obtained during F-5 flight testing. Nevertheless, a more thorough discussion of the effect of rolling moment of inertia on roll performance is presented in Paragraph 3.3.4.1.2.

Resolution

None

Recommendation

None

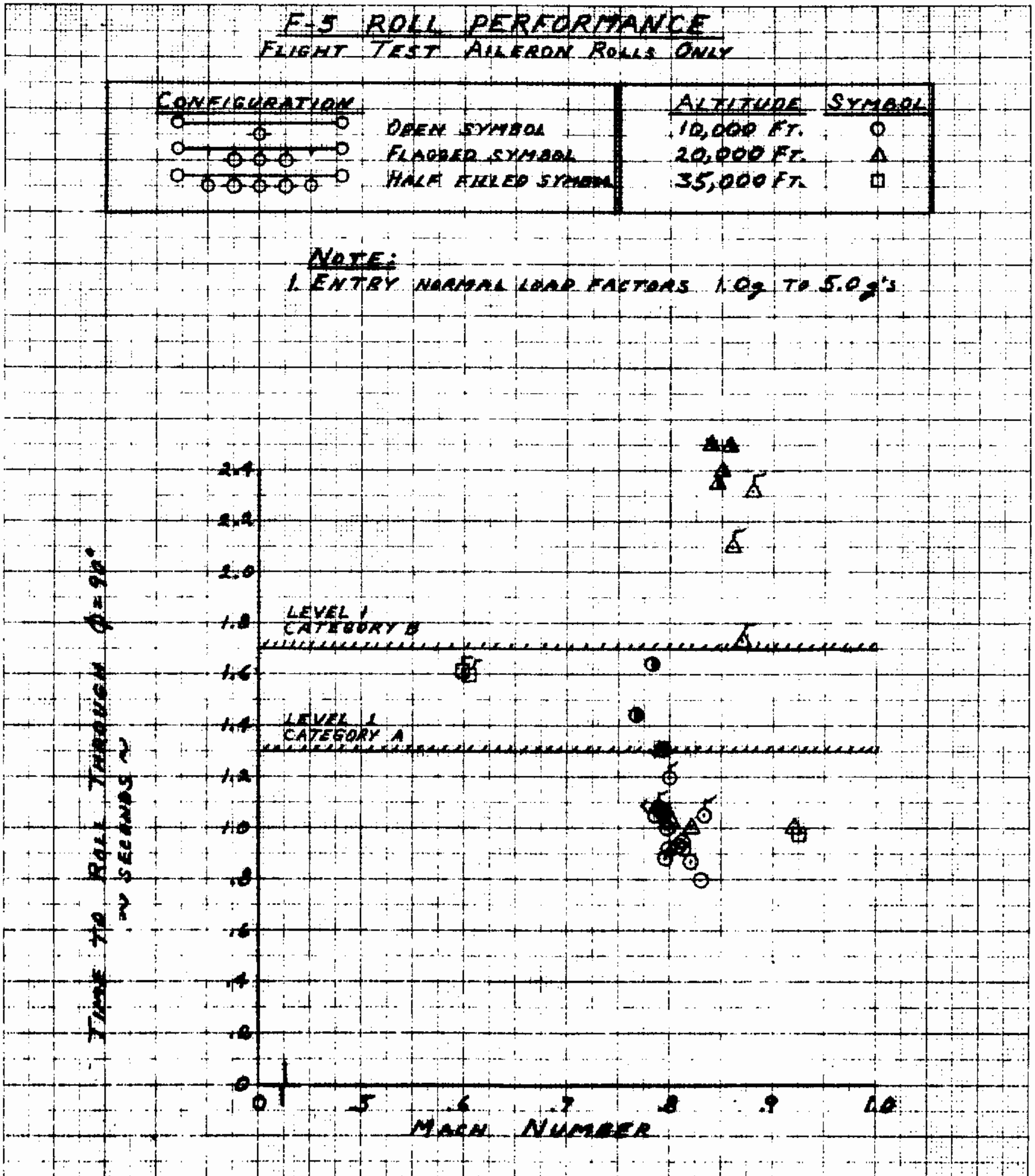
TABLE IX. Roll Performance Requirements

Class	Flight Phase Category	Level 1	Level 2**	Level 3
I	A	ϕ_t = 60° in 1.3 sec	ϕ_t = 60° in 1.7 sec	ϕ_t = 60° in 2.6 sec
	B	ϕ_t = 60° in 1.7 sec	ϕ_t = 60° in 2.5 sec	ϕ_t = 60° in 3.4 sec
	C†	ϕ_t = 30° in 1.3 sec	ϕ_t = 30° in 1.8 sec	ϕ_t = 30° in 2.6 sec
II	A	ϕ_t = 45° in 1.4 sec	ϕ_t = 45° in 1.9 sec	ϕ_t = 45° in 2.8 sec
II	B	ϕ_t = 45° in 1.9 sec	ϕ_t = 45° in 2.8 sec	ϕ_t = 45° in 3.8 sec
II-L	C†	ϕ_t = 30° in 1.8 sec	ϕ_t = 30° in 2.5 sec	ϕ_t = 30° in 3.6 sec
II-C	C†	ϕ_t = 25° in 1.0 sec	ϕ_t = 25° in 1.5 sec	ϕ_t = 25° in 2.0 sec
III	A	ϕ_t = 30° in 1.5 sec	ϕ_t = 30° in 2.0 sec	ϕ_t = 30° in 3.0 sec
	B	ϕ_t = 30° in 2.0 sec	ϕ_t = 30° in 3.0 sec	ϕ_t = 30° in 4.0 sec
	C†	ϕ_t = 30° in 2.5 sec	ϕ_t = 30° in 3.2 sec	ϕ_t = 30° in 4.0 sec
IV	A*	ϕ_t = 90° in 1.3 sec	ϕ_t = 90° in 1.7 sec	ϕ_t = 90° in 2.6 sec
	B	ϕ_t = 90° in 1.7 sec	ϕ_t = 90° in 2.5 sec	ϕ_t = 90° in 3.4 sec
	C†	ϕ_t = 30° in 1.0 sec	ϕ_t = 30° in 1.3 sec	ϕ_t = 30° in 2.0 sec

*Except as the requirements are modified in 3.3.4.1

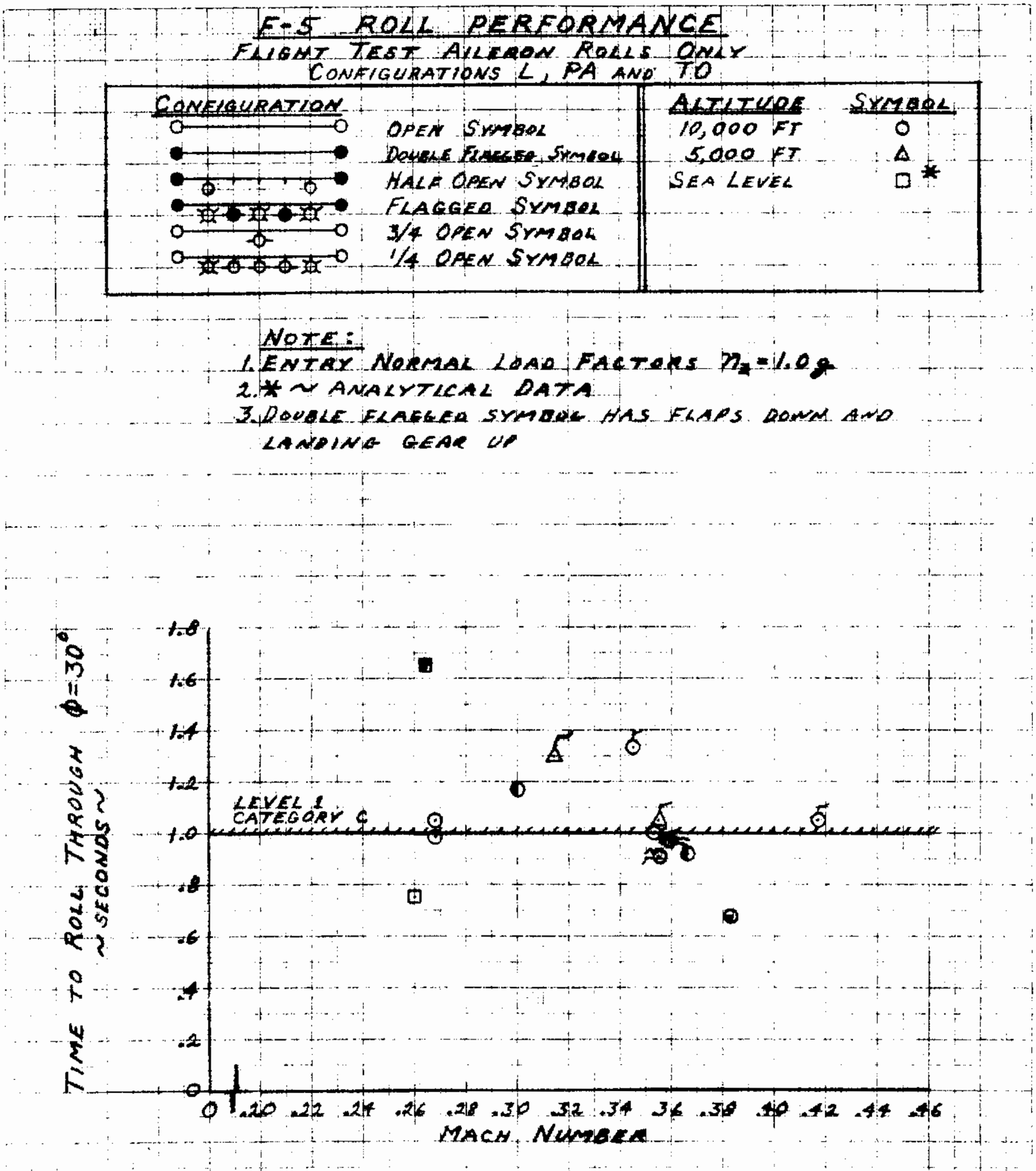
†For takeoff, the required bank angle can be reduced proportional to the ratio of the maximum rolling moment of inertia for the maximum authorized landing weight to the rolling moment of inertia at takeoff, but the Level 1 requirement shall not be reduced below the listed value for Level 3.

**At altitudes below 20,000 feet at the high-speed boundary of the Service Flight Envelope, the Level 3 requirements may be substituted for the Level 2 requirements with all systems functioning normally.



ROLL CONTROL EFFECTIVENESS

FIGURE 1 (3.3.4)



ROLL CONTROL EFFECTIVENESS

FIGURE 2 (3.3.4)

Requirement

Paragraph 3.3.4.1 Roll performance for Class IV airplanes. Additional or alternate roll performance requirements are specified for Class IV airplanes in 3.3.4.1.1 through 3.3.4.1.4. These requirements take precedence over table IX.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.4.1.1 Air-to-air combat. For Class IV airplanes in Flight Phase CO, the roll performance requirements are:

	Time to roll through	
	90 degrees	360 degrees
a. Level 1- - - - - - - - - -	1.0 second	2.8 seconds
b. Level 2- - - - - - - - - -	1.3 seconds	3.3 seconds
c. Level 3- - - - - - - - - -	1.7 seconds	4.4 seconds

Comparison

F-5 flight test data from aileron rolls with entry normal load factors of 1.0 g to 5.0 g are presented in Figure 1 (3.3.4.1.1) for comparison of the air-to-air combat configuration roll performance with the requirements of this paragraph. Agreement between the F-5 characteristics and the requirements of this paragraph prevails at Mach numbers above 0.6. At Mach numbers below 0.6, the roll performance decreases to such a level that disagreement is exhibited with the requirements.

Resolution

The decay in roll performance abounds near the low-speed operational envelope boundary where the angle of attack rises sharply, thus decreasing aileron control effectiveness. The lack of pilot comments relative to the maximum time to bank that is acceptable in this region of the operational envelope makes it difficult to specify an acceptable magnitude of roll performance with ailerons only or with ailerons and rudder to augment roll.

Relaxation of the restriction regarding the application of the rudder to augment roll performance may be in order. At high angles of attack where the aileron effectiveness decreases, the rudder roll effectiveness increases by generating favorable sideslip that augments roll performance as a result of the effective dihedral.

The F-5 has exhibited favorable roll performance in air combat situations where both the rudder and ailerons were used at low speed and at high angles of attack. Its tractability even at extreme angles of attack and sideslip, allow pilots to use the rudder without fear of losing control.

Recommendation

In general, for Class IV airplanes, roll performance decreases as the speed decreases and the angle of attack increases, especially where rolls are performed with ailerons only. It is recommended that research work be conducted to investigate the feasibility of relaxing the restriction to use the rudder to augment roll performance for Flight Phase CO by establishing the following:

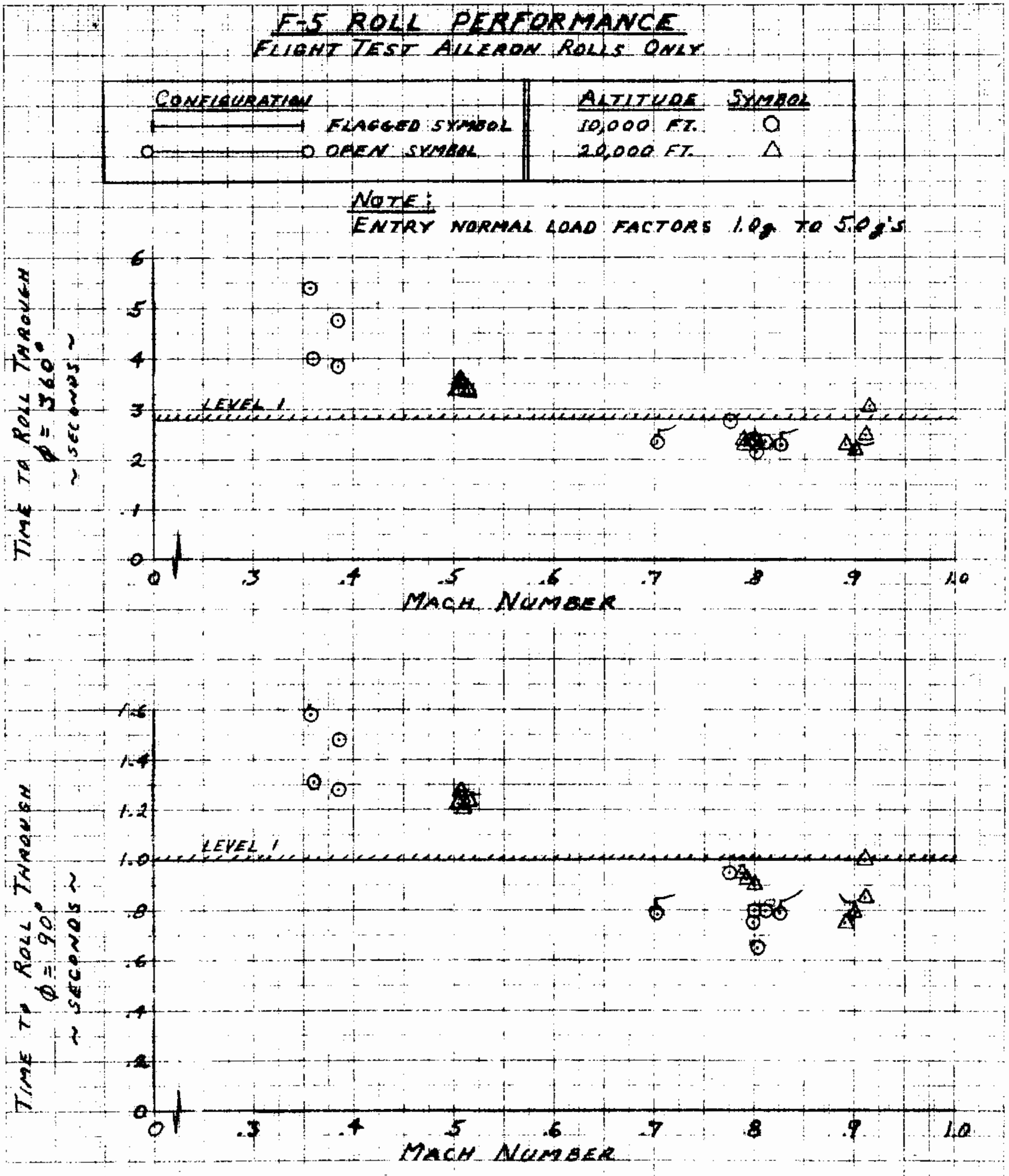
1. Roll performance improvement

2. Acceptability by pilots of aileron plus rudder application

3. Ease and compatibility of aileron plus rudder application

4. Proness to generating dangerous flight conditions such as spins or uncontrollable rolls

5. Minimum acceptable roll performance and evaluation of body axis roll angle versus stability axis roll angle



AIR-TO-AIR COMBAT

FIGURE F(3.3.4.1.1)

Requirement

Paragraph 3.3.4.1.2 Ground attack with external stores. The roll performance requirements for Class IV airplanes in Flight Phase GA with large complements of external stores may be relaxed from those specified in table IX, subject to approval by the procuring activity. For any external loading specified in the contract, however, the roll performance shall be not less than:

a. Level 1 - - - - - - - - - - - - 90 degrees in 1.7 seconds

b. Level 2 - - - - - - - - - - - - 90 degrees in 2.6 seconds

c. Level 3 - - - - - - - - - - - - 90 degrees in 3.4 seconds

For any asymmetric loading specified in the contract, aileron control power shall be sufficient, to hold the wings level at the maximum load factors specified in 3.2.3.2 in the atmospheric disturbances of 3.7.3.

Comparison

Flight test data from aileron rolls are presented in Figure 1 (3.3.4.1.2) for three different GA configurations, three altitudes, various Mach numbers and entry normal load factors throughout the Operational Flight Envelope. Agreement with the paragraph requirements is generally exhibited with the following exceptions:

1. High speed region near the Operational Flight Envelope boundary when high roll entry normal load factors were utilized.

2. Low speed region at high angles of attack.

Resolution

The resolution for this paragraph is an extension of that for paragraph 3.3.4.1.1 inasmuch as the aileron effectiveness decay is the result of increasing the angle of attack. In paragraph 3.3.4.1.1, the angle of attack increase is due mainly to low speed conditions.

In this paragraph, the increase in angle of attack is not only due to the low speed conditions, but also due to the increase in normal load factors at the high speed conditions. Consequently, aileron control effectiveness decay at the low speed and at the high speed conditions, as shown in Figure 1 (3.3.4.1.2), is caused respectively by decrease in dynamic pressure and increase in normal load factor.

Application of rudder may prove to be very effective in providing acceptable roll performance improvement not only at the low speed region, but also at the high load factors of the high speed region.

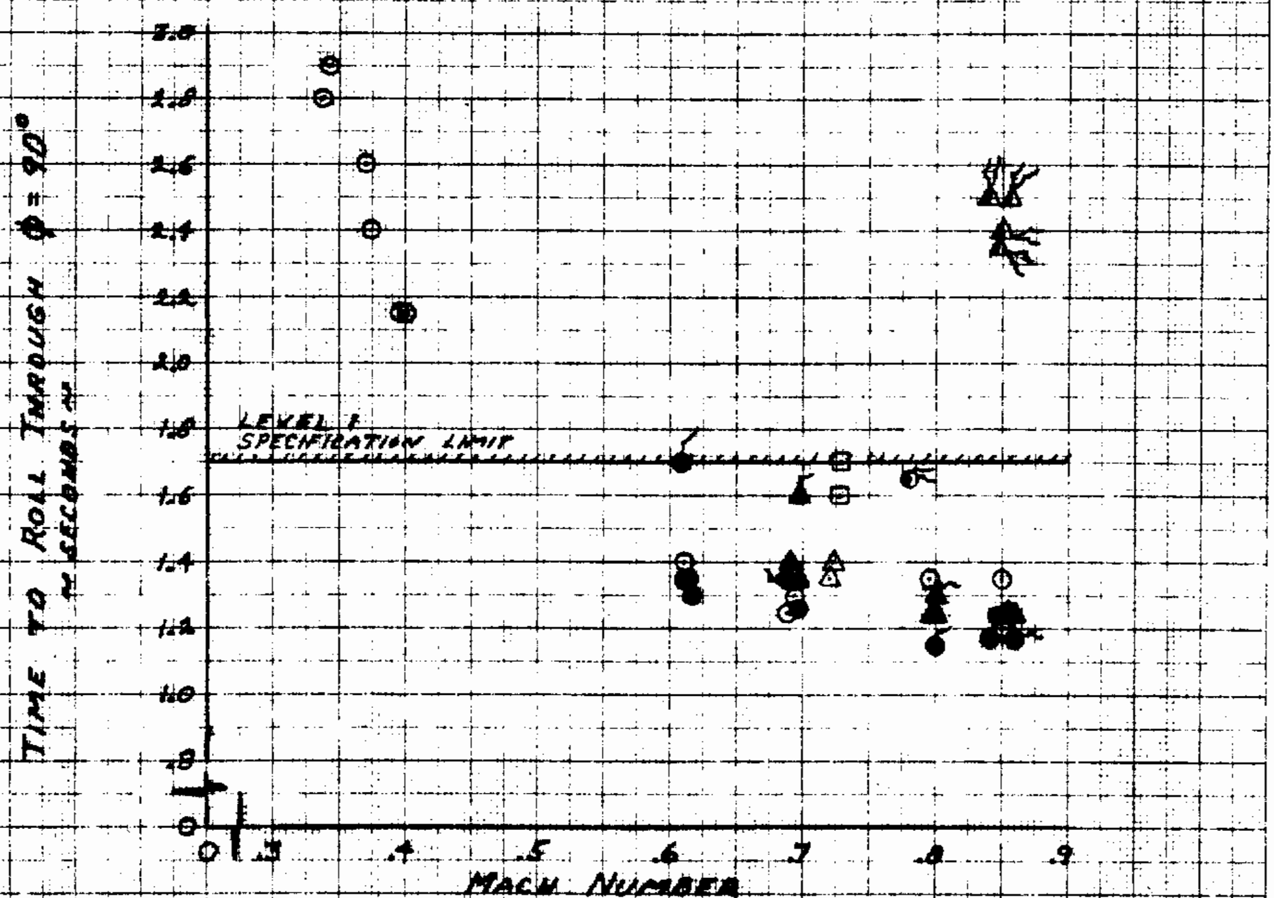
Recommendation

It is recommended that the research work outlined for paragraph 3.3.4.1.1 be conducted in conjunction with the resolution of this paragraph and extended to include high load factors at high speeds.

F-5 ROLL PERFORMANCE
FLIGHT TEST AILERON ROLLS ONLY

CONFIGURATION		ALTITUDE	SYMBOL
MK 7, BLU 1B	OPEN SYMBOL	10,000 FT.	O
LAU 3/A, BLU 1B	SOLID SYMBOL	20,000 FT.	△
BLU 1B, TANKS, BLU 1B	HALF SOLID SYMBOL	30,000 FT	□

NOTE:
1. UNFLAGGED SYMBOL ~ 1g ROLL ENTRY
2. FLAGGED SYMBOL ~ 3g ROLL ENTRY
3. DOUBLE FLAGGED SYMBOL ~ GREATER THAN 4.0g ROLL ENTRY
4. STABILITY AUGMENTERS OFF



GROUND ATTACK WITH EXTERNAL STORES

FIGURE 1 (3.3.4.1.2)

Requirement

Paragraph 3.3.4.1.3 Roll rate characteristics for ground attack. Class IV airplanes in Flight Phase GA shall be able to roll through 180 degrees in not more than twice the time to roll through 90 degrees. This requirement specifies Level 1 with the rudder pedals remaining free throughout the maneuver and Levels 2 and 3 with the rudder pedals employed to reduce sideslip in the manner described in 3.3.4.

Comparison

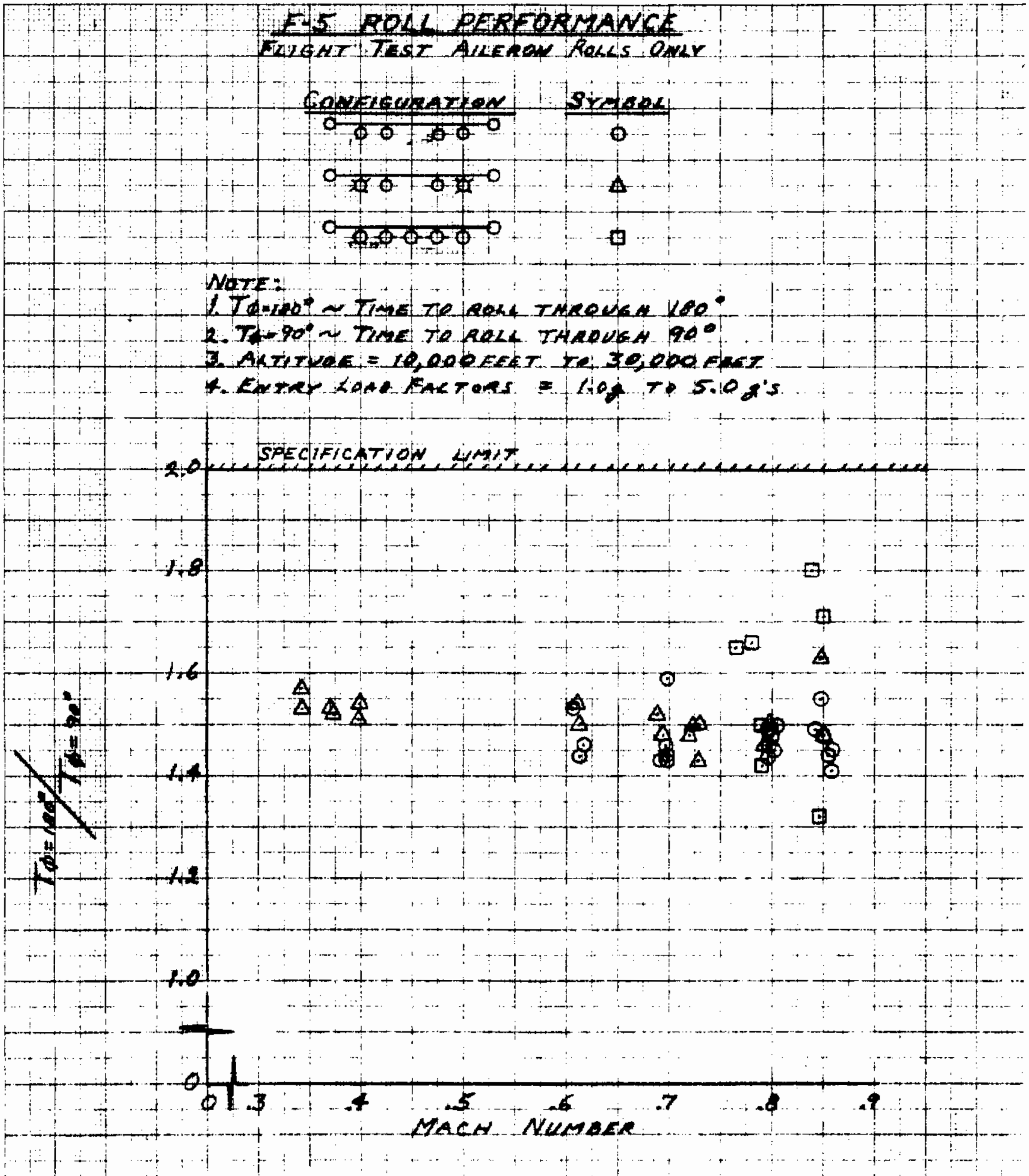
Roll rate characteristics were evaluated for the F-5 in three ground attack configurations. These characteristics are presented in Figure 1 (3.3.4.1.3) as the ratio of time to roll through 180 degrees to time to roll through 90 degrees plotted as a function of Mach number. Roll entry normal load factors ranged from 1.0 g to 5.0 g and altitude ranged from 10,000 to 30,000 feet. The F-5 compares favorably with the paragraph requirements.

Resolution

None

Recommendation

None



ROLL RATE CHARACTERISTICS FOR GROUND ATTACK

FIGURE 1 (3.3.4.1.3)

Requirement

Paragraph 3.3.4.1.4 Roll response. Stick-controlled Class IV airplanes in Category A Flight Phases shall have a roll response to aileron control force not greater than 15 degrees in 1 second per pound for Level 1, and not greater than 25 degrees in 1 second per pound for Level 2. For Category C Flight Phases, the roll sensitivity shall be not greater than 7.5 degrees in 1 second per pound for Level 1, and not greater than 12.5 degrees in 1 second per pound for Level 2. In case of conflict between the requirements of 3.3.4.1.4 and 3.3.4.2, the requirements of 3.3.4.1.4 shall govern.

Comparison

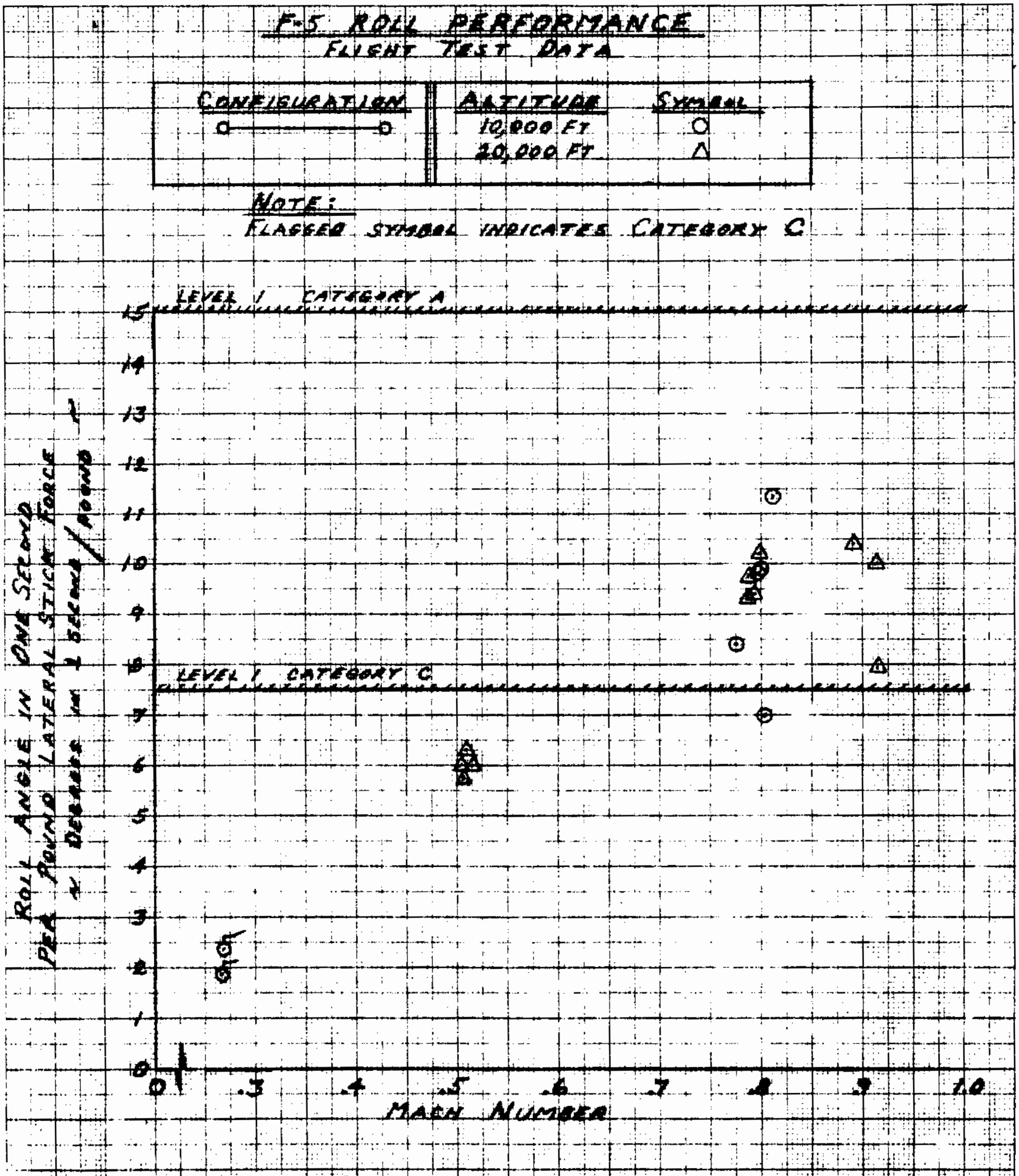
Flight test data from aileron rolls are presented in Figure 1 (3.3.4.1.4). Roll angle in one second per pound of aileron control force is plotted as a function of Mach number for the clean configuration which is the most critical. It represents the maximum roll angle in one second per pound of aileron control force. All data fall below the limits imposed by this paragraph, exhibiting full agreement.

Resolution

None

Recommendation

None



ROLL RESPONSE

FIGURE 1 (3.3.4.1.4)

Requirement

Paragraph 3.3.4.2 Aileron control forces. The stick or wheel force required to obtain the rolling performance specified in 3.3.4 and 3.3.4.1 shall be neither greater than the maximum in table X nor less than the breakout force plus:

a. Level 1 - - - - - - - - one-fourth the values in table X

b. Level 2 - - - - - - - - one-eighth the values in table X

c. Level 3 - - - - - - - - zero

TABLE X. Maximum Aileron Control Force

Level	Class	Flight Phase Category	Maximum Stick Force (lb)	Maximum Wheel Force (lb)
1	I, II-C, IV	A, B	20	40
		C	20	20
	II-L, III	A, B	25	50
		C	25	25
2	I, II-C, IV	A, B	30	60
		C	20	20
	II-L, III	A, B	30	60
		C	30	30
3	All	All	35	70

Comparison

The F-5 aileron control force characteristics are presented in Figure 1 (3.3.4.2). In order to obtain the rolling performance specified in paragraph 3.3.4 and paragraph 3.3.4.1, aileron deflections of 32.5 degrees were required for Flight Phase Category A and 60 degrees for Flight Phase Category C. The aileron control forces commensurate with the deflections required are well within the bounds specified by this paragraph.

Resolution

None

Recommendation

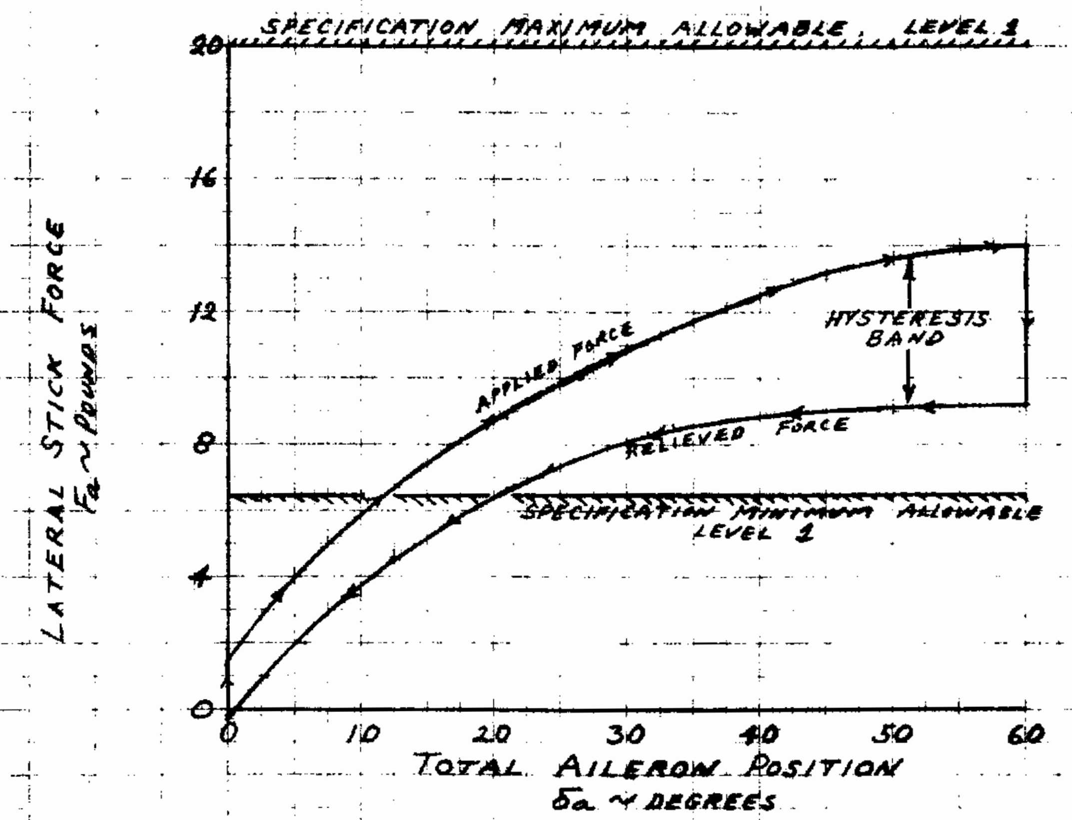
None

E-5
AILERON CONTROL FORCE CHARACTERISTICS

NOTE:

1. $\delta_a = 32.5°$ IS THE AILERON DEFLECTION REQUIRED FOR FLIGHT PHASE CATEGORY A TO OBTAIN ROLL PERFORMANCE OF PARA. 3.3.4.

2. $\delta_a = 60.0°$ IS THE AILERON DEFLECTION REQUIRED FOR FLIGHT PHASE CATEGORY C TO OBTAIN ROLL PERFORMANCE OF PARA. 3.3.4.



AILERON CONTROL FORCES

FIGURE 1 (3.3.4.2)

Requirement

Paragraph 3.3.4.3 Linearity of roll response. There shall be no objectionable nonlinearities in the variation of rolling response with aileron control deflection or force. Sensitivity or sluggishness in response to small aileron control deflections or forces shall be avoided.

Comparison

Comparison of F-5 characteristics with the requirements of this paragraph is based on analytical results obtained from an analog computer using F-5 basic aerodynamic data inputs. Data runs were conducted for aileron deflections of 10, 20 and 32.5 degrees, with flight test data substantiating the results of the analytical data obtained with 32.5 degrees aileron deflection. Figure 1 (3.3.4.3) presents these results. Slight nonlinearities are evident in the rolling response to both aileron deflections and aileron control force inputs. However, these nonlinearities are not generally recognized as objectionable.

Resolution

Although the F-5 exhibits little nonlinearity, it is not possible in general to establish the magnitude of nonlinearities that would be considered objectionable based on the qualitative requirements of this paragraph. No reference or quantitative values are specified and without this benefit, it would be difficult to recognize or prove objectionable nonlinearities; especially, for borderline cases, where there are local slope variations of roll angle with aileron deflection or control force that are not necessarily slope reversals.

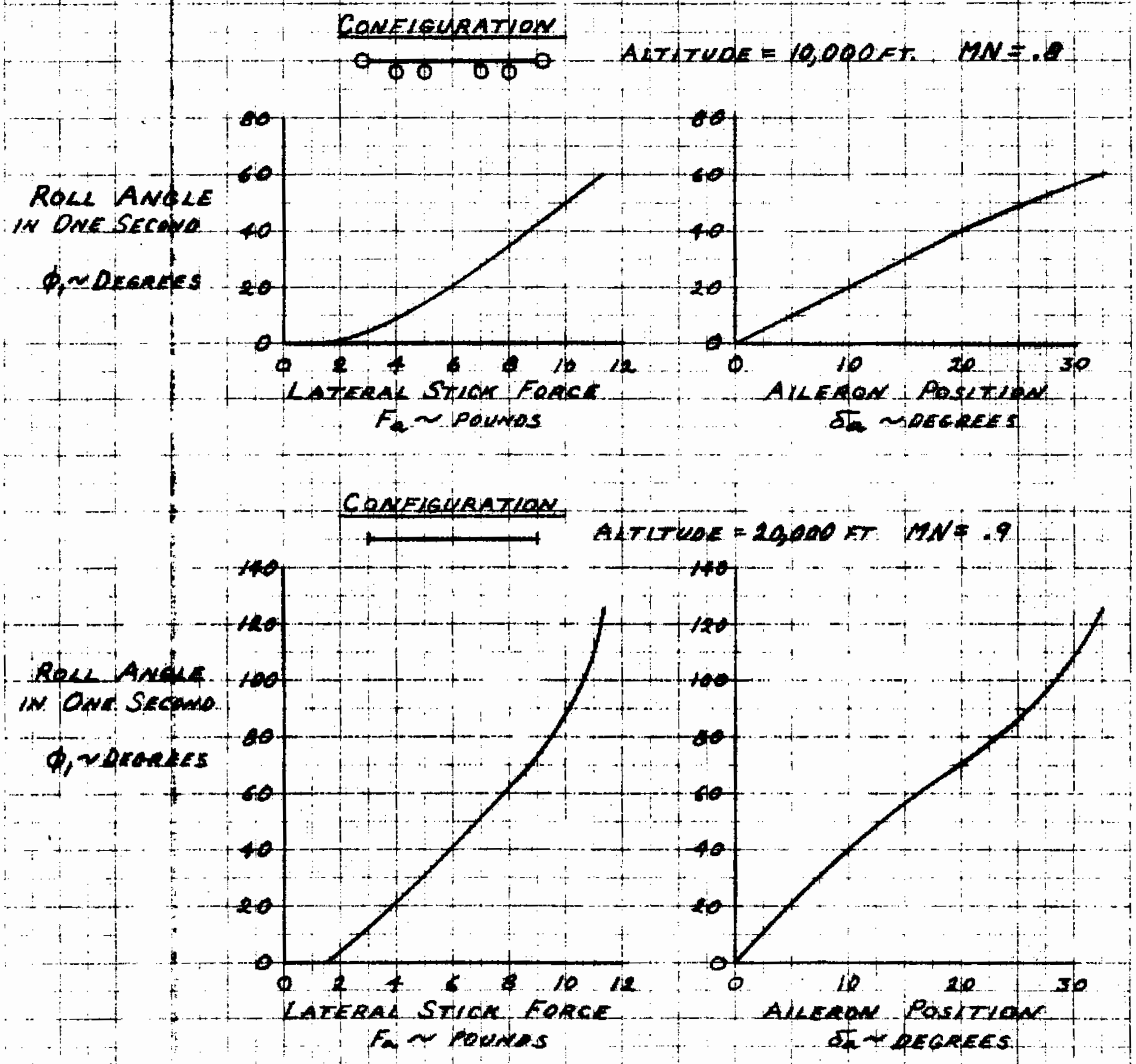
Work should be conducted on simulators or in flight with programmed nonlinearities to establish levels of objections based on pilot evaluations. As a result, quantitative requirements can be specified in terms of maximum variations allowed in local slope from a mean value.

Recommendation

It is recommended that research be conducted to investigate the approach discussed in the resolution, in order to achieve and specify quantitative requirements for this paragraph.

F-5

ANALYTICAL DATA SUBSTANTIATED BY FLIGHT TEST



LINEARITY OF ROLL RESPONSE

FIGURE 1 (3.3.4.3)

Requirement

Paragraph 3.3.4.4 Wheel control throw. For airplanes with wheel controllers, the wheel throw necessary to meet the roll performance requirements specified in 3.3.4 shall not exceed 60 degrees in either direction. For completely mechanical systems, the requirement may be relaxed to 80 degrees.

Comparison

None. The F-5 airplane has no wheel control.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.4.5 Rudder-pedal-induced-rolls. For Levels 1 and 2, it shall be possible to raise a wing by use of rudder pedal alone, with right rudder pedal force required for right rolls and left rudder pedal force required for left rolls. For Level 1, with the aileron control free, it shall be possible to produce a roll rate of 3 degrees per second with an incremental rudder pedal force of 50 pounds or less. The specified roll rate shall be attainable from coordinated turns at up to ±30 degrees bank angle with the airplane trimmed for wings-level, zero-yaw-rate flight.

Comparison

This paragraph ensures that positive effective dihedral (negative C_{ℓ_β}) exists for all classes of airplanes by demonstrating that right rolls will result with right rudder pedal force and conversely.

Figure 1 (3.3.4.5) presents data extracted from F-5 flight tests of rudder-induced rolls of 90 and 360 degrees and of rudder pulse maneuvers. The maximum roll rates obtained for rudder pedal forces of 50 pounds or less ranged from 39 to 120 degrees per second for various Mach numbers, altitudes and configurations. Right rudder deflections and right rudder pedal forces produced right rolls. Conversely, left rudder deflections and left rudder pedal forces produced left rolls. Interconnect between the rudder and aileron is non-existant for the F-5 airplane.

Although the F-5 roll rate response to rudder far exceeds the minimum requirements, agreement with the intent of this paragraph is exhibited.

Resolution

None

Recommendation

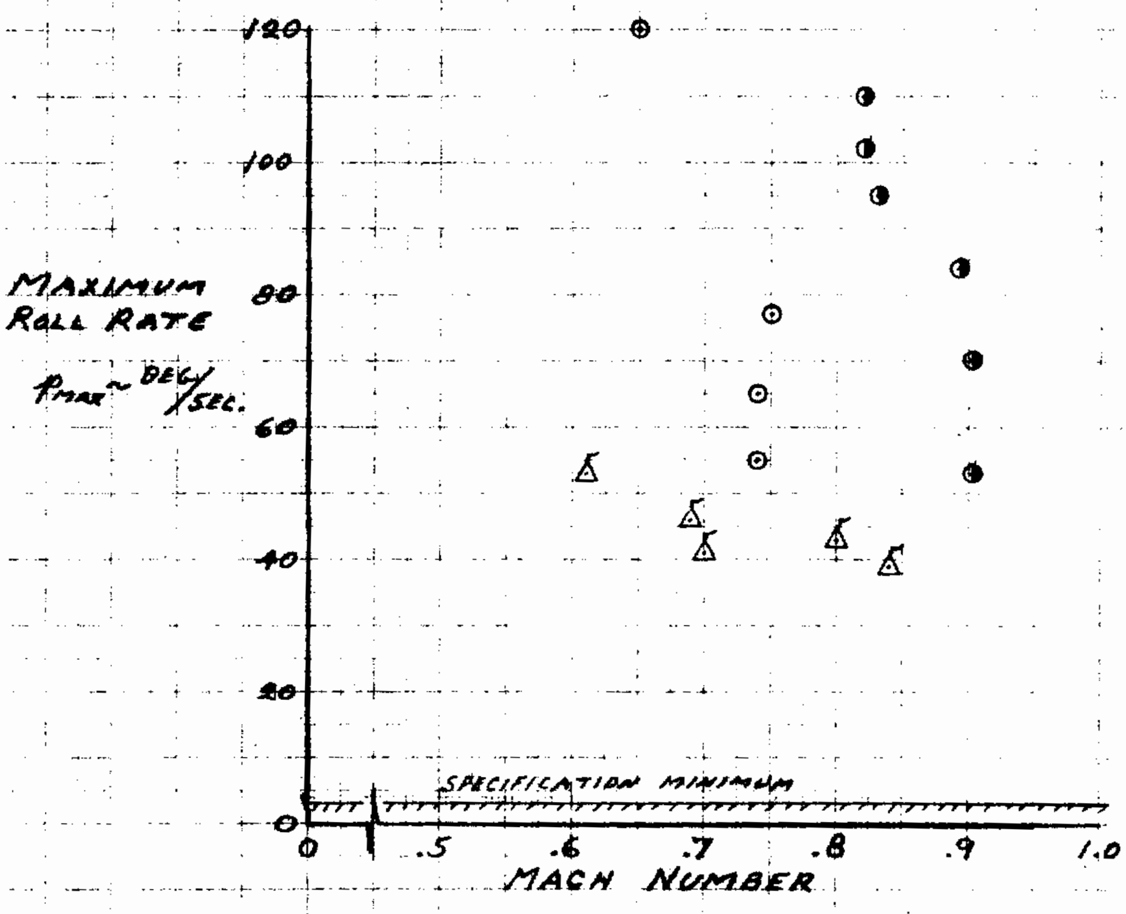
None

F-5 FLIGHT TEST DATA
MAXIMUM ROLL RATE FOR RUDDER INDUCED ROLLS

SYMBOL	CONFIGURATION
○	
△	

ALTITUDE
20,000 FT ~ OPEN SYMBOL
30,000 FT ~ ½ FILLED SYMBOL
10,000 FT ~ FLAGGED SYMBOL

NOTE:
1. RUDDER DEFLECTION USED REQUIRED RUDDER PEDAL FORCES LESS THAN OR EQUAL TO 50 POUNDS.
2. RIGHT RUDDER REQUIRED FOR RIGHT ROLL.



RUDDER PEDAL INDUCED ROLLS

FIGURE 1 (3.3.4.5)

Requirement

Paragraph 3.3.5 Directional control characteristics. Directional stability and control characteristics shall enable the pilot to balance yawing moments and control yaw and sideslip. Sensitivity to rudder pedal forces shall be sufficiently high that directional control and force requirements can be met and satisfactory coordination can be achieved without unduly high rudder pedal forces, yet sufficiently low that occasional improperly coordinated control inputs will not seriously degrade the flying qualities.

Comparison

Comparison of the F-5 directional characteristics with the requirements of this specification is presented in the appropriate succeeding sub-paragraphs.

Resolution

None

Recommendation

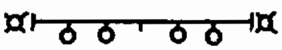
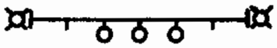
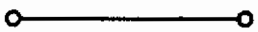
None

Requirement

Paragraph 3.3.5.1 Directional control with speed change. When initially trimmed directionally with symmetric power, the trim change of propeller-driven airplanes with speed shall be such that straight flight can be maintained over a speed range of ±30 percent of the trim speed or ±100 knots equivalent airspeed, whichever is less (except where limited by boundaries of the Service Flight Envelope) with rudder pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming. For other airplanes, rudder pedal forces shall not exceed 40 pounds at the specified conditions for Levels 1 and 2 nor 180 pounds for Level 3.

Comparison

Flight test data obtained from accelerate-decelerate maneuvers were used to compare the F-5 directional control characteristics with the requirements of this paragraph. The following is a tabulation of configurations and flight conditions representative of F-5 Category A Flight Phases that have been evaluated in flight tests.

CONFIGURATION	ALTITUDE	TRIM SPEED	SPEED RANGE
	10,000 Ft	359 Knots	276 - 476 Knots
	10,000 Ft	391 Knots	276 - 477 Knots
	5,000 Ft	426 Knots	242 - 604 Knots
	40,000 Ft	273 Knots	232 - 326 Knots

The pilot was able to maintain straight flight paths throughout these portions of the flight envelope using less than 20 pounds rudder pedal including breakout force of 13 pounds.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.5.1.1 Directional control with asymmetric loading. When initially trimmed directionally with each asymmetric loading specified in the contract at any speed in the Operational Flight Envelope, it shall be possible to maintain a straight flight path throughout the Operational Flight Envelope with rudder pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming.

Comparison

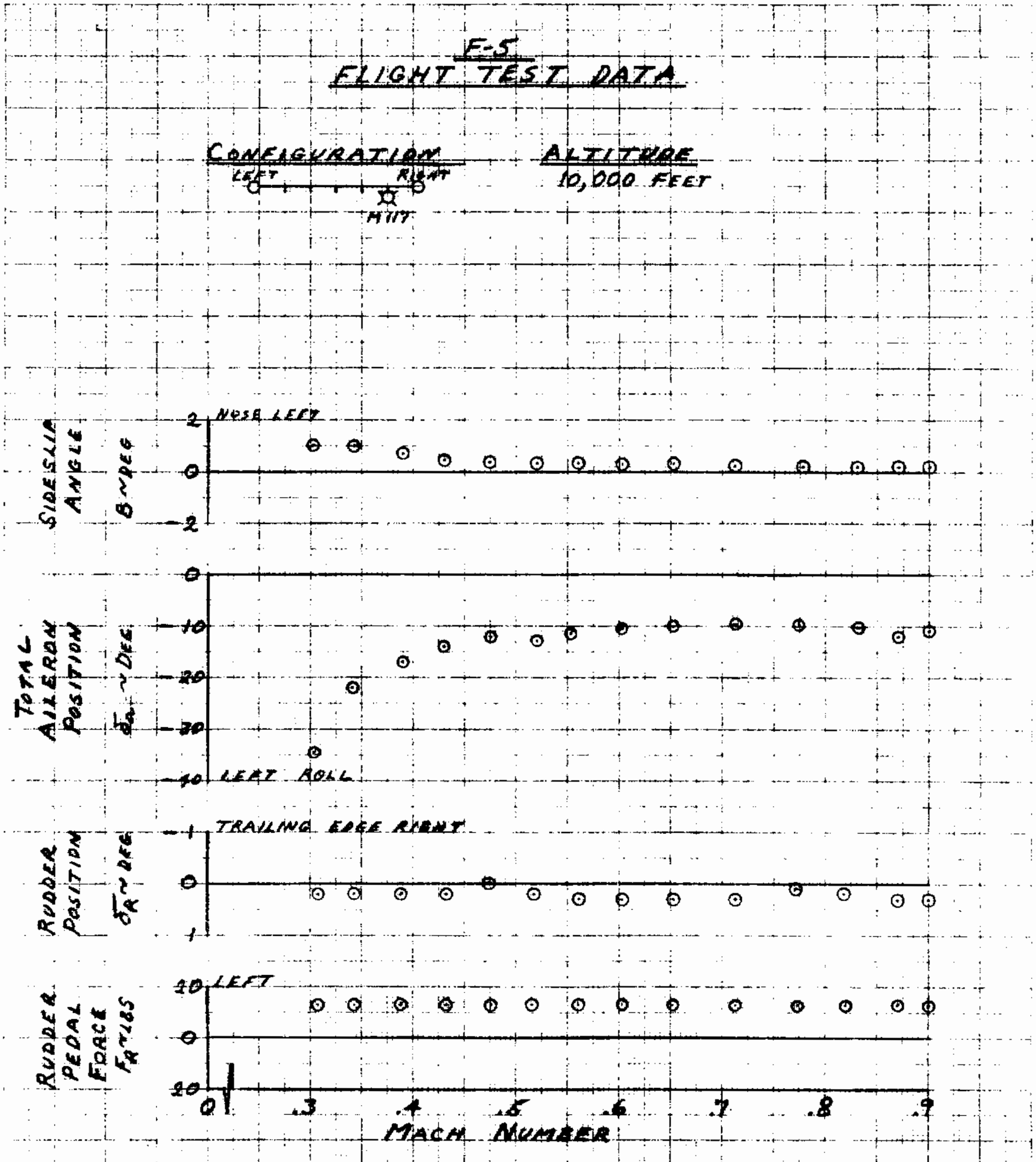
Flight test data, from a decelerate maneuver for a configuration representative of an extreme asymmetric loading, were used for comparison of F-5 characteristics with the paragraph requirements. Figure 1 (3.3.5.1.1) presents pertinent control and airplane response parameters plotted as a function of Mach number. Increasing left roll aileron control is applied to hold wings level as Mach number decreases. Nose-left sideslip is maintained to balance the yawing moment induced by the right-wing-heavy asymmetry. Rudder pedal forces were less than 20 pounds throughout the Mach number range.

Resolution

None

Recommendation

None



DIRECTIONAL CONTROL WITH ASYMMETRIC LOADING

FIGURE 1 (3.3.5.1.1)

Requirement

Paragraph 3.3.5.2 Directional control in wave-off (go-around). For propeller-driven Class IV, and all propeller-driven carrier-based airplanes, the response to thrust, configuration, and airspeed change shall be such that the pilot can maintain straight flight during wave-off (go-around) initiated at speeds down to V_S (PA) with rudder pedal forces not exceeding 100 pounds when trimmed at $V_{o_{min}}$ (PA). For other airplanes, rudder pedal forces shall not exceed 40 pounds for the specified conditions. The preceding requirements apply for Levels 1 and 2. For all airplanes the Level 3 requirement is to maintain straight flight in these conditions with rudder pedal forces not exceeding 180 pounds. For all Levels, bank angles up to 5 degrees are permitted.

Comparison

The accelerate portion (representing go-around) of a decelerate-accelerate maneuver was used to compare F-5 directional control characteristics with the paragraph requirements. With the airplane initially trimmed at 195 KCAS, near $V_{o_{min}}$ (PA), the airplane was decelerated to 152 KCAS, near V_S (PA). Power was then applied initiating the go-around and the airplane was accelerated to 199 KCAS. Rudder pedal forces and roll angle did not exceed 15 pounds and 2 degrees, respectively, as shown in Figure 1 (3.3.5.2).

Resolution

None

Recommendation

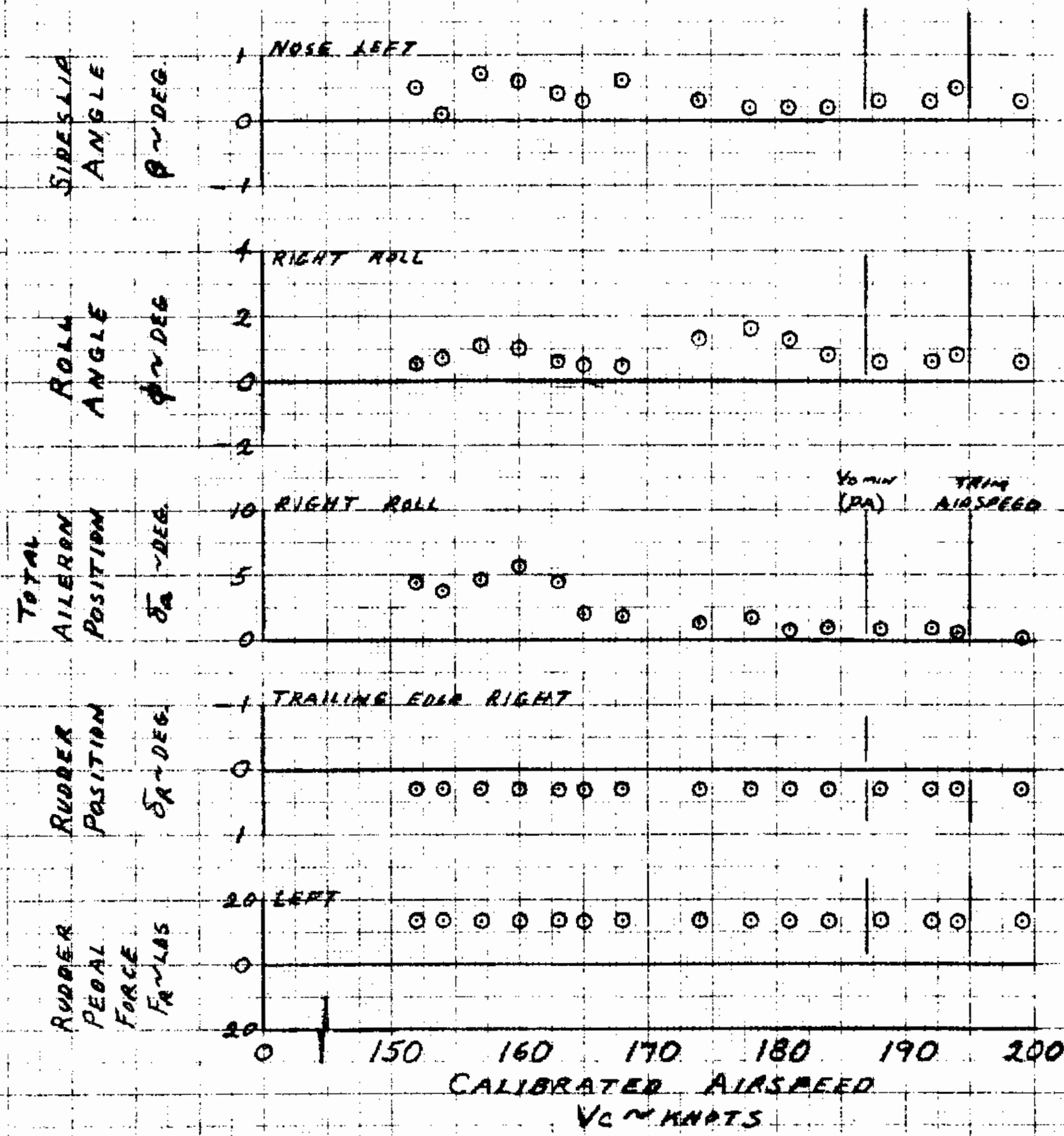
None

F-5
FLIGHT TEST DATA

CONFIGURATION

ALTITUDE
10,000 FEET

LANDING GEAR AND FLAPS EXTENDED
FLIGHT PHASE CATEGORY C



DIRECTIONAL CONTROL IN WAVE-OFF (GO-AROUND)

FIGURE 1 (3.3.5.2)

Requirement

Paragraph 3.3.6 Lateral-directional characteristics in steady sideslips. The requirements of 3.3.6.1 through 3.3.6.3.1 and 3.3.7.1 are expressed in terms of characteristics in rudder-pedal-induced steady, zero-yaw-rate sideslips with the airplane trimmed for wings-level straight flight. Paragraphs 3.3.6.1 through 3.3.6.3 apply at sideslip angles up to those produced or limited by:

a. Full rudder pedal deflection, or

b. 250 pounds of rudder pedal force, or

c. Maximum aileron control or surface deflection,

except that for single-propeller-driven airplanes during wave-off (go-around), rudder pedal deflection in the direction opposite to that required for wings-level straight flight need not be considered beyond the deflection for a 10-degree change in sideslip from the wings-level straight flight condition.

Comparison

Data to validate the requirements of this paragraph appear in the appropriate succeeding paragraphs.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.6.1 Yawing moments in steady sideslips. For the sideslips specified in 3.3.6, right rudder pedal deflection and force shall produce left sideslips and left rudder pedal deflection and force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with rudder pedal deflection shall be essentially linear for sideslip angles between +15 degrees and -15 degrees. For larger sideslip angles, an increase in rudder pedal deflection shall always be required for an increase in sideslip. The variation of sideslip angle with rudder pedal force shall be essentially linear for sideslip angles between +10 degrees and -10 degrees. Although a lightening of rudder pedal force is acceptable for sideslip angles outside this range, the rudder pedal force shall never reduce to zero .

Comparison

Flight test data from steady sideslips were used for comparison of F-5 characteristics with the paragraph requirements. The configurations analyzed were clean, three-stores, and five stores. The test altitude was 20,000 feet with Mach numbers ranging from 0.44 to 0.89. Figures 1 (3.3.6.1) through 9 (3.3.6.1) present the reduced flight test results.

The flight test airplane was not instrumented to measure rudder pedal deflection. However, the gearing is such that right rudder pedal deflection produces right rudder position and left rudder pedal deflection produces left rudder position. The variation of sideslip angle with rudder position is essentially linear in all cases. Right rudder position produces left sideslip and left rudder position produces right sideslip. The variation of sideslip angle with rudder pedal force is also essentially linear excluding the region of zero rudder position where breakout rudder pedal forces occur.

At the maximum sideslip angles where maximum rudder deflection was reached,the rudder pedal force exhibits a local discontinuity from linearity with sideslip angle. This is due to the continued application of rudder pedal force by the test pilot until he realizes that maximum rudder deflection is reached.

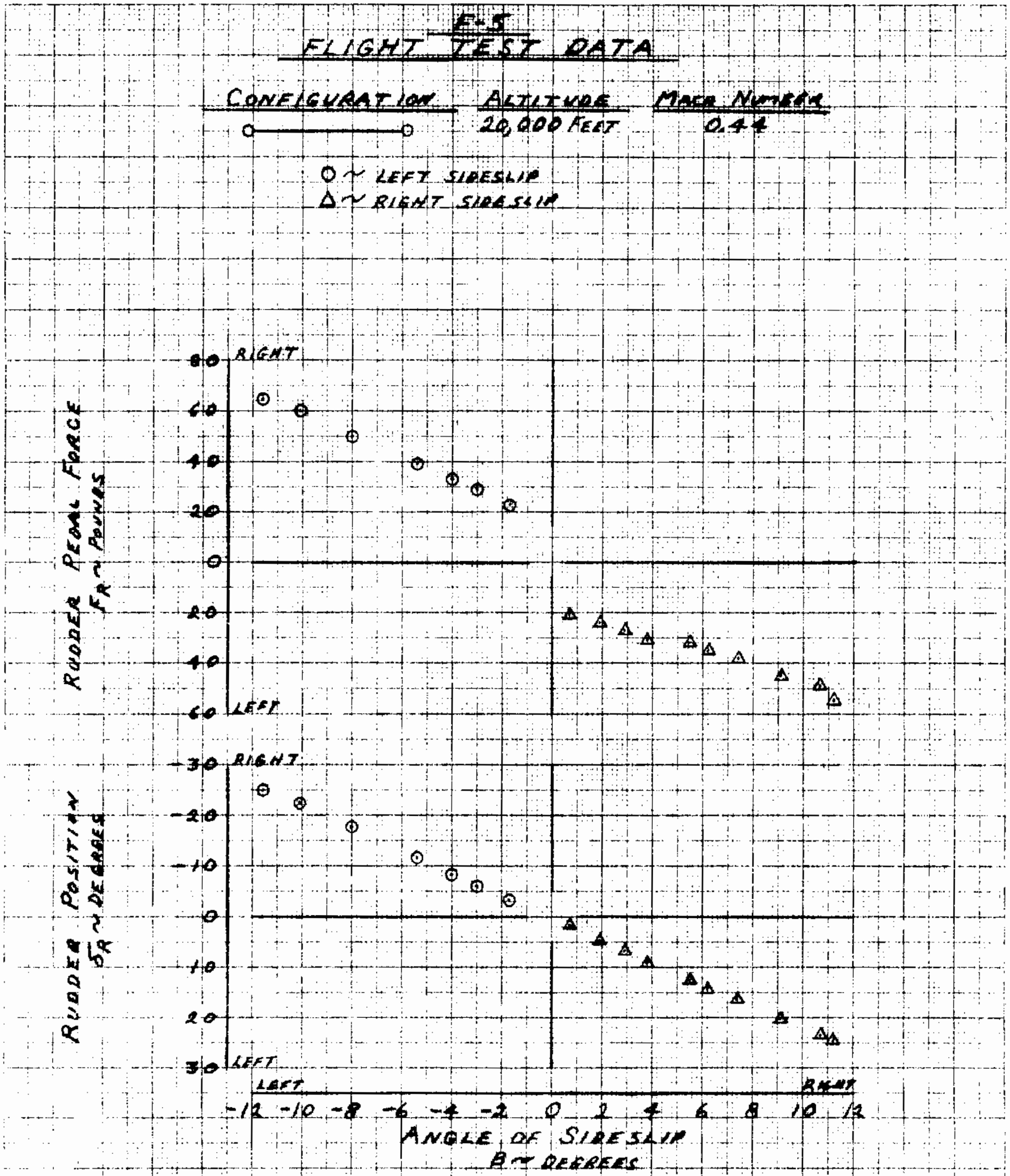
The data presented were taken from an NF-5 (Netherlands version) airplane having mechanical rudder limits of ±30 degrees but the rudder limits are a function of compressible dynamic pressure (q_c). Full (±30 degrees) rudder deflection is attained for q_c from zero to 150 pounds per square foot with an approximate exponential decay to a maximum deflection of ±5 degrees at q_c = 1600 pounds per square foot.

Resolution

None

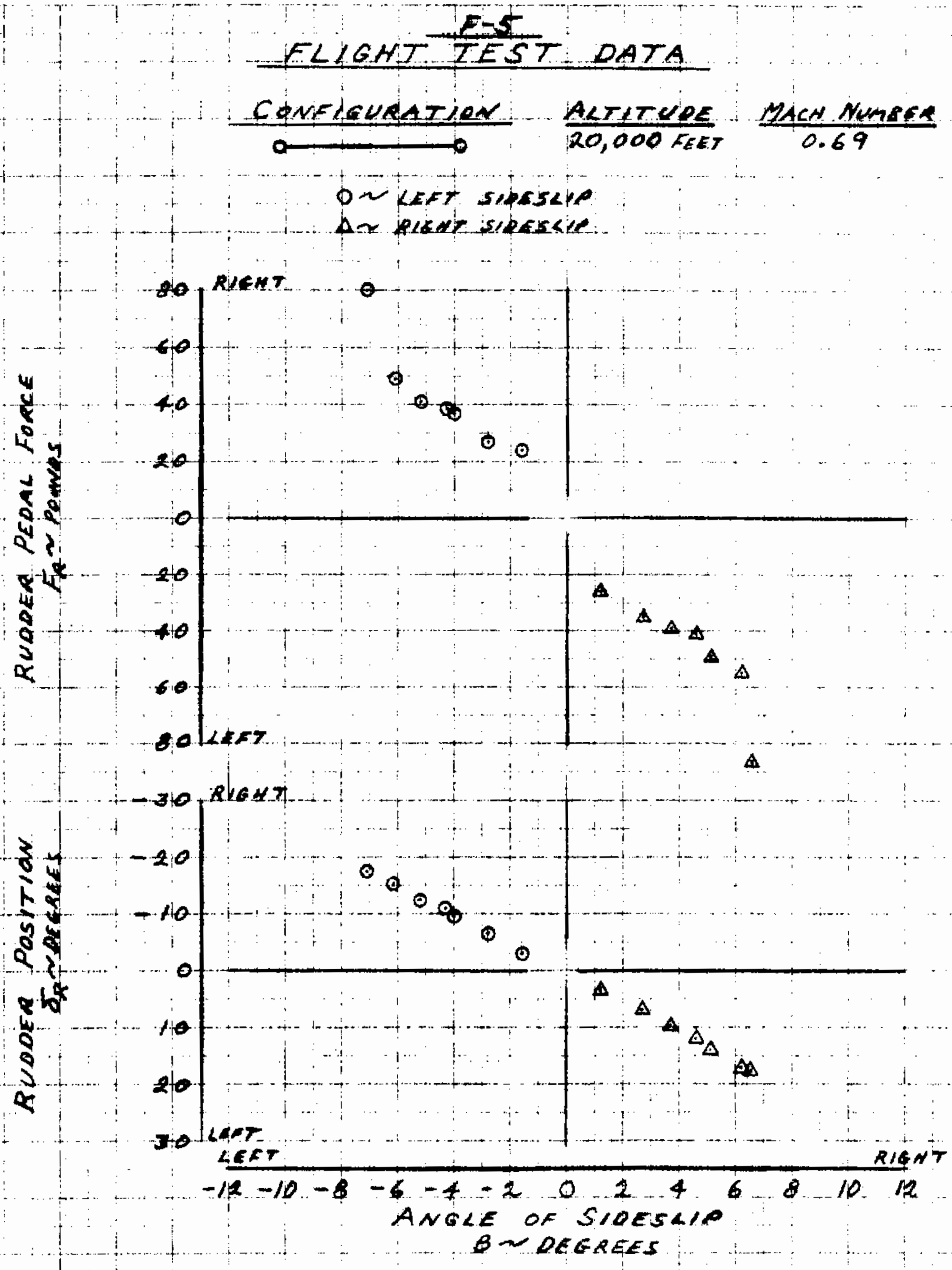
Recommendation

None



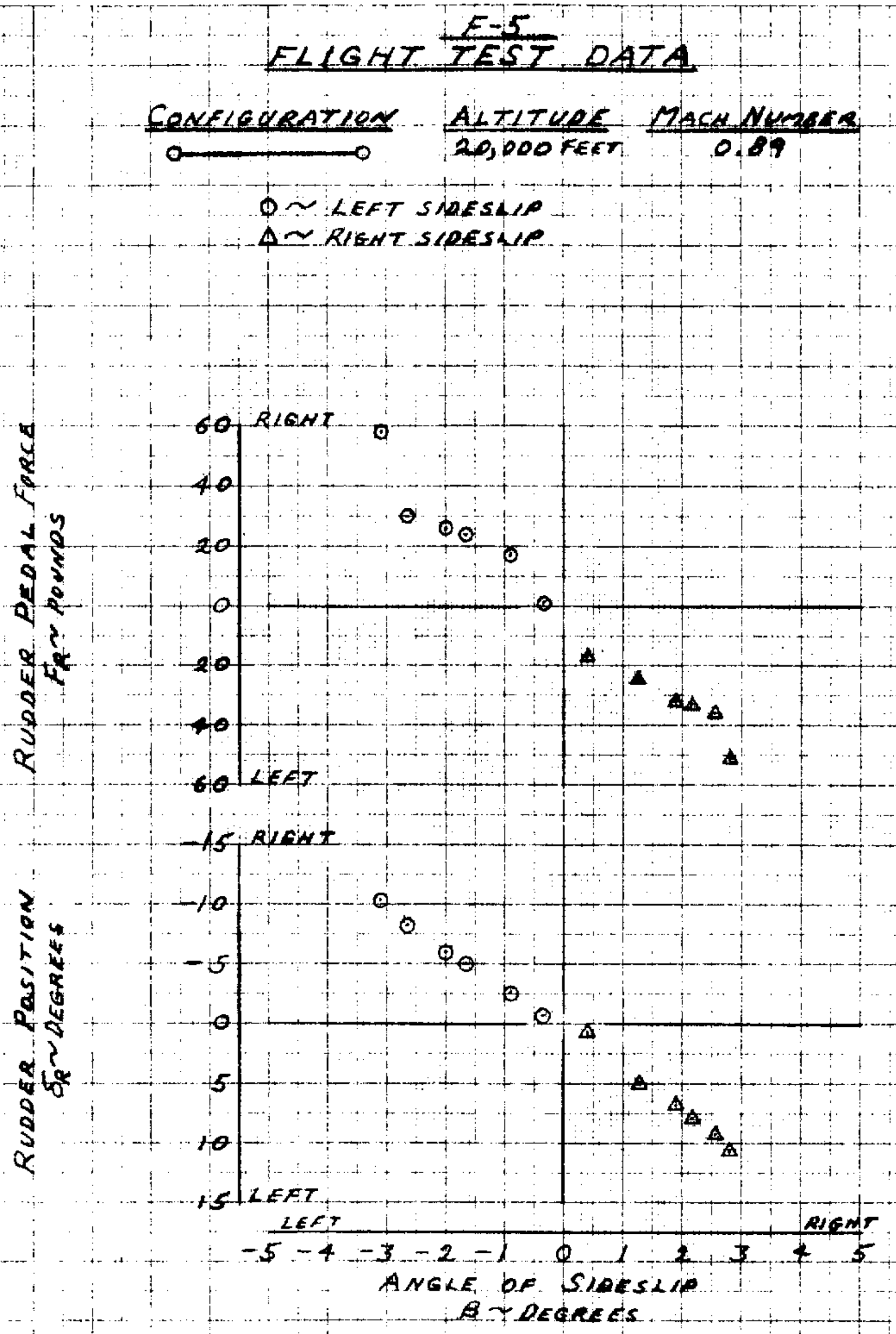
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 1 (3.3.6.1)



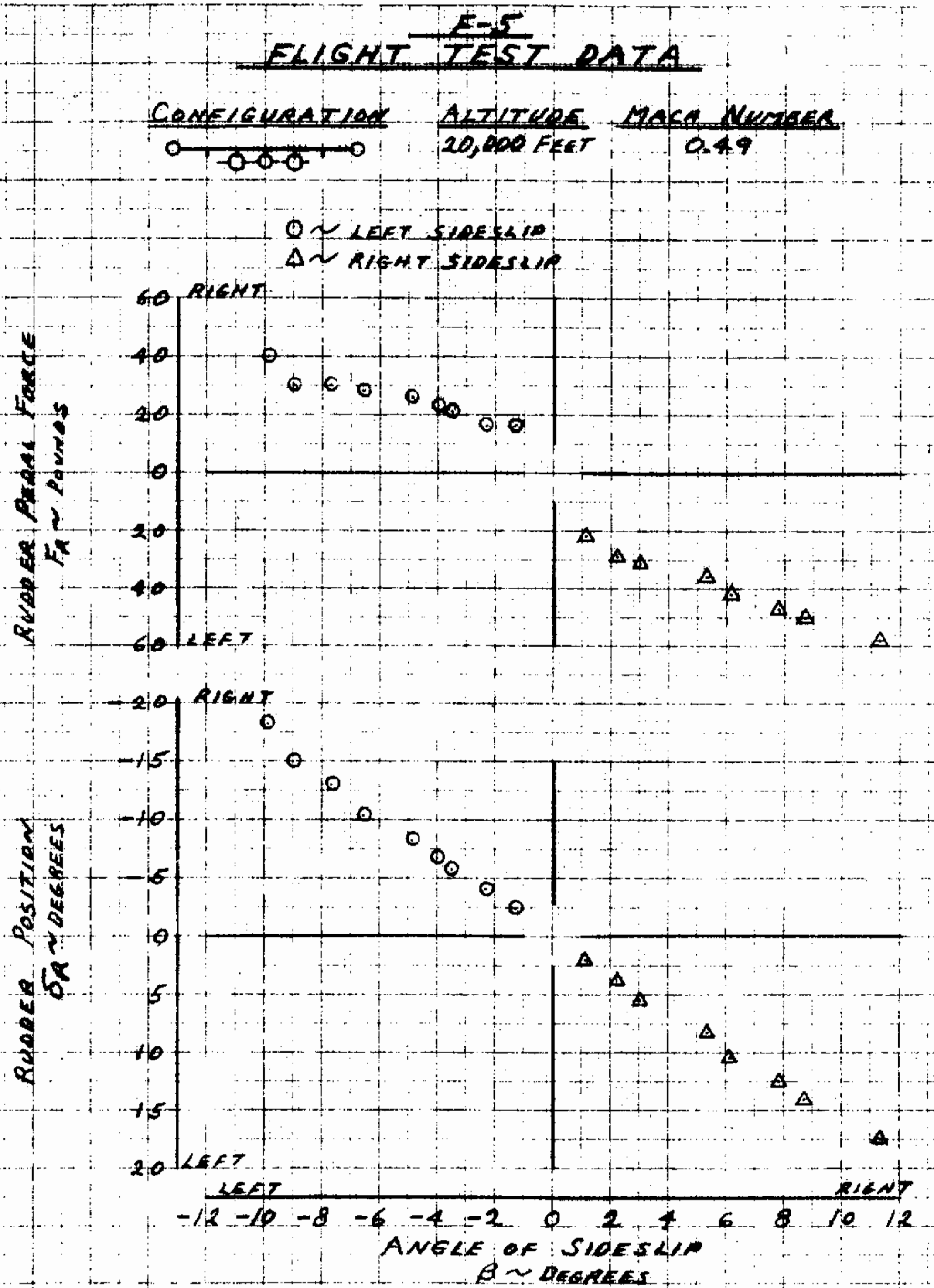
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 2 (3.3.6.1)



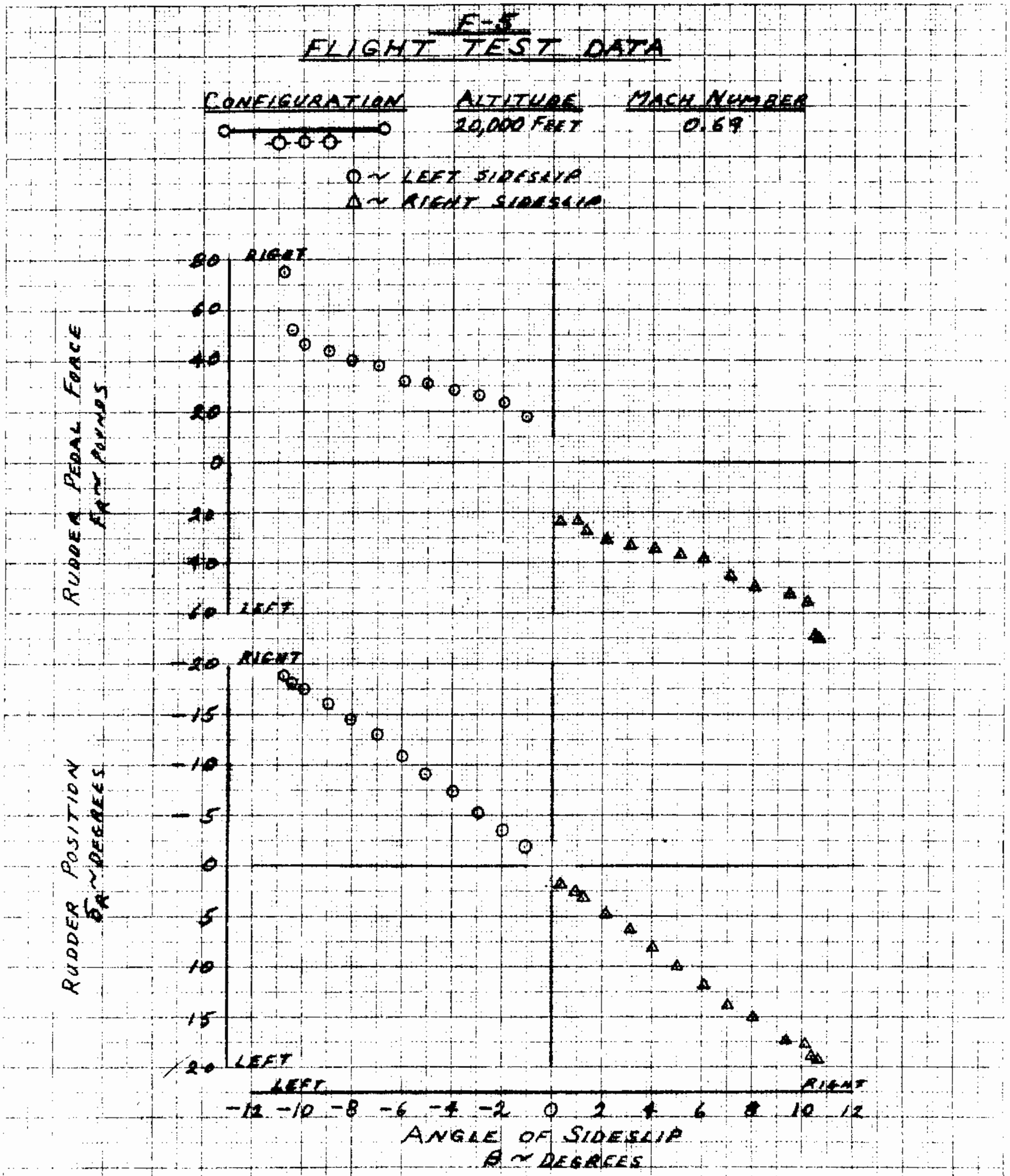
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 3.(3.3.6.1)



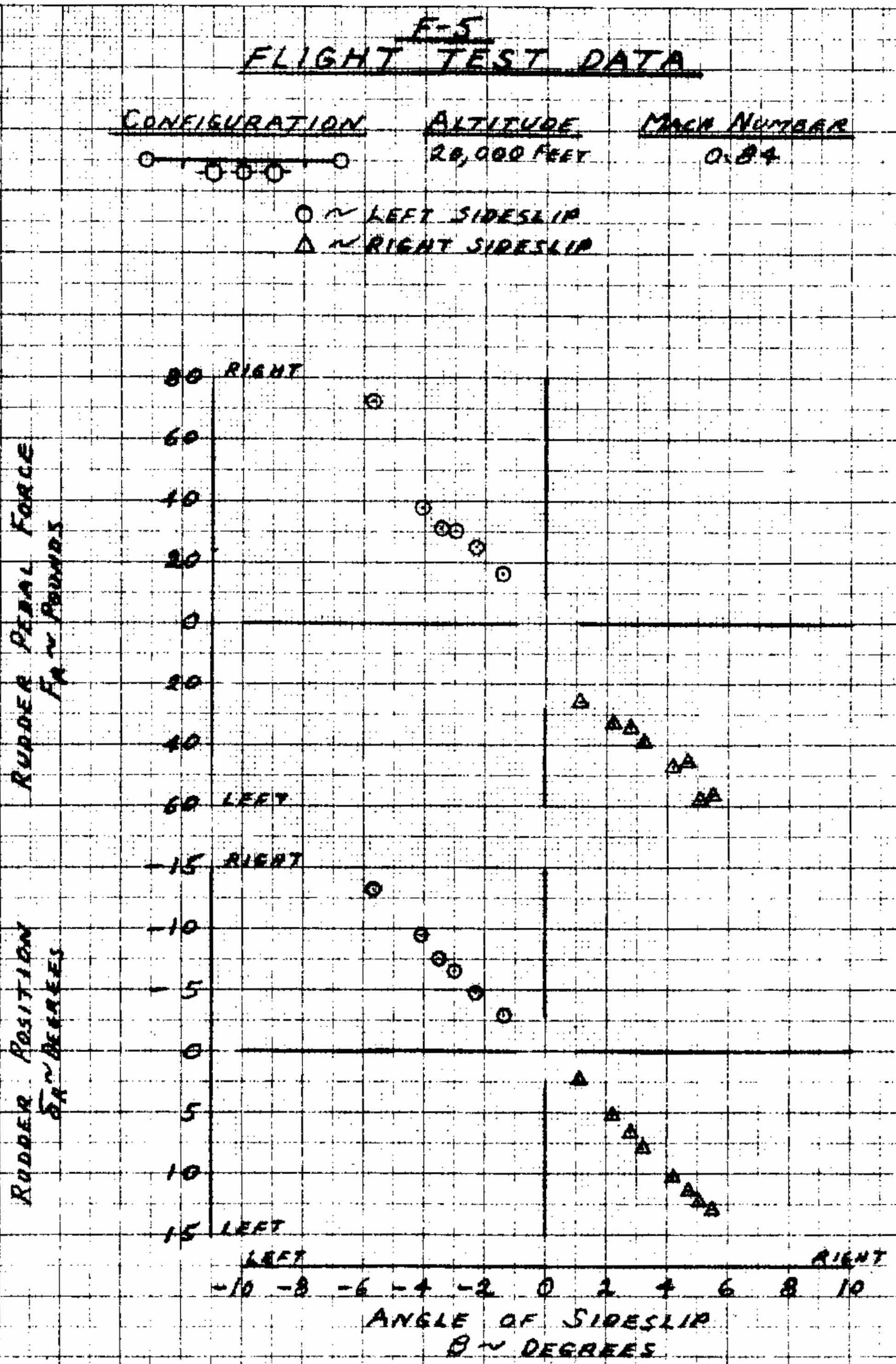
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 4 (3.3.6.1)



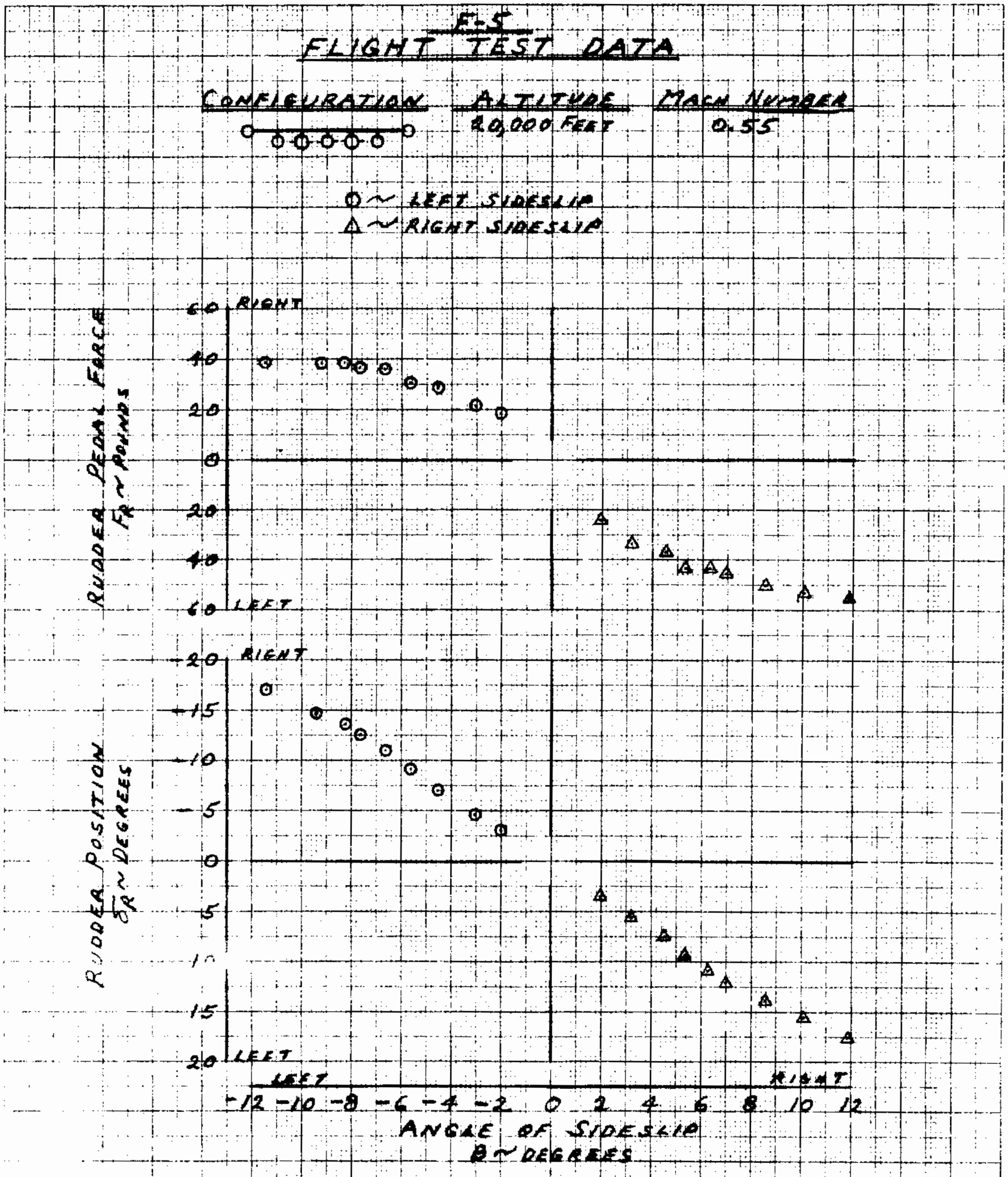
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 5 (3.3.6.1)



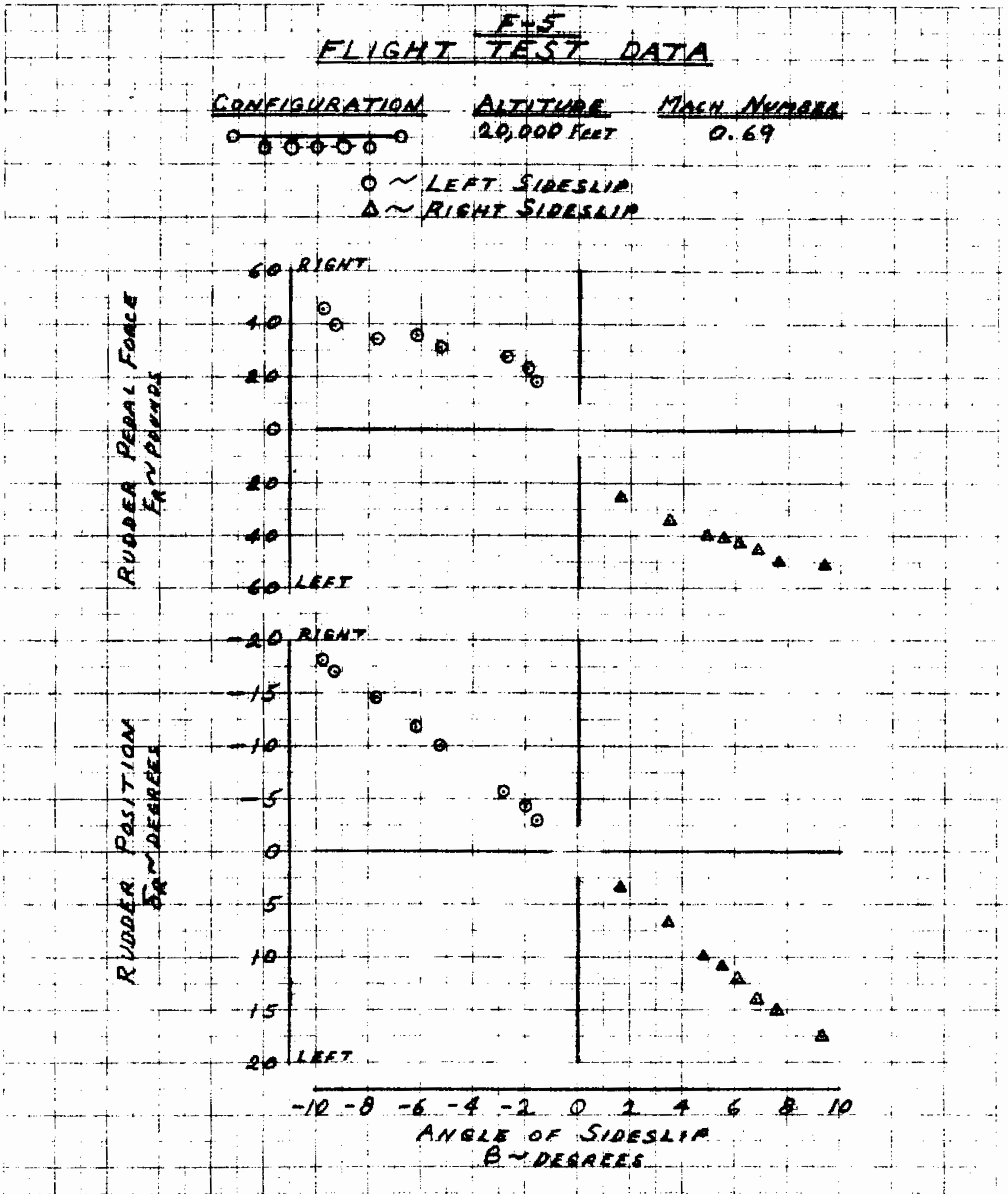
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 6 (3.3.6d)



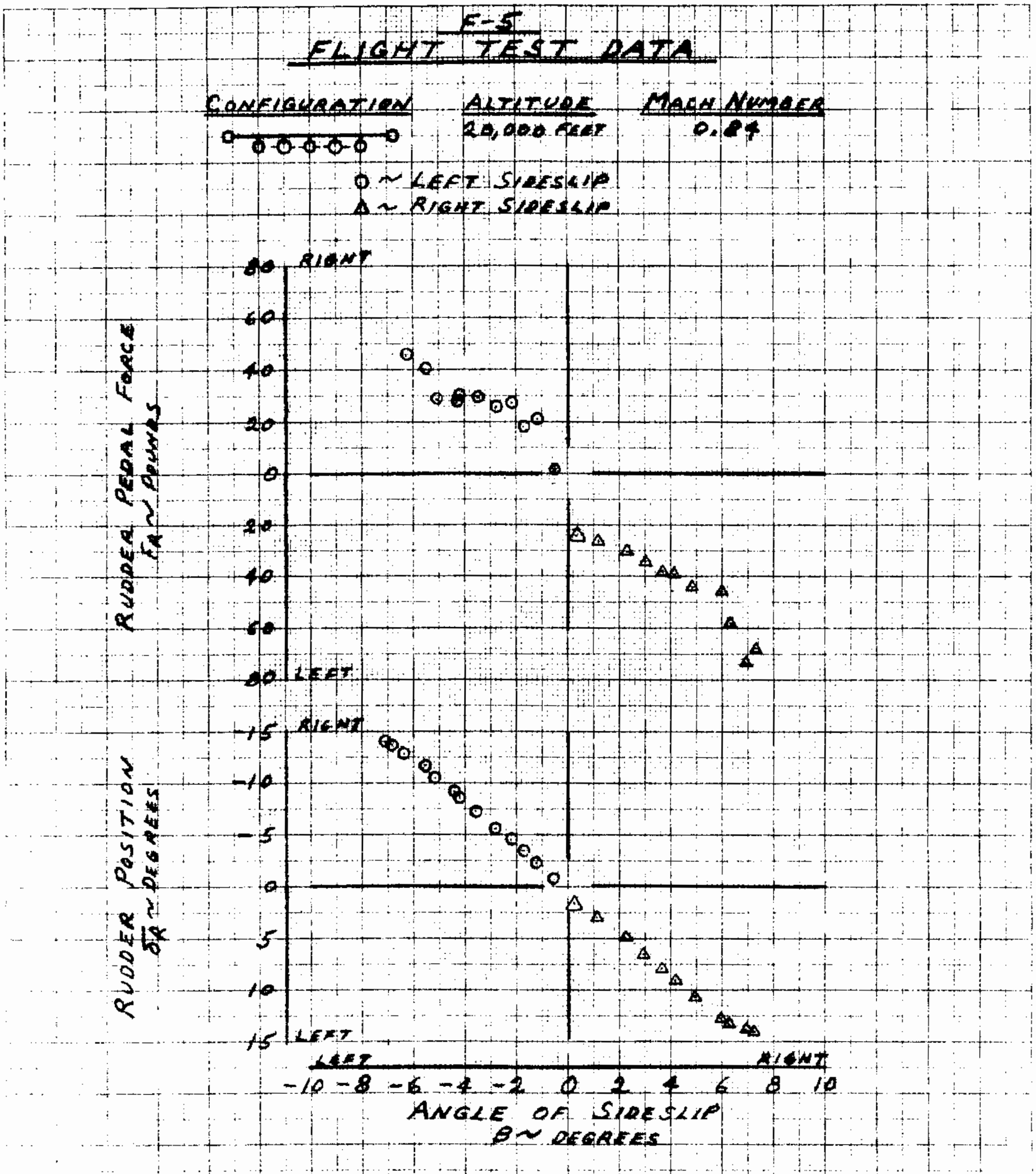
YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 7 (3.3.6.1)



YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 8 (3.3.6.1)



YAWING MOMENTS IN STEADY SIDESLIPS

FIGURE 9 (3.3.6.1)

Requirement

Paragraph 3.3.6.2 Side forces in steady sideslips. For the sideslips of 3.3.6, an increase in right bank angle shall accompany an increase in right sideslip, and an increase in left bank angle shall accompany an increase in left sideslip.

Comparison

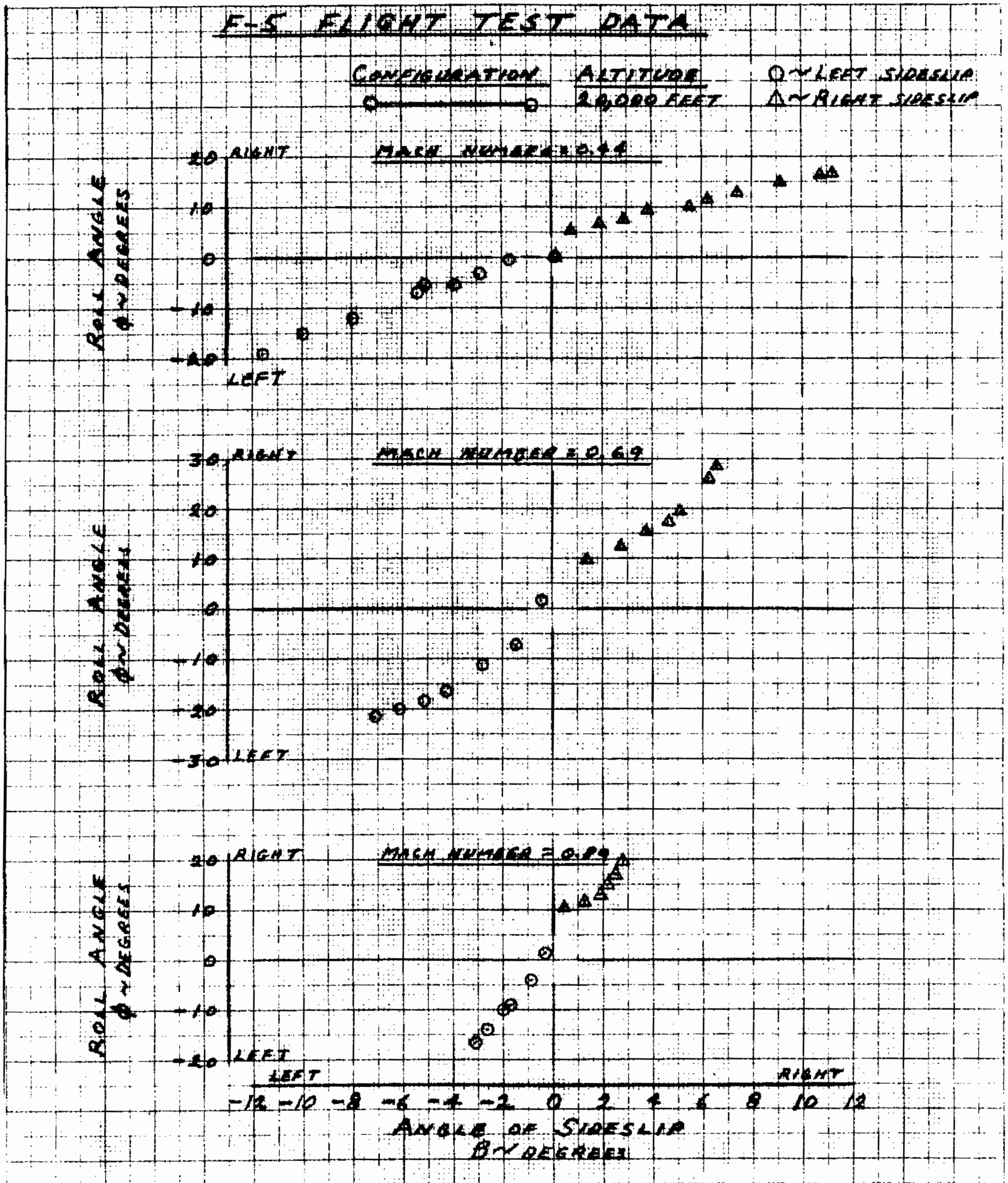
Figures 1 (3.3.6.2) through 3 (3.3.6.2) present roll angle as a function of sideslip angle from the steady sideslip maneuvers of Paragraph 3.3.6.1. Increasing right roll angle accompanies increasing right sideslip and increasing left roll angle accompanies increasing left sideslip in all cases.

Resolution

None

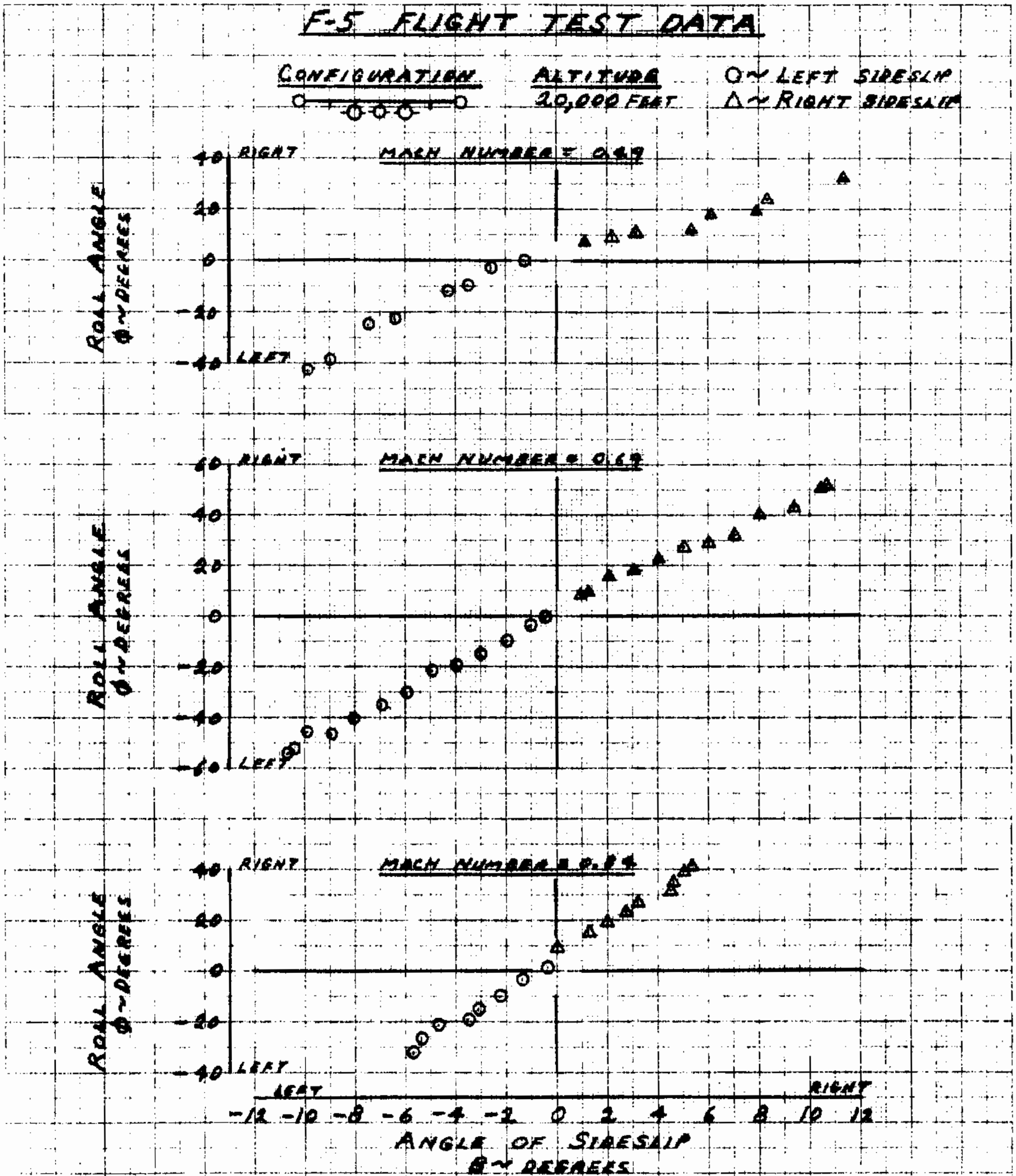
Recommendation

None



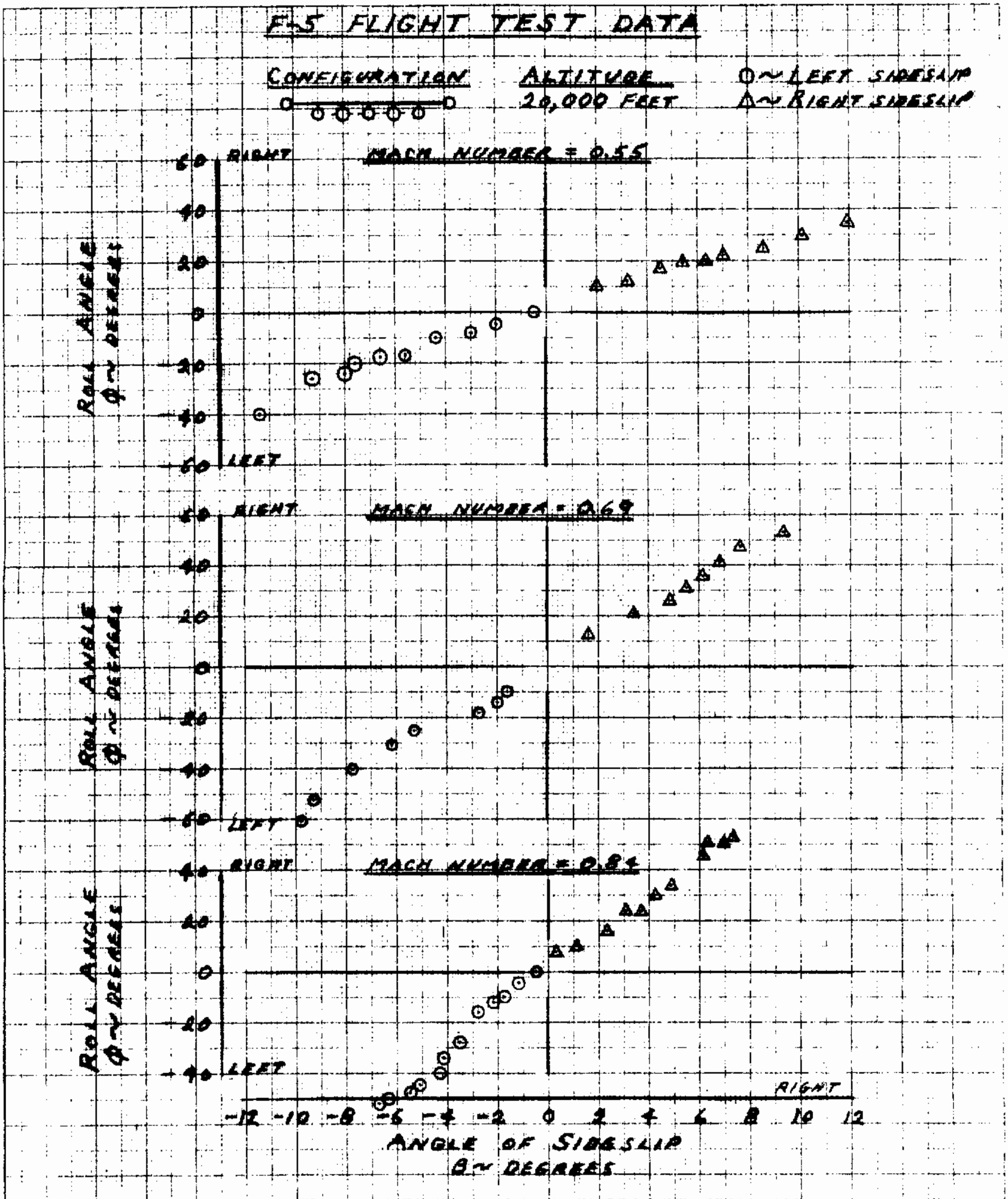
SIDE FORCES IN STEADY SIDESLIPS

FIGURE 1 (3.3.6.2)



SIDE FORCES IN STEADY SIDESLIPS

FIGURE 2 (3.3.6.2)



SIDE FORCES IN STEADY SIDESLIPS

FIGURE 3 (3.3.6.2)

Requirement

Paragraph 3.3.6.3 Rolling moments in steady sideslips. For the sideslips of 3.3.6, left aileron-control deflection and force shall accompany left sideslips, and right aileron-control deflection and force shall accompany right sideslips. For Levels 1 and 2, the variation of aileron-control deflection and force with sideslip angle shall be essentially linear.

Comparison

Figures 1 (3.3.6.3) through 9 (3.3.6.3) present lateral stick forces and total aileron position plotted as a function of sideslip angle. The control gearing is such that left aileron control deflection and force produce left aileron position which accompanies left sideslip and right aileron control deflection and force produce right aileron position which accompanies right sideslip. The data presented were reduced from the steady sideslip maneuvers of Paragraph 3.3.6.1.

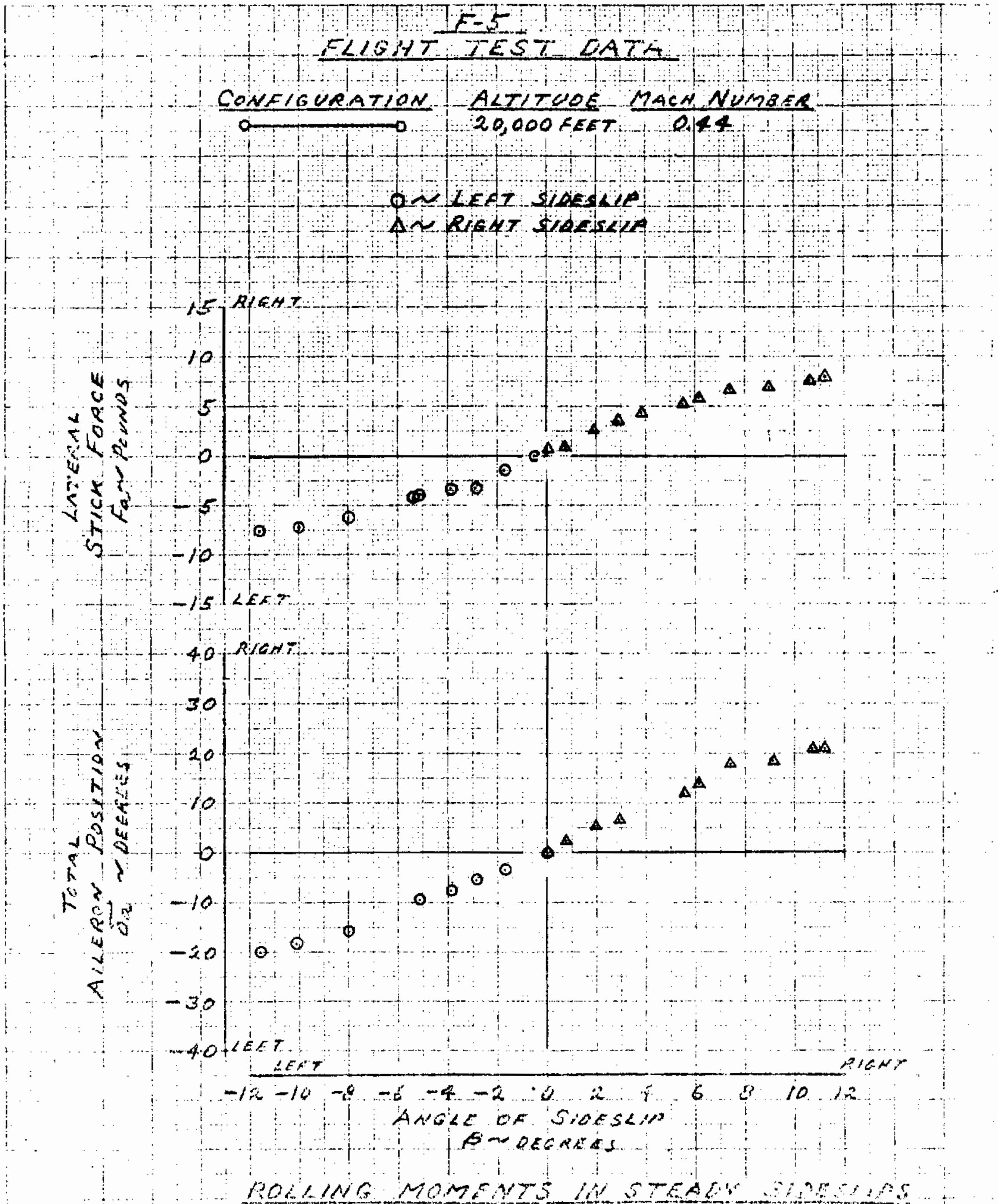
The variation of aileron position and lateral stick forces with sideslip angle is essentially linear for all cases. The gearing of aileron position with aileron control deflection is also essentially linear.

Resolution

None

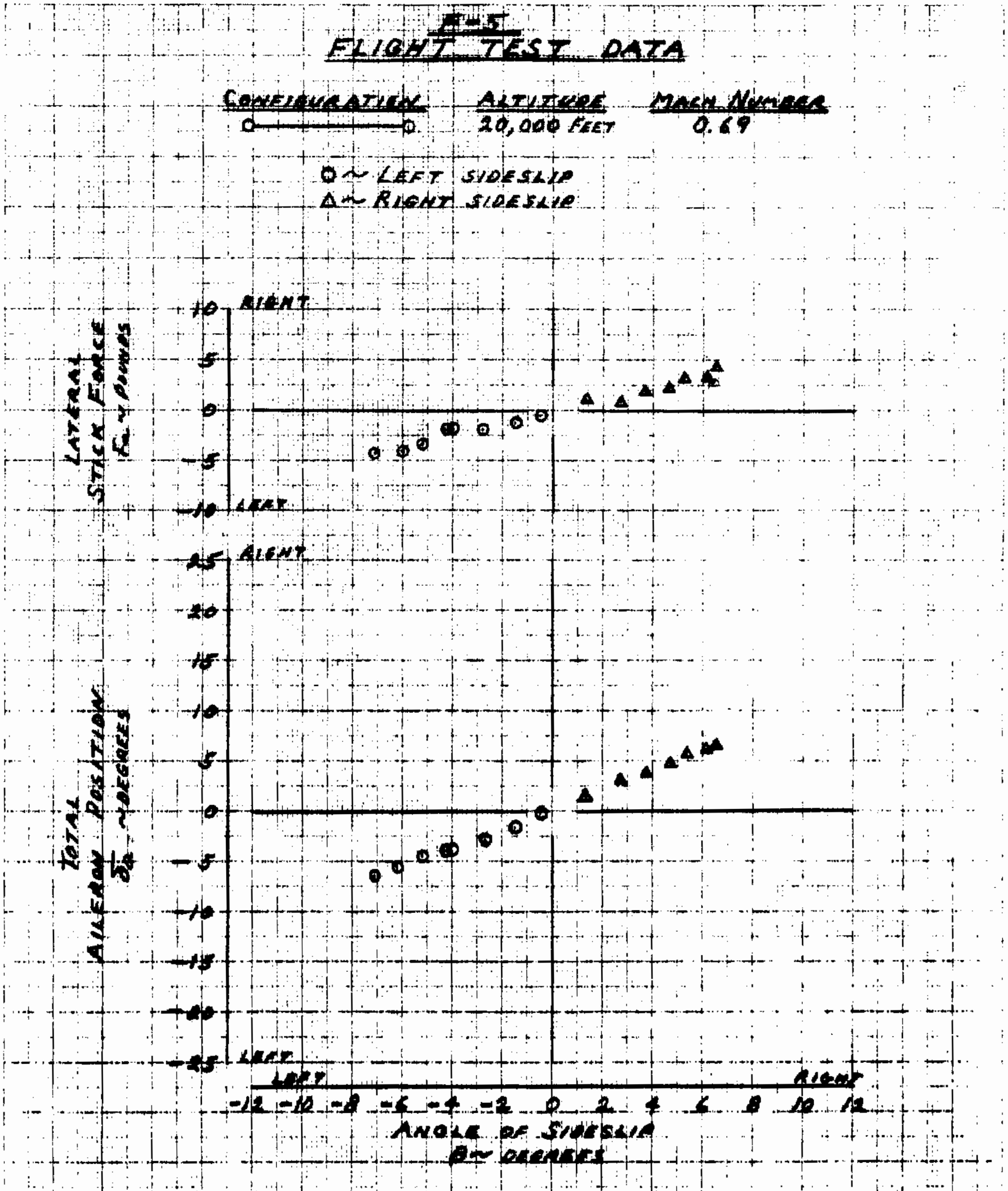
Recommendation

None



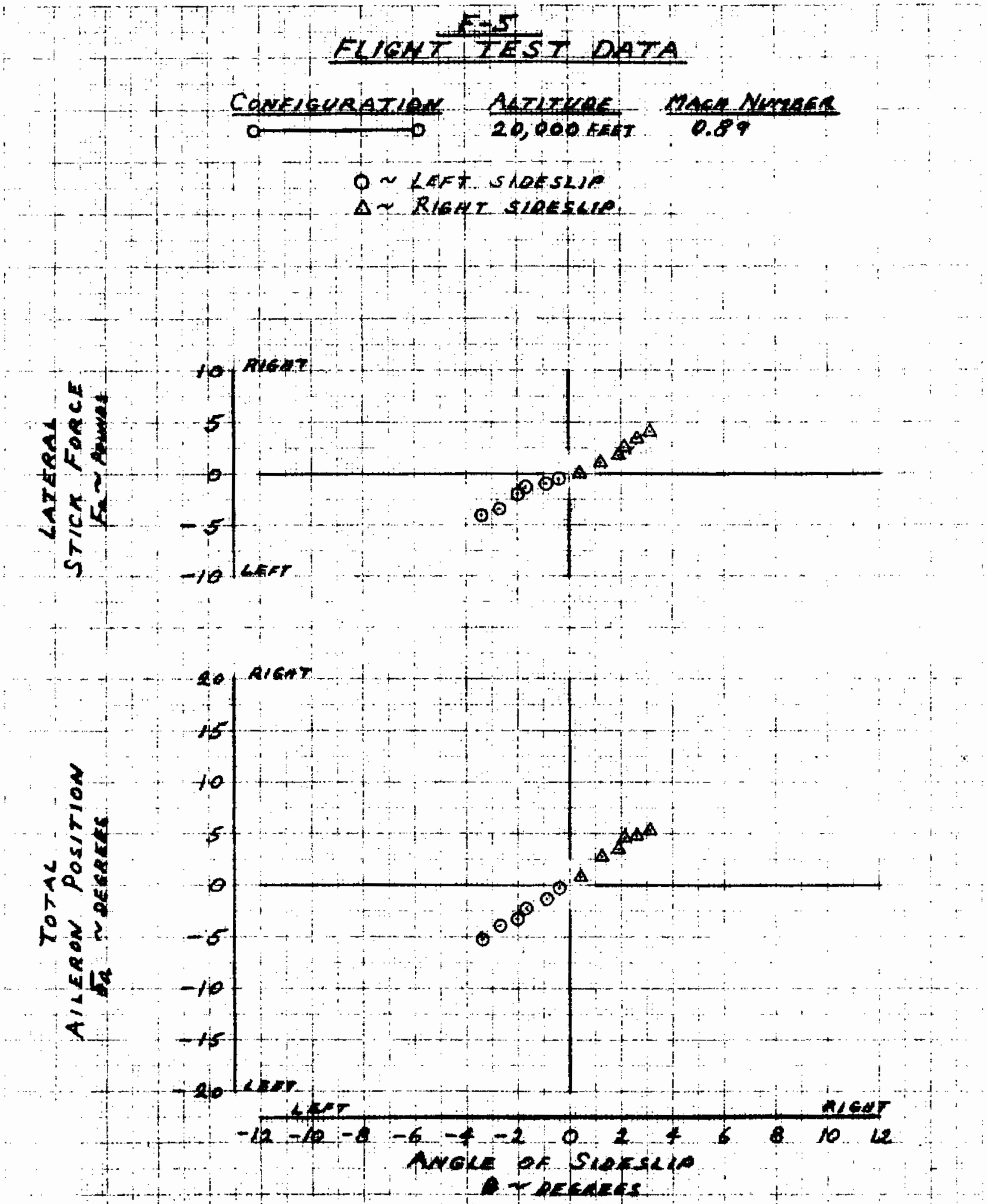
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 1 (3.3.6.3)



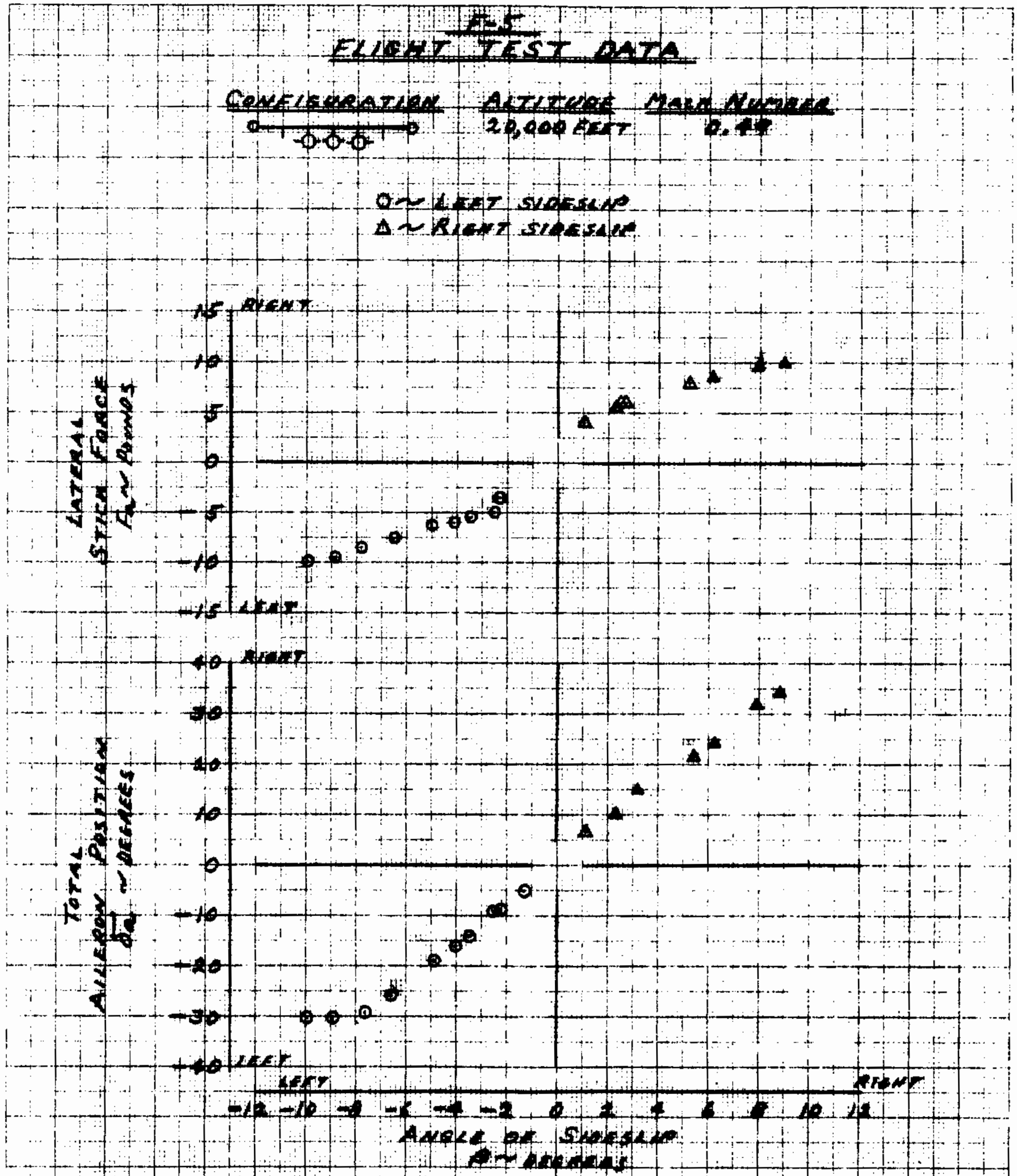
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 2 (3.3.6.3)



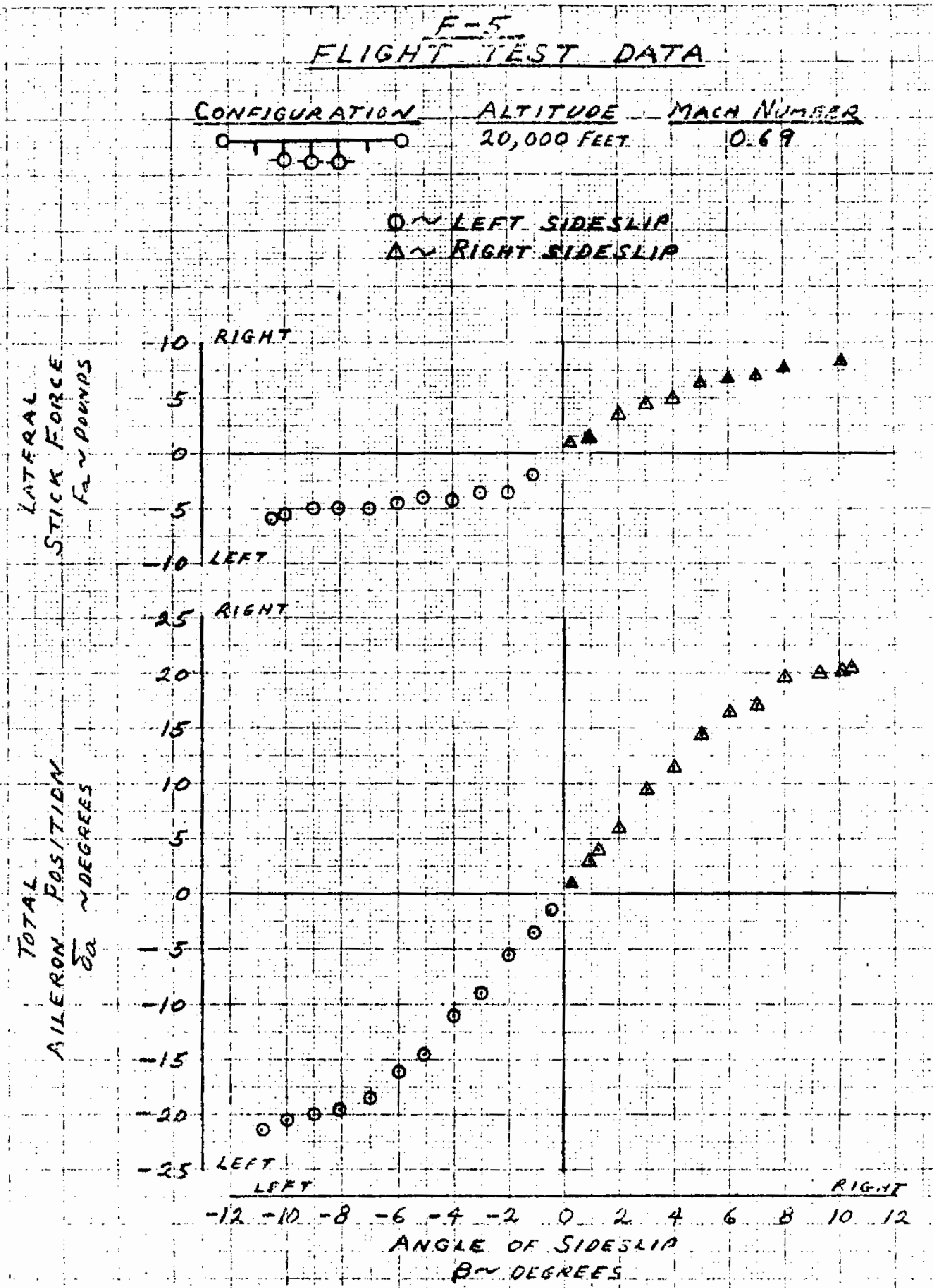
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 3 (3.3.6.3)



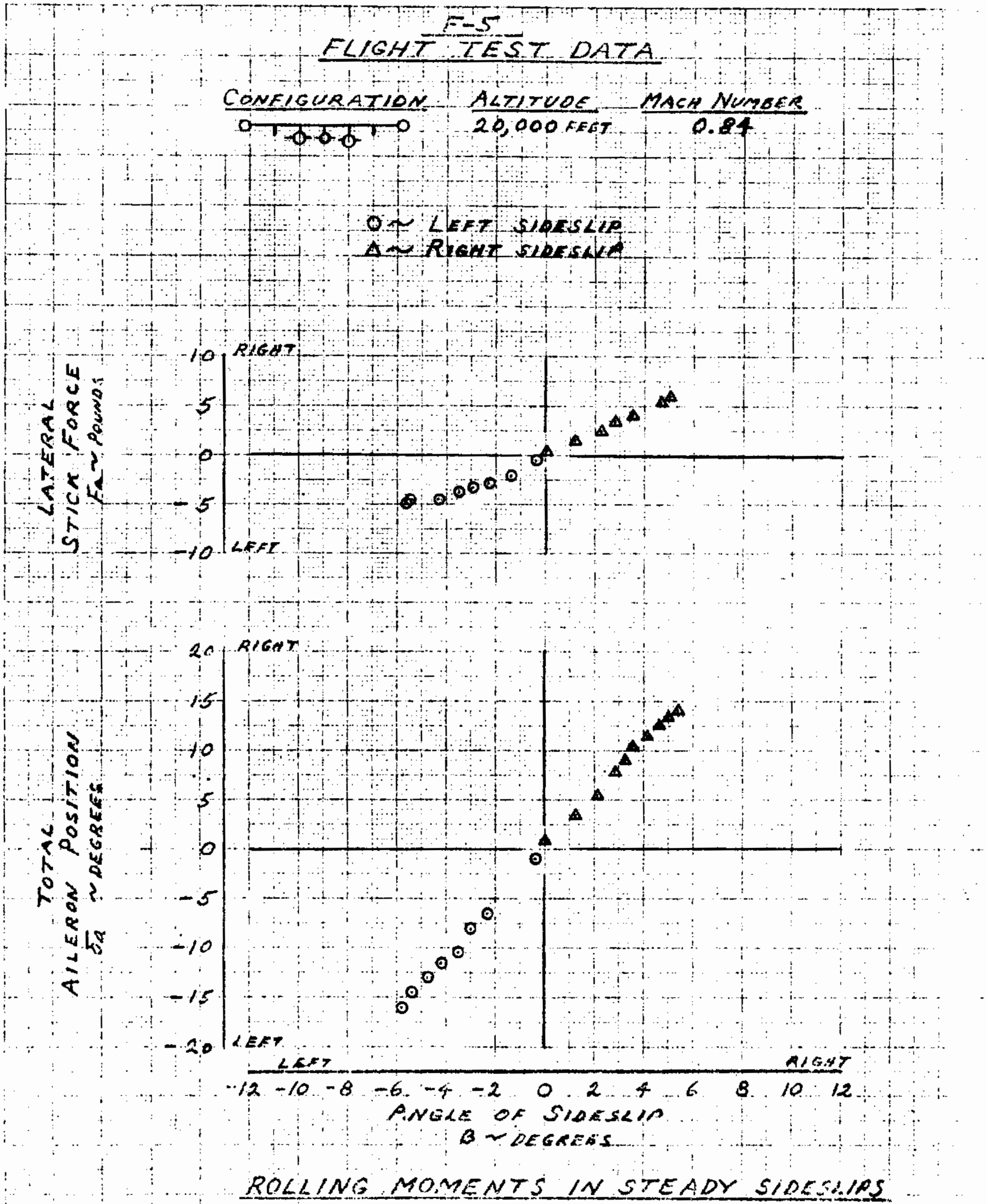
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 4 (3.3.6.3)



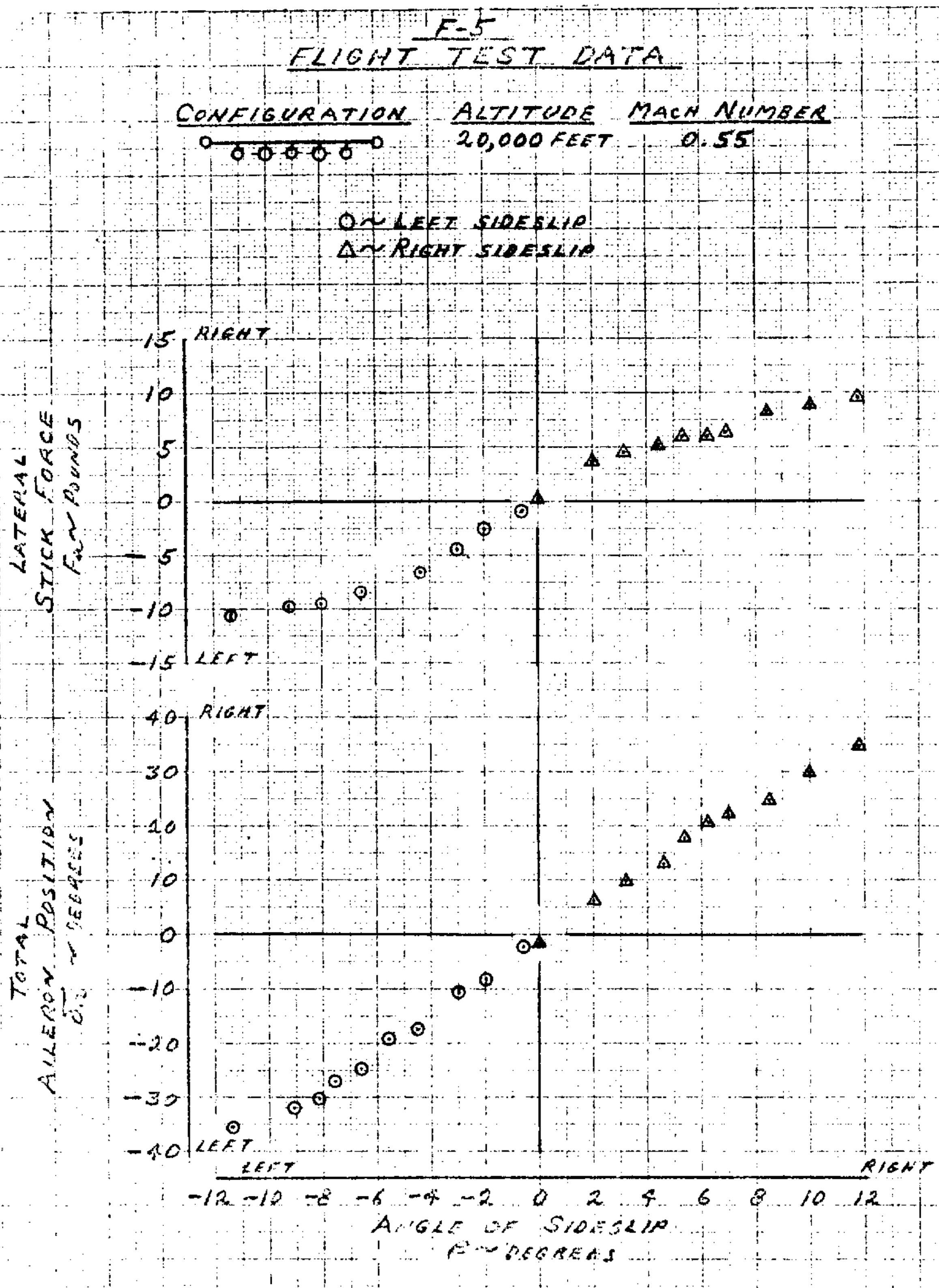
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 5 (3.3.6.3)



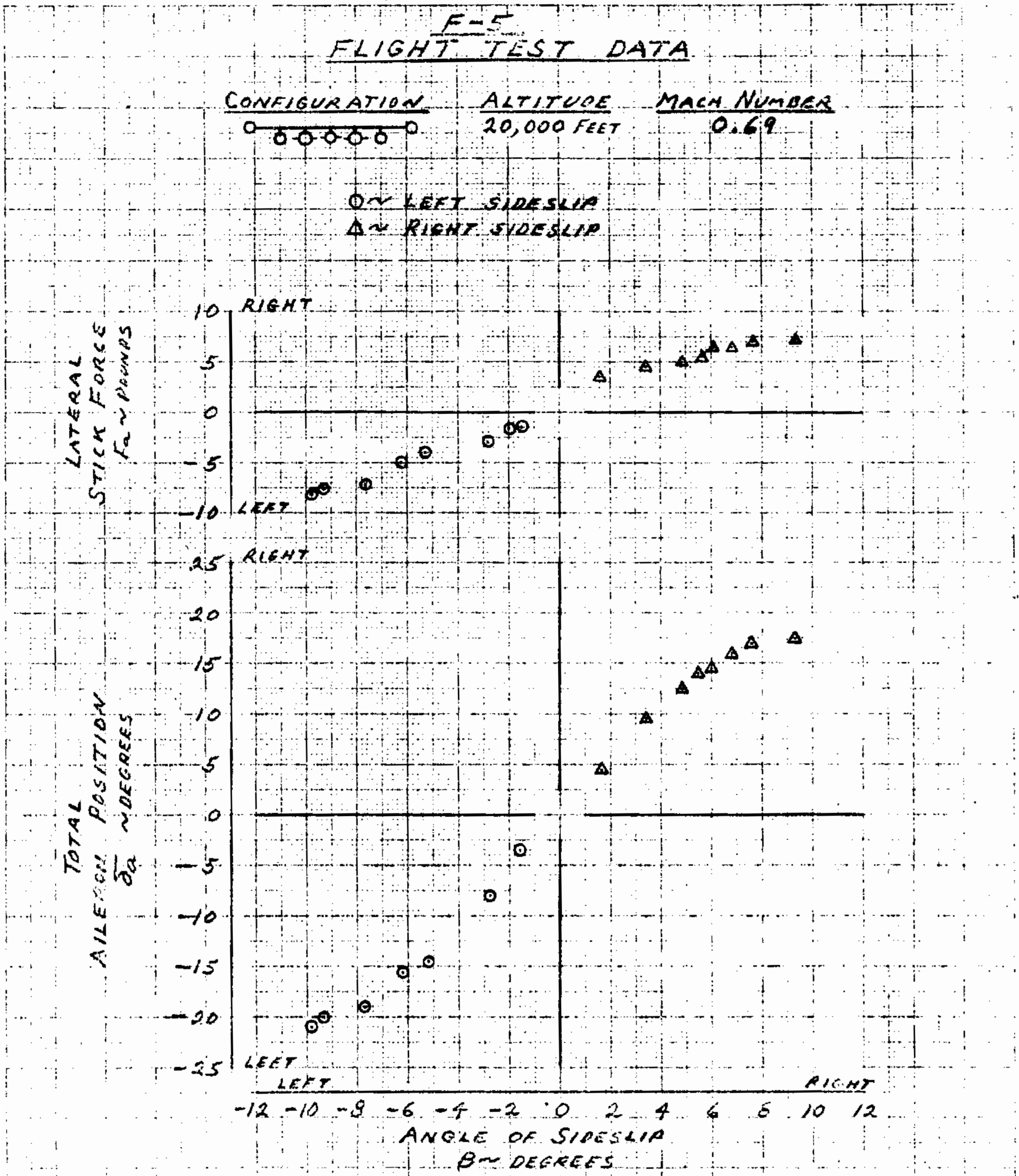
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 6 (3.3.6.3)



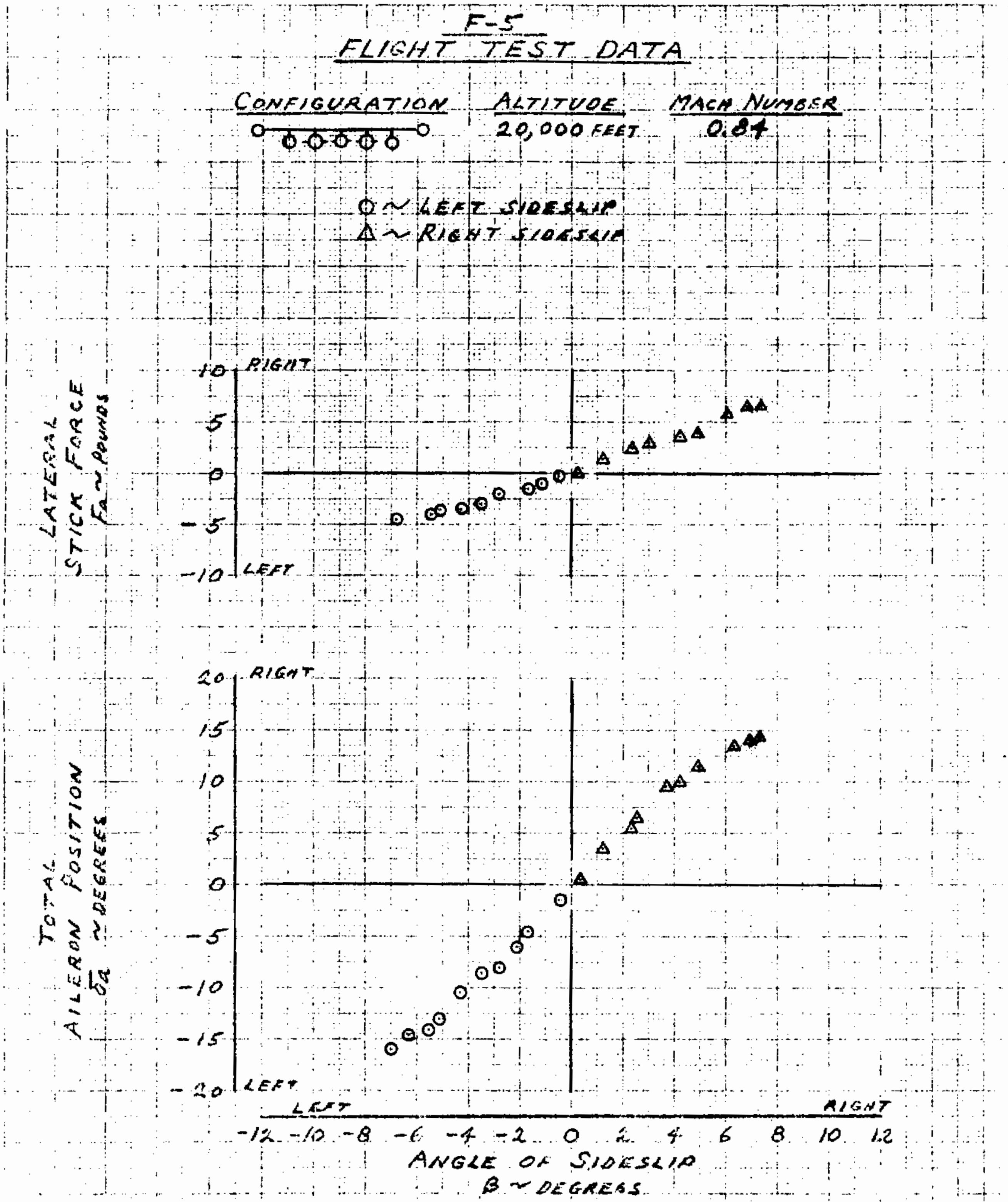
ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 7 (3.3.6.3)



ROLLING MOMENTS IN STEADY SIDESLIPS

FIGURE 8 (3.3.6.3)



ROLLING MOMENTS IN STEADY SIDESLIP

FIGURE 9 (3.3.6.3)

Requirement

Paragraph 3.3.6.3.1 Exception of wave-off (go-around). The requirement of 3.3.6.3 may, if necessary, be excepted for wave-off (go-around) if task performance is not impaired and no more than 50 percent of roll control power available to the pilot, and no more than 10 pounds of aileron-control force, are required in a direction opposite to that specified in 3.3.6.3.

Comparison

Flight test data for a steady sideslip maneuver with landing gear and flaps down, representing Category C Flight Phase, are presented in Figure 1 (3.3.6.3.1). Positive effective dihedral (negative C_{ℓ_β}) prevails even for low speed and high angle of attack conditions. Left aileron control deflection and force which produce left aileron position accompany left sideslip and right aileron control deflection and force which produce right aileron position accompany right sideslip.

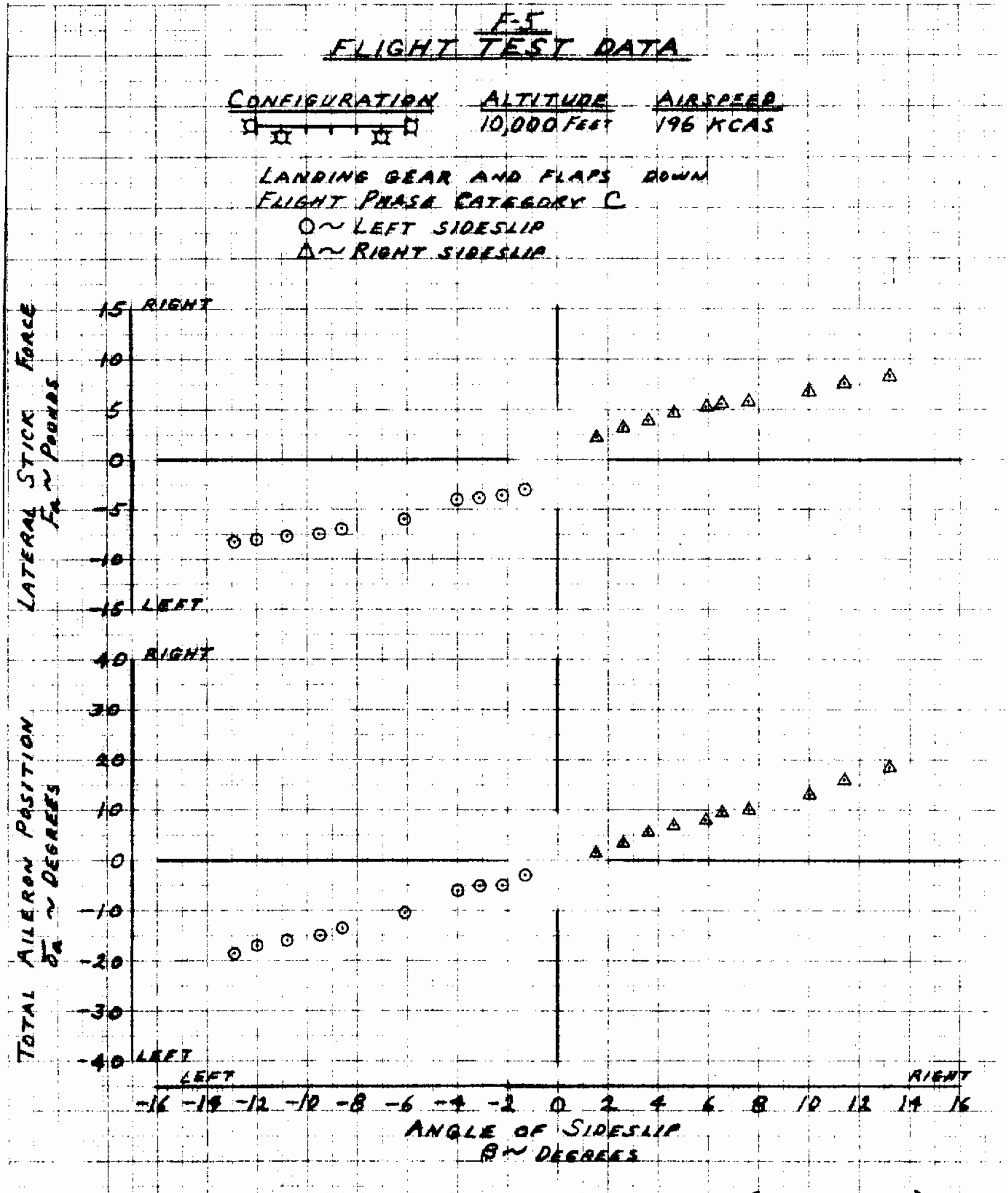
Although the exception for go-around as allowed in this paragraph is not needed for the F-5 airplane, there is no apparent disagreement between this paragraph and the F-5 characteristics.

Resolution

None

Recommendation

None



EXCEPTION FOR WAVEOFF (GO-AROUND)

FIGURE 1 (3.3.6.3.1)

Requirement

Paragraph 3.3.6.3.2 Positive effective dihedral limit. For Levels 1 and 2, positive effective dihedral (right aileron control for right sideslip and left aileron control for left sideslip) shall never be so great that more than 75 percent of roll control power available to the pilot, and no more than 10 pounds of aileron-stick force or 20 pounds of aileron-wheel force, are required for sideslip angles which might be experienced in service employment.

Comparison

The total aileron deflection of the F-5 is ±60 degrees. The aileron deflection needed to perform the maximum rudder sideslips of paragraphs 3.3.6.3 and 3.3.6.3.1 did not exceed ±35 degrees. This is well within the 75 percent of the total roll control power specified in this paragraph to be available to the pilot. In no case does the aileron control force exceed 10 pounds.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.7 Lateral-directional control in cross winds. It shall be possible to take off and land with normal pilot skill and technique in 90-degree cross winds, from either side, of velocities up to those specificed in Table XI. Aileron-control forces shall be within the limits specified in 3.3.4.2, and rudder pedal forces shall not exceed 100 pounds for Level 1 nor 180 pounds for Levels 2 and 3. This requirement can normally be met through compliance with 3.3.7.1 and 3.3.7.2.

TABLE XI. Cross-Wind Velocity

Level	Class	Cross Wind
1 and 2	I	20 knots
	II, III, & IV	30 knots
	Water-based airplanes	20 knots
3	All	one-half the values for Levels 1 and 2

Comparison

Figure 1 (3.3.7) presents flight test time history data of a 50-to 80-degree right 20-knot cross wind takeoff. Flight test data are not available for 30-knot cross wind in takeoff or in landing. The aileron-control and rudder pedal forces needed are sufficiently less than the maximum available for the F-5 that in 30-knot cross wind, it is considered that agreement with the requirements of this paragraph will still prevail in takeoff as well as in landing. The F-5 is allowed to takeoff or land in a 90-degree cross wind of 35 knots (Reference 1).

Resolution

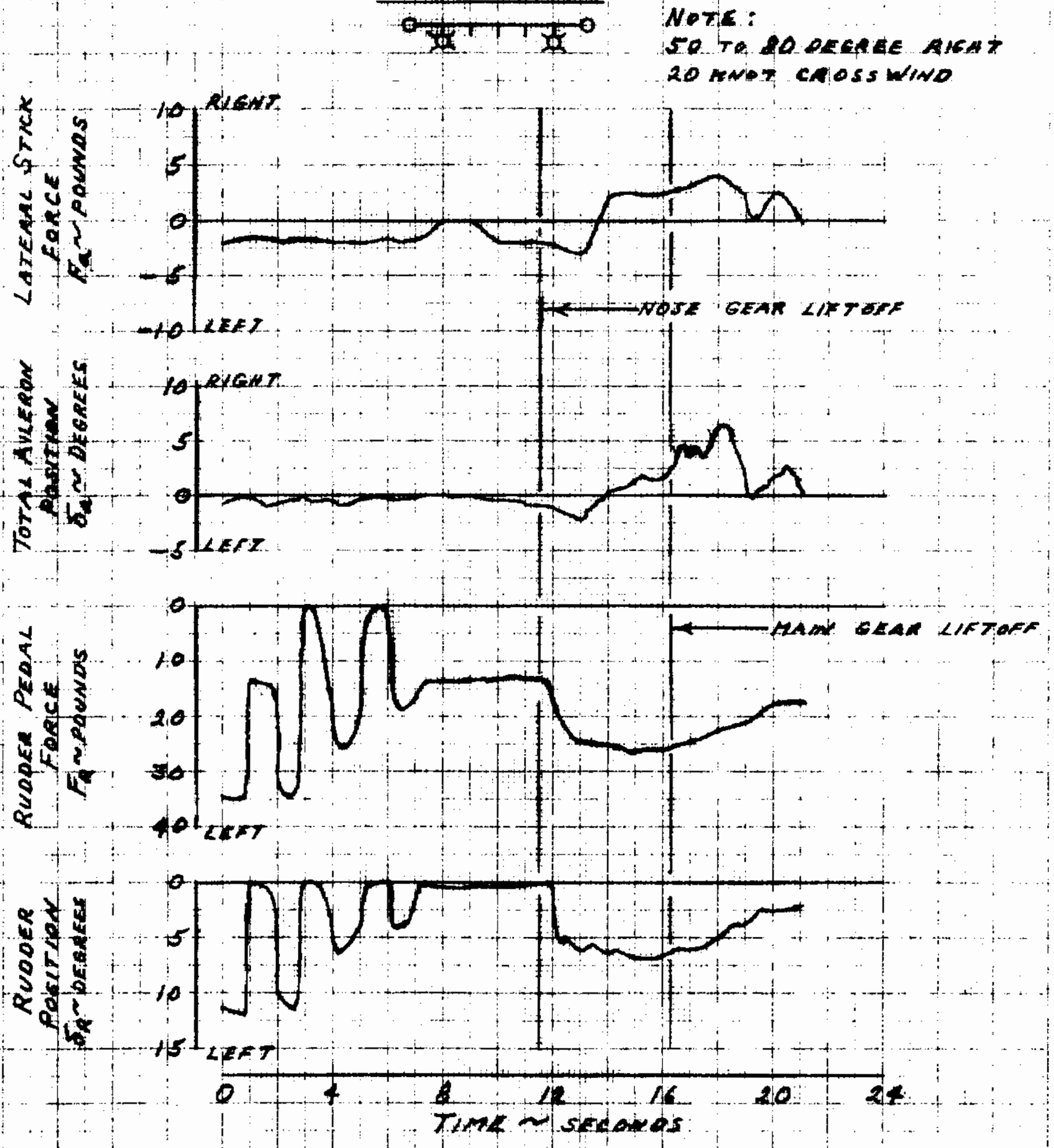
None

Recommendation

None

F-5 FLIGHT TEST DATA

TIME HISTORY OF A CROSSWIND TAKE OFF



LATERAL-DIRECTIONAL CONTROL IN CROSS WINDS

FIGURE 1 (3.3.7)

Requirement

Paragraph 3.3.7.1 Final approach in cross winds. For all airplanes except land-based airplanes equipped with cross-wind landing gear, or otherwise constructed to land in a large crabbed attitude, rudder and aileron-control power shall be adequate to develop at least 10 degrees of sideslip (3.3.6) in the power approach with rudder pedal forces not exceeding the values specified in 3.3.7. For Level 1, aileron control shall not exceed either 10 pounds of force or 75 percent of control power available to the pilot. For Levels 2 and 3, aileron-control force shall not exceed 20 pounds.

Comparison

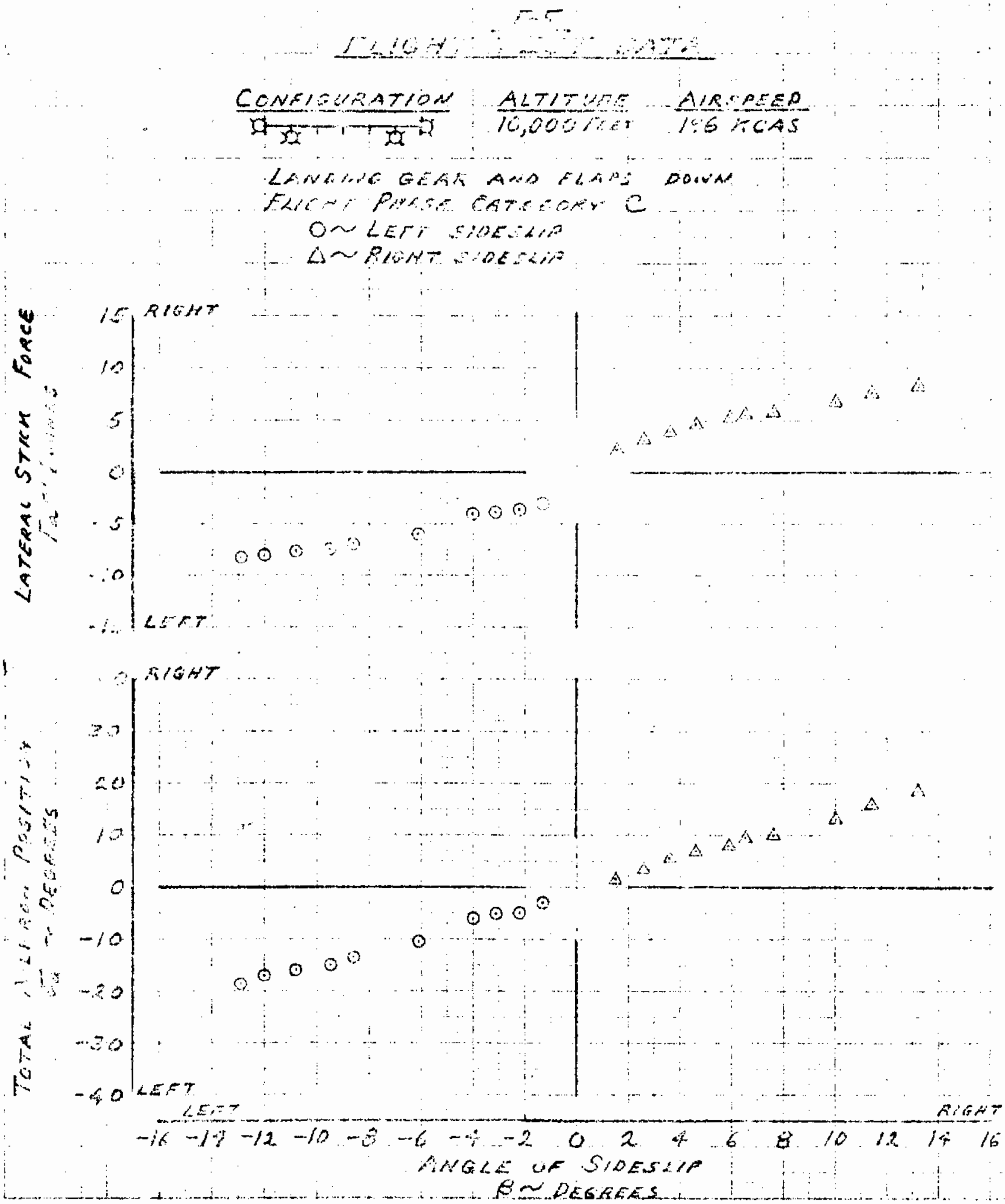
The F-5 airplane has been structurally designed and qualified in flight test to land in a large crabbed attitude and it is so recommended in Reference 1. Nevertheless, data are presented to supplement validation of this paragraph by comparing the capability of the F-5 to generate sideslip angles in the power approach flight phase with the requirements of this paragraph. Figures 1 (3.3.7.1) and 2 (3.3.7.1) present flight test data of a full rudder deflection steady sideslip maneuver. Maximum sideslip angles of ±13 degrees were attained with aileron control forces and rudder pedal forces less than 10 pounds and 80 pounds, respectively. Only 30 percent of the control power available to the pilot was utilized.

Resolution

None

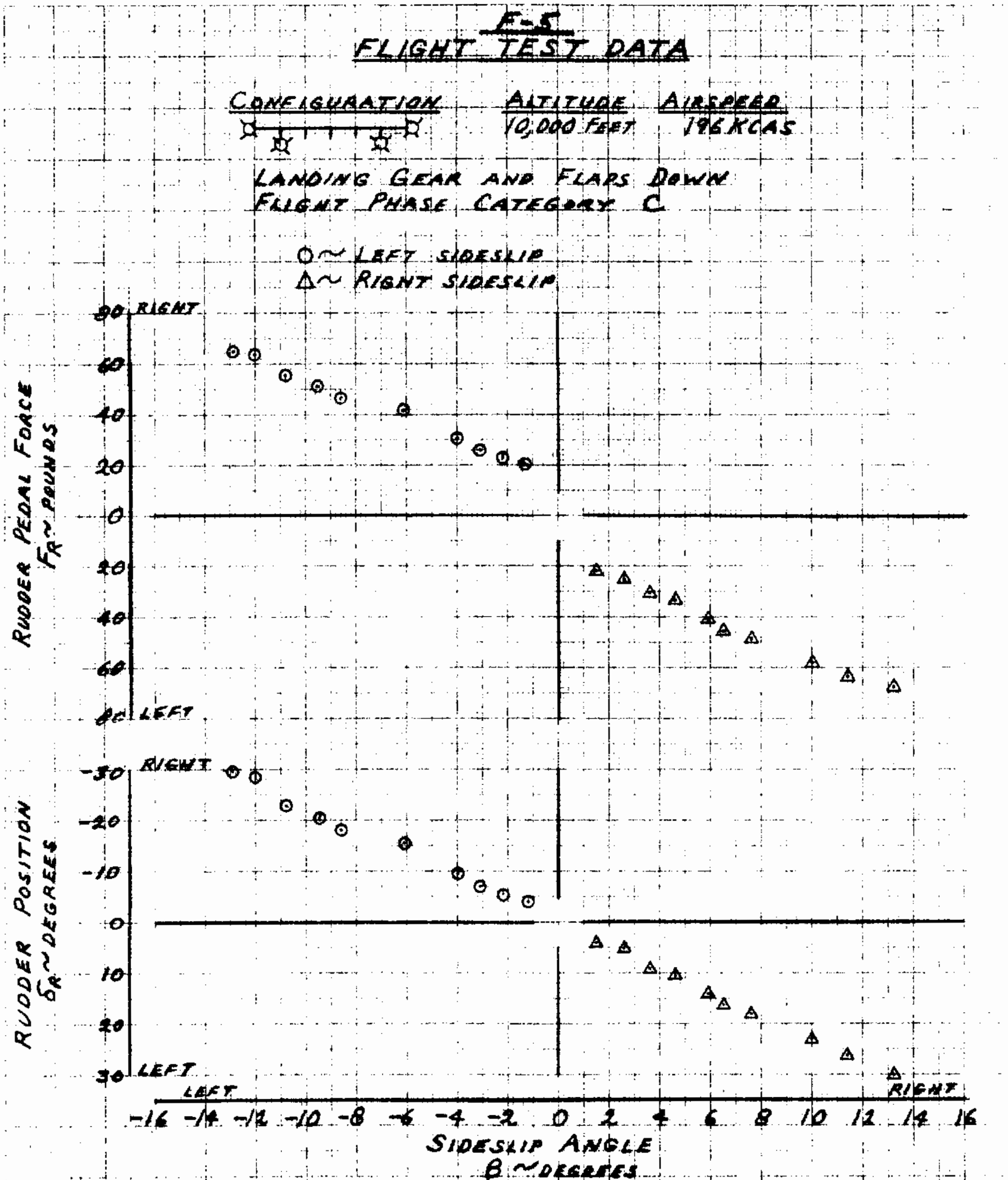
Recommendation

None



FINAL APPROACH IN CROSS WINDS

FIGURE 1 (3.3.7.1)



FINAL APPROACH IN CROSS WINDS

FIGURE 2 (3.3.7.1)

Requirement

Paragraph 3.3.7.2 Takeoff run and landing rollout in cross winds. Rudder and aileron-control power, in conjunction with other normal means of control, shall be adequate to maintain a straight path on the ground or other landing surface. This requirement applies in calm air and in cross winds up to the values specified in table XI with cockpit control forces not exceeding the values specified in 3.3.7.

Comparison

Figure 1 (3.3.7) presents a time history plot of a cross wind takeoff. The wind was 50 degrees to 80 degrees from the right at a velocity of 20 knots. The pilot was able to maintain a straight path on the ground using rudder pedal forces and aileron control forces of less than 40 pounds and 5 pounds, respectively, in agreement with this paragraph. The speed range prevailing during a landing rollout is within the speed range prevailing during a takeoff run. Hence the results presented for the takeoff are considered representative for the landing.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.7.2.1 Cold- and wet-weather operation. The requirements of 3.3.7.2 apply on wet runways for all airplanes, and on snow-packed and icy runways for airplanes intended to operate under such conditions. If compliance is not demonstrated under these adverse runway conditions, directional control shall be maintained by use of aerodynamic controls alone at all airspeeds above 50 knots for Class IV airplanes and above 30 knots for all others. For very slippery runways, the requirement need not apply for cross-wind components at which the force tending to blow the airplane off the runway exceeds the opposing tire-runway frictional force with the tires supporting all of the airplane's weight.

Comparison

No quantitative data are available for comparison with this paragraph. However, a total of 17 flights were flown by an F-5 airplane in Alaska for the Category II evaluation. Thirteen of these were flown in an environment where the ground temperature at takeoff ranged from -14° F to -48°F. The takeoffs and landings were made on an icy, snow-packed runway. As the results of this test, the following qualitative comments concerning the ground handling characteristics were obtained.

1. The effectiveness of nose steering was satisfactory at normal taxi speed although differential braking was required to supplement directional control.

2. Handling characteristics of the airplane during takeoff ground roll in a cold climate were not noticeably different from those in normal temperatures.

3. Directional control was affected on most landings by the main landing gear struts not compressing equally. The nose-wheel steering was ineffective during the high-speed portion of the landing roll.

The primary control was main wheel brakes which were only partially effective on icy portions of the runway, but became very effective on the packed snow. Directional control remained an annoyance until the main gear struts had returned to an even extension.

The above pilot comments were based on the tests conducted in calm air. No severe cross wind takeoffs or landings were made. It is, however, considered that the F-5 airplane exhibited acceptable handling characteristics under severely cold weather conditions.

The third comment was attributed to leakage of hydraulic fluid from landing gear shock struts, but not to the basic handling characteristics of the airplane. Corrective measures have been taken since to stop hydraulic fluid leakage. The above information constitutes a summary narrative of qualifying the F-5 in cold- and wet-weather operation.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.7.2.2 Carrier-based airplanes. All carrier-based airplanes shall be capable of maintaining a straight path on the ground without the use of wheel brakes, at airspeeds of 30 knots and above, during takeoffs and landings in a 90-degree cross wind of at least 10 percent $V_S(L)$. Cockpit control forces shall be as specified in 3.3.7.

Comparison

None. The F-5 airplane is not carrier-qualified.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.7.3 Taxiing wind speed limits. It shall be possible to taxi at any angle to a 35-knot wind for Class I airplanes and to a 45-knot wind for Class II, III, and IV airplanes.

Comparison

No data are available to show the maximum wind speed, from any direction, in which the F-5 can taxi. The F-5 operational limit for takeoff and landing is a 35-knot, 90-degree cross wind (Reference 1). Although the 45-knot wind specified in this paragraph cannot be quantitatively validated based on flight test, it appears to be a reasonable requirement with respect to the F-5 airplane.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.8 Lateral-directional control in dives. Rudder and aileron control power shall be adequate to maintain wings level and sideslip zero, without retrimming, throughout the dives and pullouts of 3.2.3.5 and 3.2.3.6. In the Service Flight Envelope, aileron control forces shall not exceed 20 pounds for propeller-driven airplanes nor 10 pounds for other airplanes. Rudder pedal forces shall not exceed 180 pounds for propeller-driven airplanes nor 50 pounds for other airplanes.

Comparison

Figures 1 (3.3.8) and 2 (3.3.8) present time history plots of the dives and pullouts presented in Paragraph 3.2.3.5 and 3.2.3.6. Rudder and aileron control power were adequate to maintain wings level, without retrimming, throughout the dives and pullouts. The roll exhibited near the end of the pullout was planned and induced by the pilot. The sideslip angle was not recorded but remained near zero as evidenced by the near zero rudder applied. Aileron control forces applied were less than 3 pounds during the dive portion of the maneuvers and less than 10 pounds during the pullouts.

Rudder position remained near zero throughout the dive and pullout, and the rudder pedal forces, although not recorded in flight, were just beyond the breakout forces, altogether less than 15 pounds.

Resolution

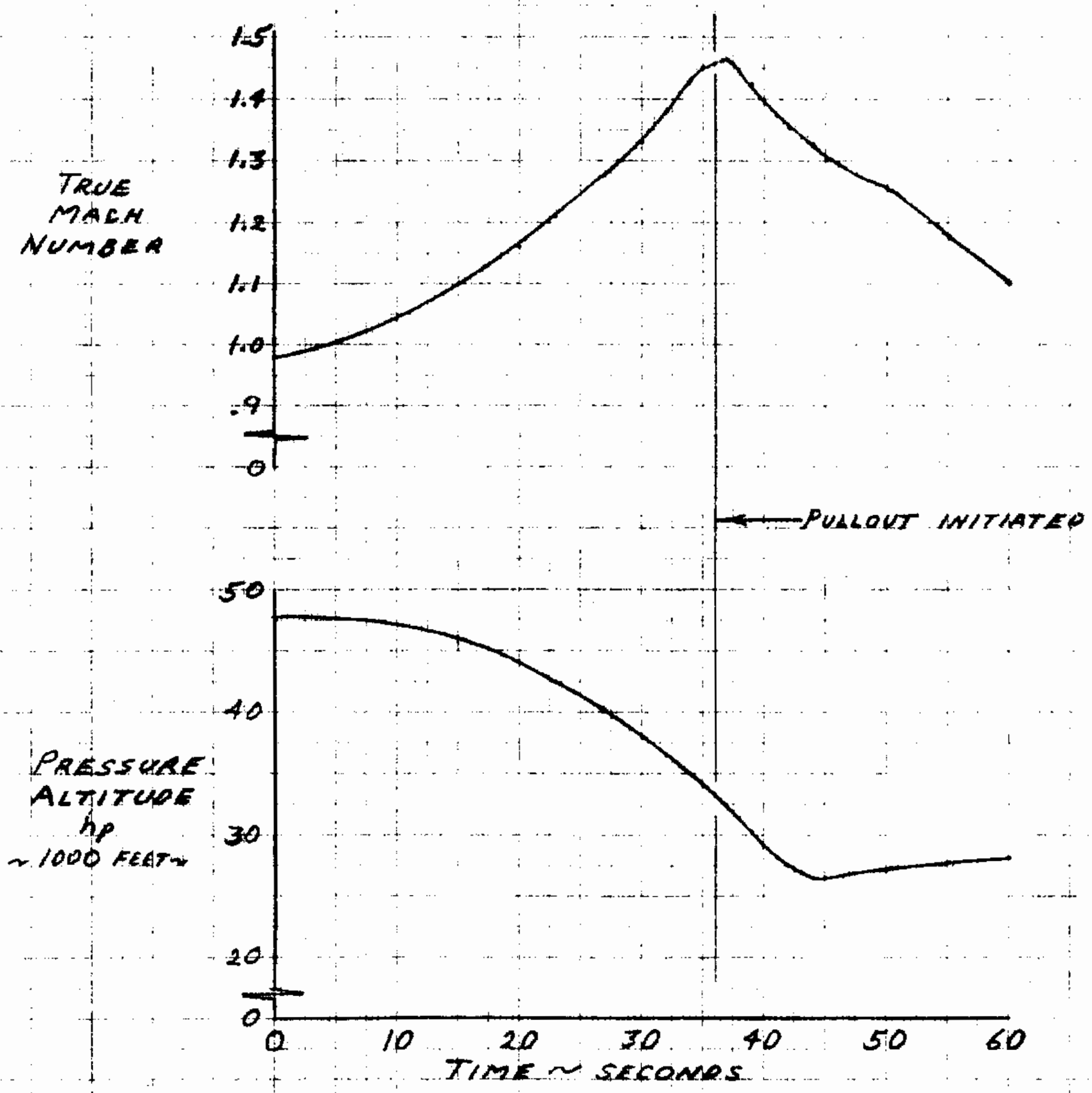
None

Recommendation

None

F-5 FLIGHT TEST DATA

TIME HISTORY OF A DIVE AND SYMMETRICAL PULLOUT
(SERVICE ENVELOPE)
FLIGHT PHASE (CO)

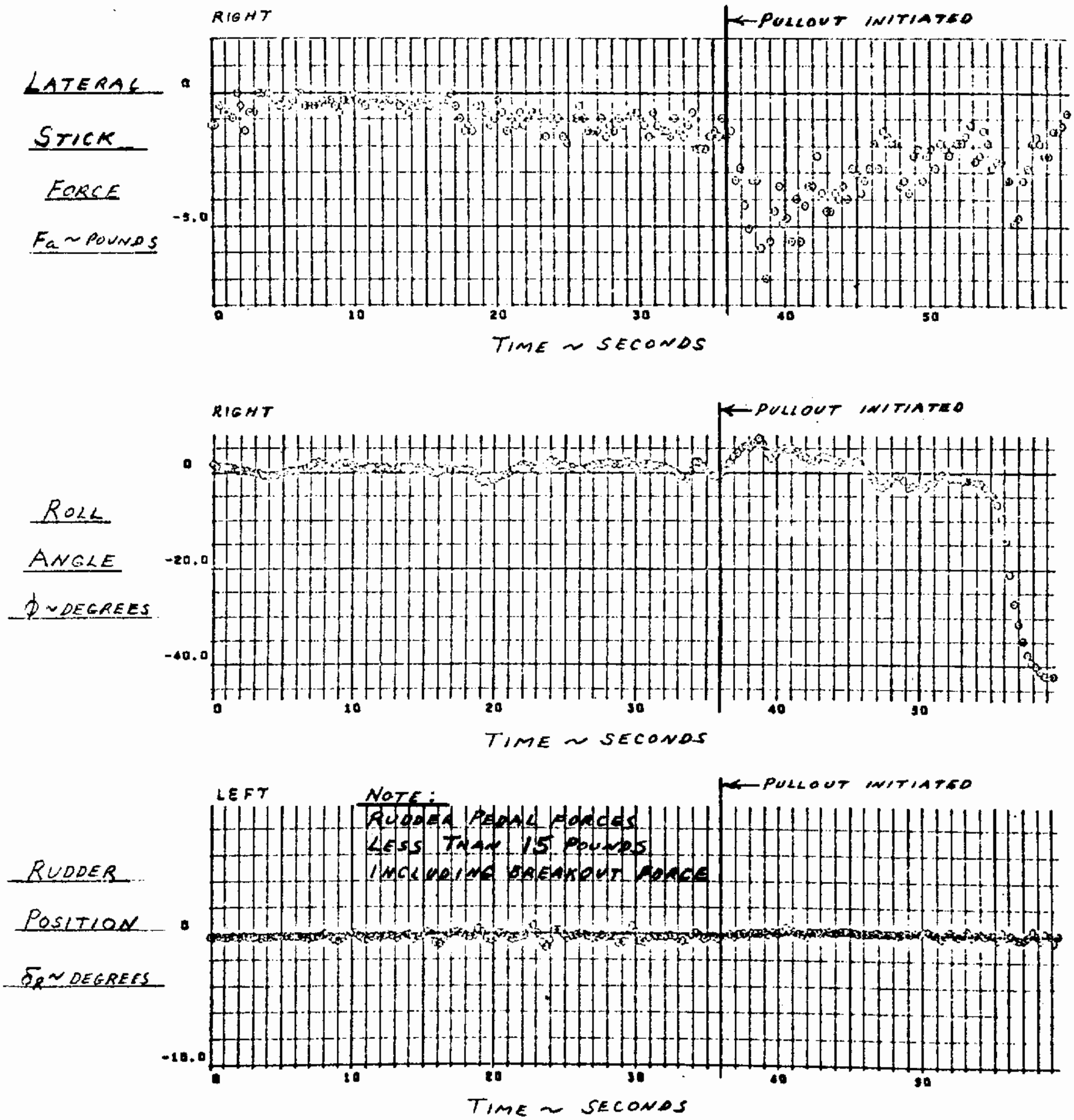


LATERAL-DIRECTIONAL CONTROL IN DIVES

FIGURE 1a. (3.3.8)

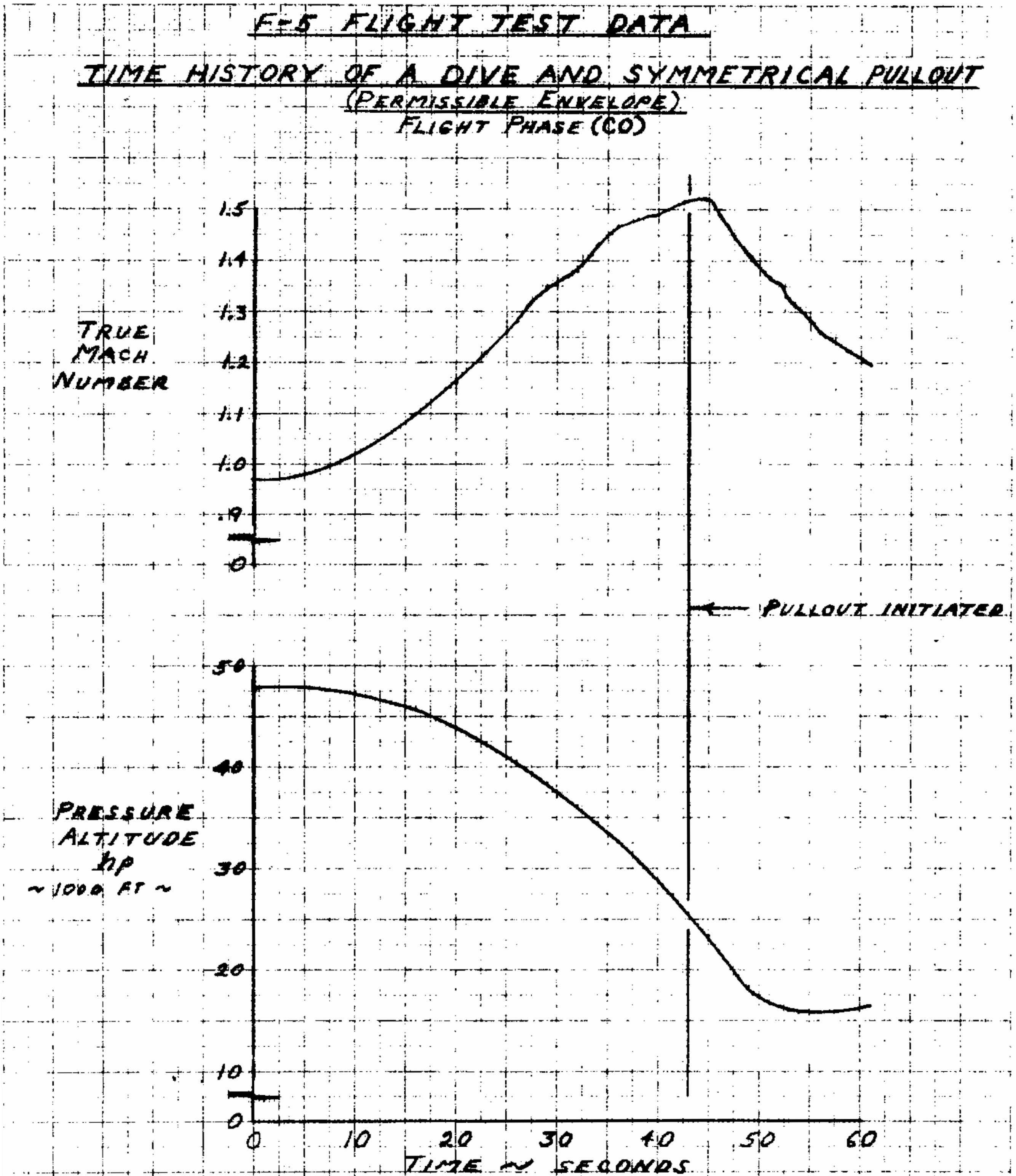
F-5 FLIGHT TEST DATA

TIME HISTORY OF A DIVE AND SYMMETRICAL PULLOUT
(SERVICE ENVELOPE)
FLIGHT PHASE (CO)



LATERAL-DIRECTIONAL CONTROL IN DIVES

FIGURE 1b (3.3.8)



LATERAL-DIRECTIONAL CONTROL IN DIVES

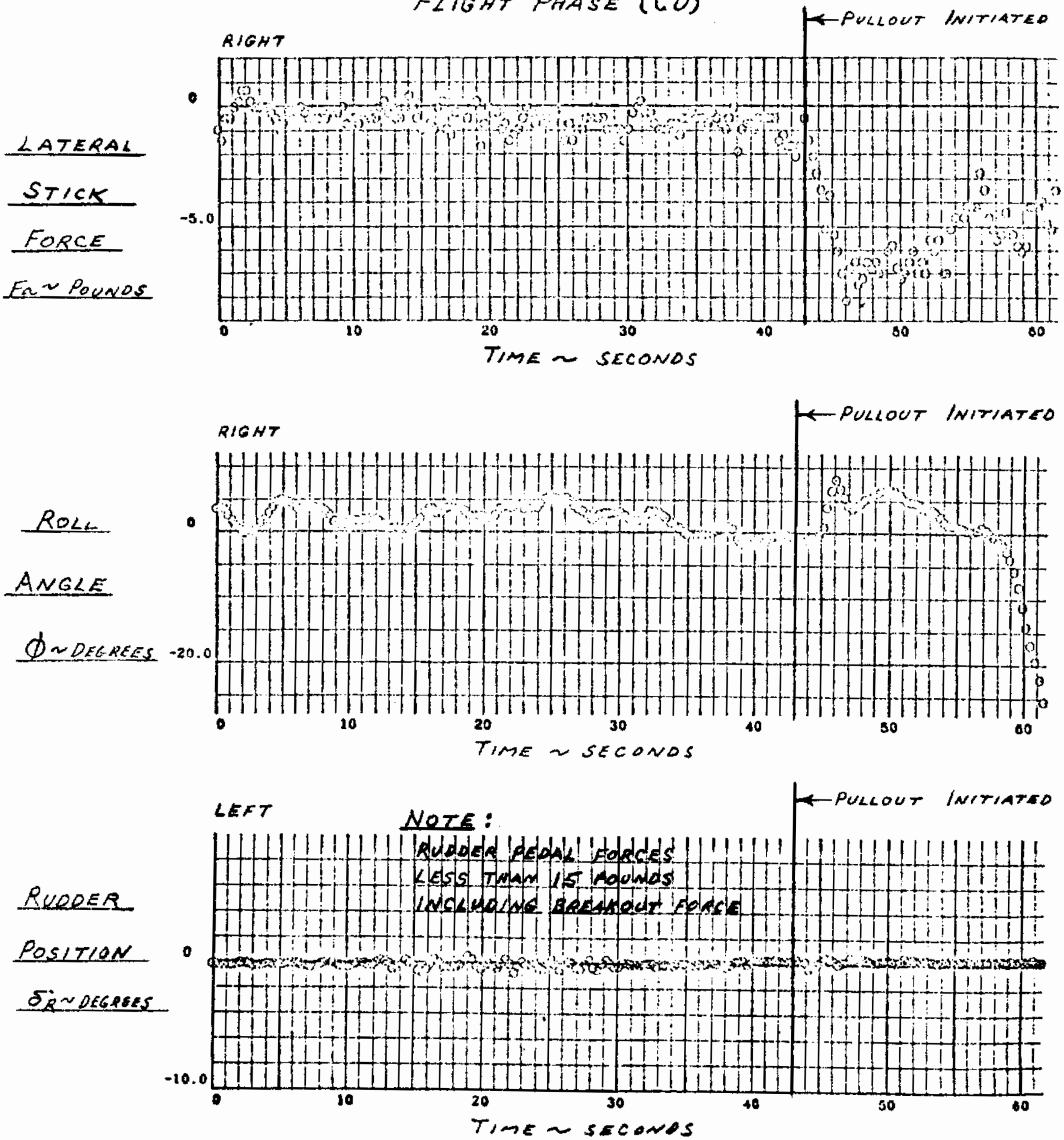
FIGURE 2a (3.3.8)

F-5 FLIGHT TEST DATA

TIME HISTORY OF A DIVE AND SYMMETRICAL PULLOUT

PERMISSIBLE ENVELOPE

FLIGHT PHASE (CO)



LATERAL-DIRECTIONAL CONTROL IN DIVES

FIGURE 2b (3.3.8)

Requirement

Paragraph 3.3.9 Lateral-directional control with asymmetric thrust. Asymmetric loss of thrust may be caused by many factors including engine failure, inlet unstart, propeller failure, or propeller-drive failure. Following sudden asymmetric loss of thrust from any factor, the airplane shall be safely controllable. The requirements of 3.3.9.1 through 3.3.9.4 apply for the appropriate Flight Phases when any single failure or malperformance of the propulsive system, including inlet or exhaust, causes loss of thrust on one or more engines or propellers, considering also the effect of the failure or malperformance on all subsystems powered or driven by the failed propulsive system.

Comparison

Comparison data and discussion are presented in the appropriate succeeding paragraphs.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.9.1 Thrust loss during takeoff run. It shall be possible for the pilot to maintain control of an airplane on the takeoff surface following sudden loss of thrust from the most critical factor. Thereafter, it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than 30 feet from the path originally intended, with rudder pedal forces not exceeding 180 pounds. For the continued takeoff, the requirement shall be met when thrust is lost at speeds from the refusal speed (based on the shortest runway from which the airplane is designed to operate) to the maximum takeoff speed, with takeoff thrust maintained on the operative engine(s), using only elevator, aileron, and rudder controls. For the aborted takeoff, the requirement shall be met at all speeds below the maximum takeoff speed; however, additional controls such as nose wheel steering and differential braking may be used. Automatic devices which normally operate in the event of a thrust failure may be used in either case.

Comparison

Figure 1 (3.3.9.1) presents a diagram of F-5 directional control required to maintain a straight path on the takeoff surface or in flight with one engine inoperative. Nosewheel steering is shown to be effective to 65 knots and differential braking may be used to the minimum takeoff speed, thus exhibiting the capability of effective directional control in aborted takeoffs. Only 40 percent of maximum available rudder is needed to balance the yawing moment induced by the asymmetric thrust (assuming maximum thrust of one engine) at the minimum takeoff speed. At the maximum takeoff speed, only 15 percent of maximum available rudder is needed. The control system is such that the maximum rudder pedal force required is 34 pounds.

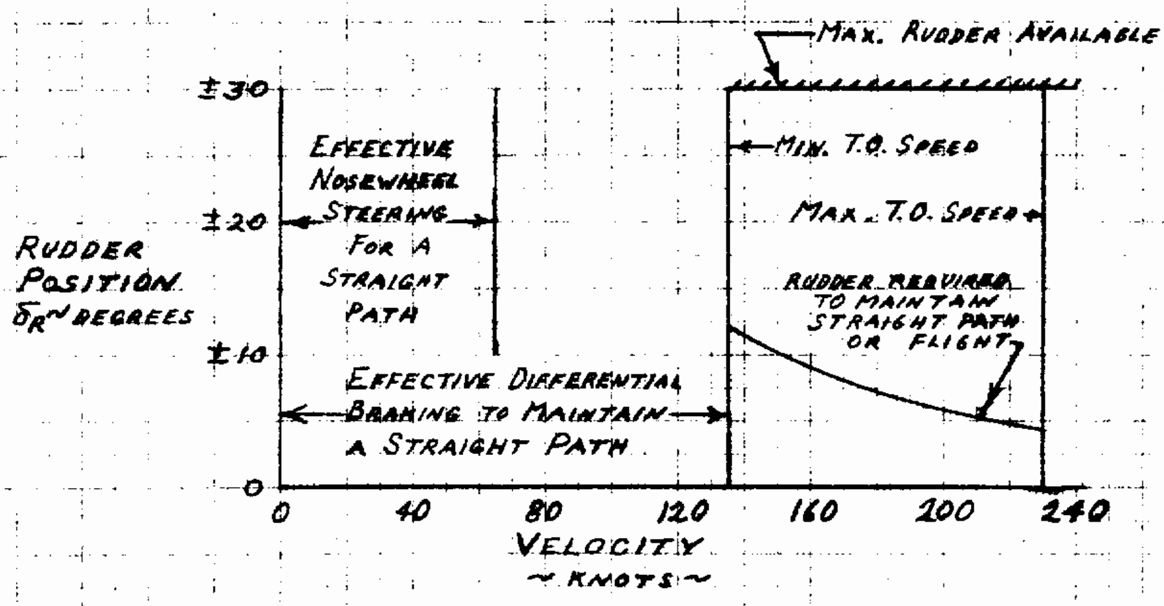
Resolution

None

Recommendation

None

F-5 DIRECTIONAL CONTROL TO MAINTAIN A STRAIGHT PATH ON TAKEOFF SURFACE OR IN FLIGHT WITH ONE ENGINE INOPERATIVE



THRUST LOSS DURING TAKEOFF RUN

FIGURE 1 (3.3.9.1)

Requirement

Paragraph 3.3.9.2 Thrust loss after takeoff. During takeoff, it shall be possible without a change in selected configuration to achieve straight flight following sudden asymmetric loss of thrust from the most critical factor at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climb-out. The rudder pedal force required to maintain straight flight with asymmetric thrust shall not exceed 180 pounds. Aileron control shall not exceed either the force limits specified in 3.3.4.2 or 75 percent of available control power, with takeoff thrust maintained on the operative engine(s) and trim at normal settings for takeoff with symmetric thrust. Automatic devices which normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

Comparison

Figure 1 (3.3.9.2) presents a plot of an asymmetrical power acceleration with rudder position, rudder pedal force, sideslip angle, aileron control force, and roll angle plotted versus calibrated airspeed. Both engines were at maximum power to 187 knots, then the left engine rpm was reduced to idle.

The yawing moment induced by the asymmetric thrust produced a sideslip angle of 2.5 degrees. An initial rudder pedal force of 10 pounds was applied to counteract the asymmetry with a continued force of 5 pounds required to maintain the sideslip angle at zero. The maximum roll angle was 3 degrees and the initial aileron control force applied was 6.5 pounds with a continued force of approximately 2 pounds required to maintain zero roll rate.

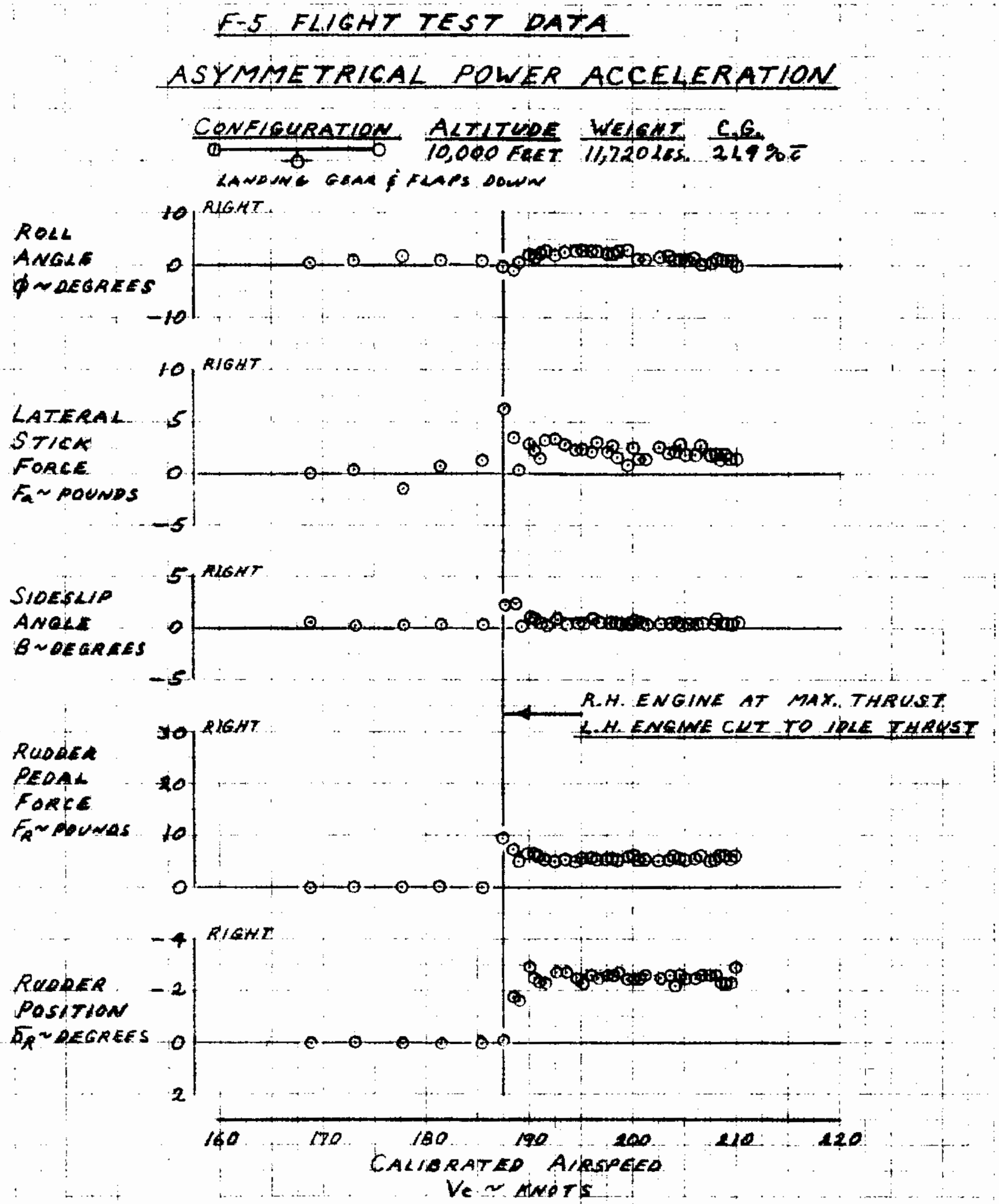
Although this test was not performed at the minimum takeoff velocity, Figure 1 (3.3.9.1) exhibits more than adequate rudder availability.

Resolution

None

Recommendation

None



THRUST LOSS AFTER TAKEOFF

FIGURE 1 (3.3.7.2)

Requirement

Paragraph 3.3.9.3 Transient effects. The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay (3.4.9) of at least 1 second shall be considered.

Comparison

The sudden loss of thrust on one engine of the F-5 airplane will not result in a condition that the pilot cannot counteract through normal control application due to the close proximity of the engines to the plane of symmetry of the airplane. The F-5 response to this sudden asymmetry is sufficiently slow to allow the pilot ample time to take proper corrective action. Although no flight test time history data are available to validate the one-second time delay, it is considered to be a reasonable requirement.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.3.9.4 Asymmetric thrust-rudder pedals free. The static directional stability shall be such that at all speeds above 1.4 V_{min}, with asymmetric loss of thrust from the most critical factor while the other engine(s) develop normal rated thrust, the airplane with rudder pedals free may be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the failure. Aileron-control forces shall not exceed the Level 2 upper limits specified in 3.3.4.2 for Levels 1 and 2 and shall not exceed the Level 3 upper limits for Level 3.

Comparison

Figure 1 (3.3.9.4) presents a plot of total aileron position required for the F-5 to maintain straight flight with one engine inoperative and the other engine developing normal rated thrust. Analytical data were obtained for altitudes of 10,000, 20,000, and 30,000 feet at speeds from 1.4 V_{min} to the boundary of the Operational Flight Envelope. These data were calculated by solving a three by three matrix using F-5 basic aerodynamic data. The aileron control forces corresponding to the maximum total aileron position required for the F-5 are less than 5 pounds, in agreement with the requirements of this paragraph.

Resolution

None

Recommendation

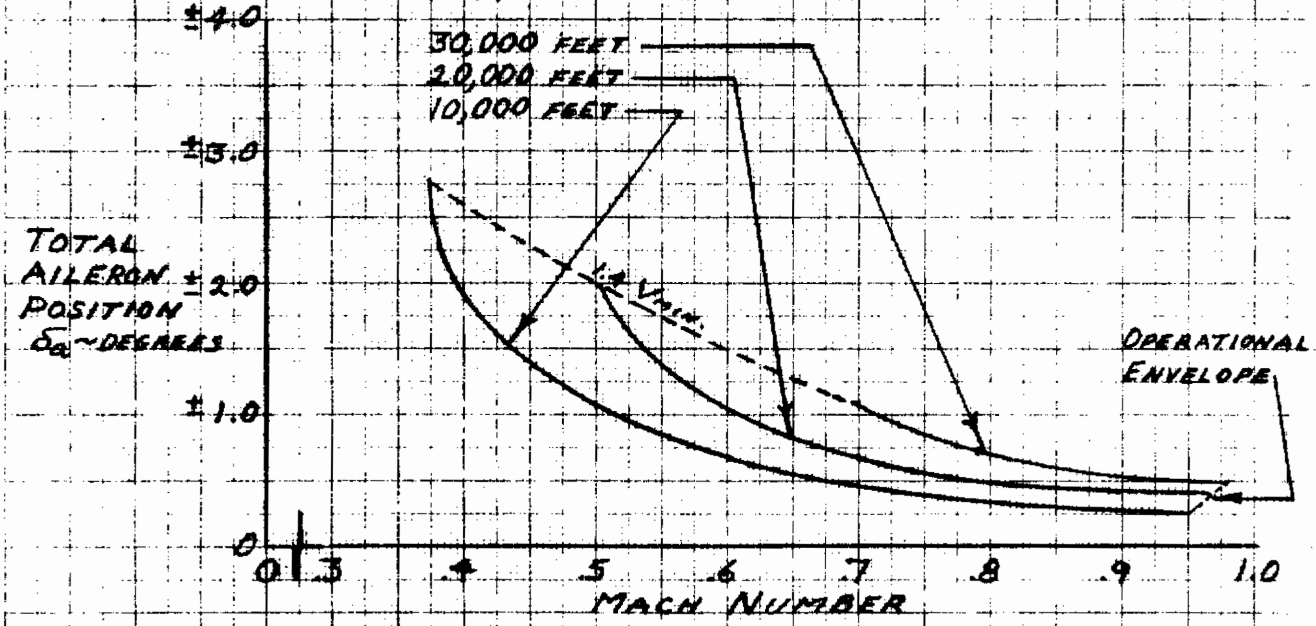
None

AILERON CONTROL FOR STRAIGHT FLIGHT
F-5

AILERON CONTROL FORCES

SPEC REQUIREMENT ≦ 30 POUNDS

F-5 AILERON CONTROL FORCES FOR AILERON POSITION PRESENTED IN FIGURE BELOW < 5 POUNDS



ASYMMETRIC THRUST - RUDDER PEDALS FREE

FIGURE 1 (3.3.9.4)

Requirement

Paragraph 3.3.9.5 Two engines inoperative. With any engine initially failed, it shall be possible upon failure of the most critical remaining engine to stop the transient motion at the one-engine-out speed for maximum range, and thereafter to maintain straight flight from that speed to the speed for maximum range, with both engines failed. In addition, it shall be possible to effect a safe recovery at any service speed above Vo_{min} (CL) following sudden simultaneous failure of the two critical failing engines.

Comparison

None. The F-5 is a two-engined airplane.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4 Miscellaneous flying qualities

Paragraph 3.4.1 Approach to dangerous flight conditions. Dangerous conditions may exist where the airplane should not be flown. When approaching these flight conditions, it shall be possible by clearly discernible means for the pilot to recognize the impending dangers and take preventive action. Final determination of the adequacy of all warning of impending dangerous flight conditions will be made by the procuring activity, considering functional effectiveness and reliability. Devices may be used to prevent entry to dangerous conditions only if the criteria for their design, and the specific devices, are approved by the procuring activity.

Paragraph 3.4.1.1 Warning and indication. Warning or indication of approach to a dangerous condition shall be clear and unambiguous. For example, a pilot must be able to distinguish readily among stall warning (which requires pitching down or increasing speed), Mach buffet (which may indicate a need to decrease speed), and normal airplane vibration (which indicates no need for pilot action). If a warning or indication device is required, functional failure of the device shall be indicated to the pilot.

Paragraph 3.4.1.2 Prevention. As a minimum, dangerous-condition-prevention devices shall perform their function whenever needed, but shall not limit flight within the Operational Flight Envelope. Hazardous operation, normal or inadvertent, shall never be possible. For Levels 1 and 2, neither hazardous nor nuisance operation shall be possible.

Comparison

No known dangerous flight conditions exist for the F-5 airplane to warrant the design or use of warning or entry preventive devices. Consequently, it has not been necessary for the F-5 to possess or employ such devices.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.2 Stalls. The requirements of 3.4.2 through 3.4.2.4.1 are to assure that the airflow separation induced by high angle of attack, which causes loss of aerodynamic lift or control about any one axis, does not result in a dangerous or mission-limiting condition. The stall is further defined in terms of speed and angle of attack in 6.2.2 and 6.2.5 respectively.

Comparison

The F-5 stalls do not result in dangerous or mission-limiting conditions, Discussion of the F-5 stall characteristics and flight test data are presented in the appropriate succeeding paragraphs.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.2.1 Required conditions. The requirements for stall characteristics apply for all Airplane Normal States in straight unaccelerated flight, and in turns and pullups with normal acceleration up to $n_{o_{max}}$. Specifically, the Airplane Normal States associated with the configurations, throttle settings, and trim settings of 6.2.2 shall be investigated; also, the requirements apply to Airplane Failure States that affect stall characteristics.

Comparison

The F-5 stall characteristics were investigated in flight test. The maneuvers conducted consisted of straight unaccelerated flight, wind-up turns and pullups to full aft stick or n_L, whichever came first. Various external loading configurations were tested accounting for Airplane Normal States. Some flight tests were conducted with Airplane Failure States consisting of an inoperative augmentation system. The succeeding appropriate paragraphs present the flight test results and discussion.

Resolution

The requirement specifies that the turns and pullups are to be performed only to $n_{o_{max}}$. In air combat, fighter pilots will pull to full aft stick either for evasive action or to gain advantage over the threat fighter. The fighter pilot is in fact generating maximum angle of attack and, consequently, maximum drag to accomplish this objective. This will take him beyond the stall angle of attack. Although he may not generate more load factor, he will still apply more aft stick to overotate.

In these conditions the airplane must not possess undesirable or poor flight characteristics. If it does, the pilot in air combat will be limited in gaining superiority over the threat airplane.

Recommendation

Based on the resolution presented, it is recommended that "$n_{o_{max}}$" appearing at the end of the first sentence be replaced by the following:

"full aft displacement of the elevator control, or n_L, whichever comes first."

Requirement

Paragraph 3.4.2.2 Stall warning requirements. The stall approach shall be accompanied by an easily perceptible warning. Acceptable stall warning for all types of stalls consists of shaking of the cockpit controls, buffeting or shaking of the airplane, or a combination of both. The onset of this warning shall occur within the ranges specified in 3.4.2.2.1 and 3.4.2.2.2 but not within the Operational Flight Envelope. The increase in buffeting intensity with further increase in angle of attack shall be sufficiently marked to be noted by the pilot. This warning may be provided artificially only if it can be shown that natural stall warning is not feasible. These requirements apply whether V_S is as defined in 6.2.2 or as allowed in 3.1.9.2.1.

Comparison

The F-5 has no stall limitations and, as a result of the extremely stable stall characteristics (both accelerated and unaccelerated), there is no need to provide artificial warning to the pilot about impending stall. This reflects the F-5 characteristics and it is not intended to imply that stall warnings are not necessary for Class IV airplanes.

A major contributing factor to the F-5 acceptable stall characteristics is the large increase in longitudinal stability at high angles of attack which precludes violent pitching motion. Instead there is a mushing motion which is accompanied by increasing sink rates as the stall progresses. A high sink rate can be maintained with little pitching motion as the airplane is held nose high with full aft stick.

The irreversible power control system allows no changes in stick force except due to stick motion and load factor; therefore, no force changes due to aerodynamic feedback are encountered in the stall. The sink rate in the stalled condition can be terminated easily by releasing aft stick pressure and adding power.

Aerodynamic flow separation yields a very adequate and clear stall warning at subsonic speeds. Buffet onset occurs before limit load factor is reached and the buffeting intensity increases as load factor or angle of attack increases. The pilot is amply warned; nevertheless, buffet is not a warning of impending danger for the F-5 due to the absence of both spin entry and pitch-up tendencies at stall.

Good lateral control allows the maximum lift capability of the wing to be used. In the transonic speed range at lower altitudes, limit load factor is reached before buffet onset; at high altitudes, buffet is experienced before reaching limit load factor. In general, above Mach 1.0 the maximum load factor obtainable is limited by available horizontal tail deflection on the F-5, and buffet is not encountered.

In summary, aerodynamic stall warnings prevail for the F-5, thus negating the need for artifical warning. Agreement exists between the F-5 and this paragraph. Flight test buffet onset and stall data are presented in the succeeding two paragraphs.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.2.2.1 Warning speed for stalls at 1 g normal to the flight path. Warning onset for stalls at 1 g normal to the flight path shall occur between the following limits:

Flight Phase	Minimum Stall Warning Speed	Maximum Stall Warning Speed
Approach	Higher of $1.05V_S$ or V_S +5 knots	Higher of $1.10V_S$ or V_S +10 knots
All Other	Higher of $1.05V_S$ or V_S +5 knots	Higher of $1.15V_S$ or V_S +15 knots

Comparison

Figures 1 (3.4.2.2.1) and 2 (3.4.2.2.1) present the stall warning characteristics of the F-5. Lift coefficient (C_L) is plotted versus Mach number for Flight Phase Category A and versus center of gravity location (c.g.) for Flight Phase Category C. The stall warning buffet onset occurs on the F-5 above 0.6 Mach number at a higher percentage of V_S than the specification allows for Category A. For the Category C case, stall buffet onset occurs at a higher percentage of V_S than the specification allows throughout the entire c.g. range.

Partial disagreement is exhibited for Category A and total disagreement is exhibited for Category C due to the early buffet onset. However, the intensity of buffet at its onset is not restrictive and is sufficiently mild that the effectiveness and flying qualities of the F-5 are not severely impaired as C_L is increased up to and through stall. As C_L is increased, the buffet intensity increases, serving as an adequate aerodynamic warning of impending stall.

Due to the mild stall characteristics of the F-5, no pilot complaints have been registered regarding the early buffet onset or inadequacy of stall warning. Although a disagreement exists between the requirements of this paragraph and F-5 characteristics, the requirements of this paragraph are considered acceptable since other airplanes may not exhibit mild stall characteristics but rather dangerous stall characteristics, making it imperative to have stall warnings in exact compliance with the requirements of this paragraph.

Resolution

None

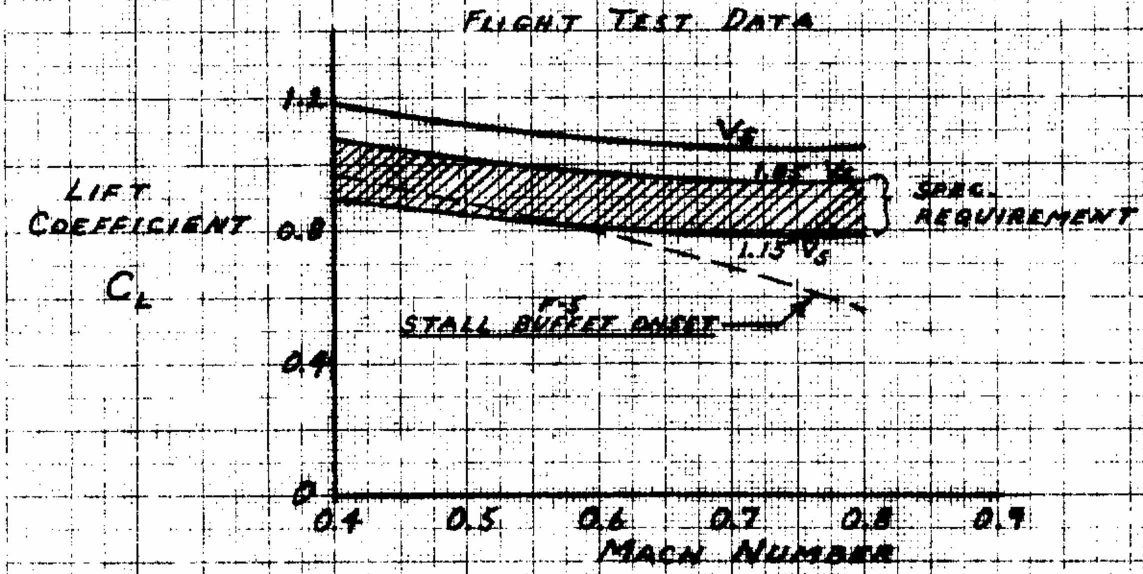
Recommendation

None

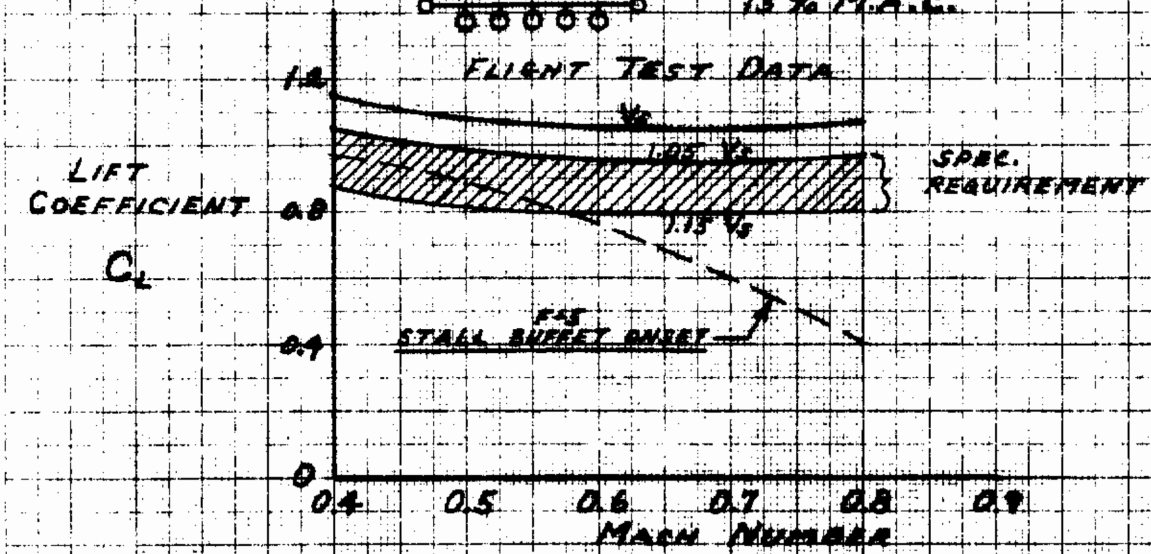
F-5 STALL WARNING CHARACTISTICS

FLIGHT PHASE CATEGORY A

CONFIGURATION — C.G. POSITION 13% M.A.C



CONFIGURATION — C.G. POSITION 13% M.A.C.

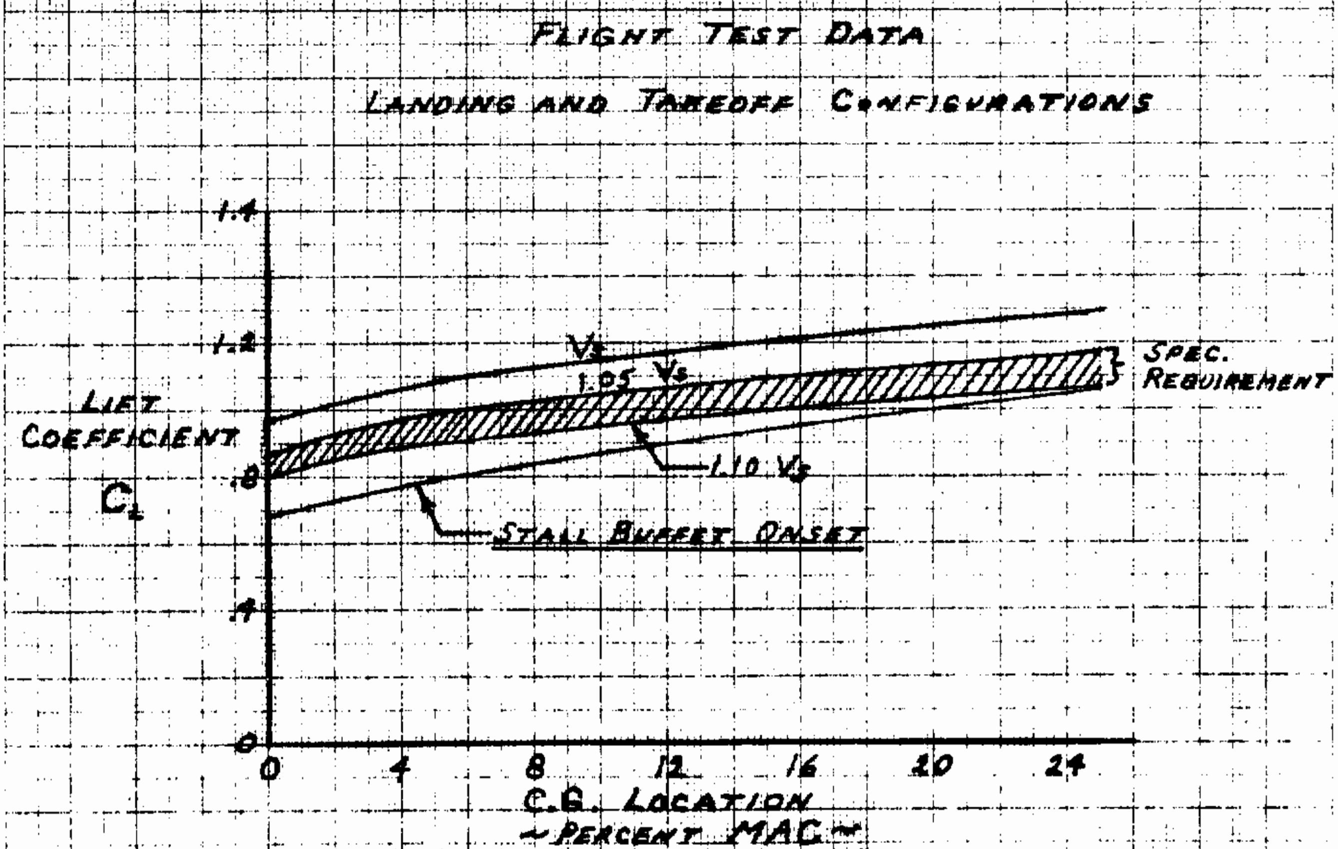


WARNING SPEED FOR STALLS AT 1g NORMAL TO THE FLIGHT PATH

FIGURE 1 (3.4.2.2.1)

F-5 STALL WARNING CHARACTERISTICS

FLIGHT PHASE CATEGORY C



WARNING SPEED FOR STALLS AT 1g NORMAL TO THE FLIGHT PATH

FIGURE 2 (3.4.2.2.1)

Requirement

Paragraph 3.4.2.2.2 Warning range for acclerated stalls. Onset of stall warning shall occur outside the Operational Flight Envelope associated with the Airplane Normal State and within the following angle-of-attack ranges:

Flight Phase	Minimum Stall Warning Angle of Attack	Maximum Stall Warning Angle of Attack
Approach	$\alpha_o + 0.82\ (\alpha_s - \alpha_o)$	$\alpha_o + 0.90\ (\alpha_s - \alpha_o)$
All Other	$\alpha_o + 0.75\ (\alpha_s - \alpha_o)$	$\alpha_o + 0.90\ (\alpha_s - \alpha_o)$

where α_s is the stall angle of attack and α_o is the angle of attack for zero lift (α_s is defined in 6.2.5; α_o may be estimated from wind tunnel tests).

Comparison

Figure 1 (3.4.2.2.2) presents the accelerated stall warning characteristics of the F-5 in the form of angle of attack versus Mach number and normal load factor transients versus angle of attack for a representative flight condition. The normal load factor transients (peak-to-peak amplitudes) are presented to illustrate the intensity of buffeting. These transients exhibited by the buffeting envelope are only felt by the pilot when nearing $\Delta n_z = 0.08$ at approximately 9 degrees angle-of-attack. Stall warning buffet onset occurs at a lower angle of attack than the specification requires, exhibiting similar disagreement as in paragraph 3.4.2.2.1. Consequently, the discussion regarding stall in the comparison part of that paragraph applies equally for this paragraph.

Resolution

None

Recommendation

None

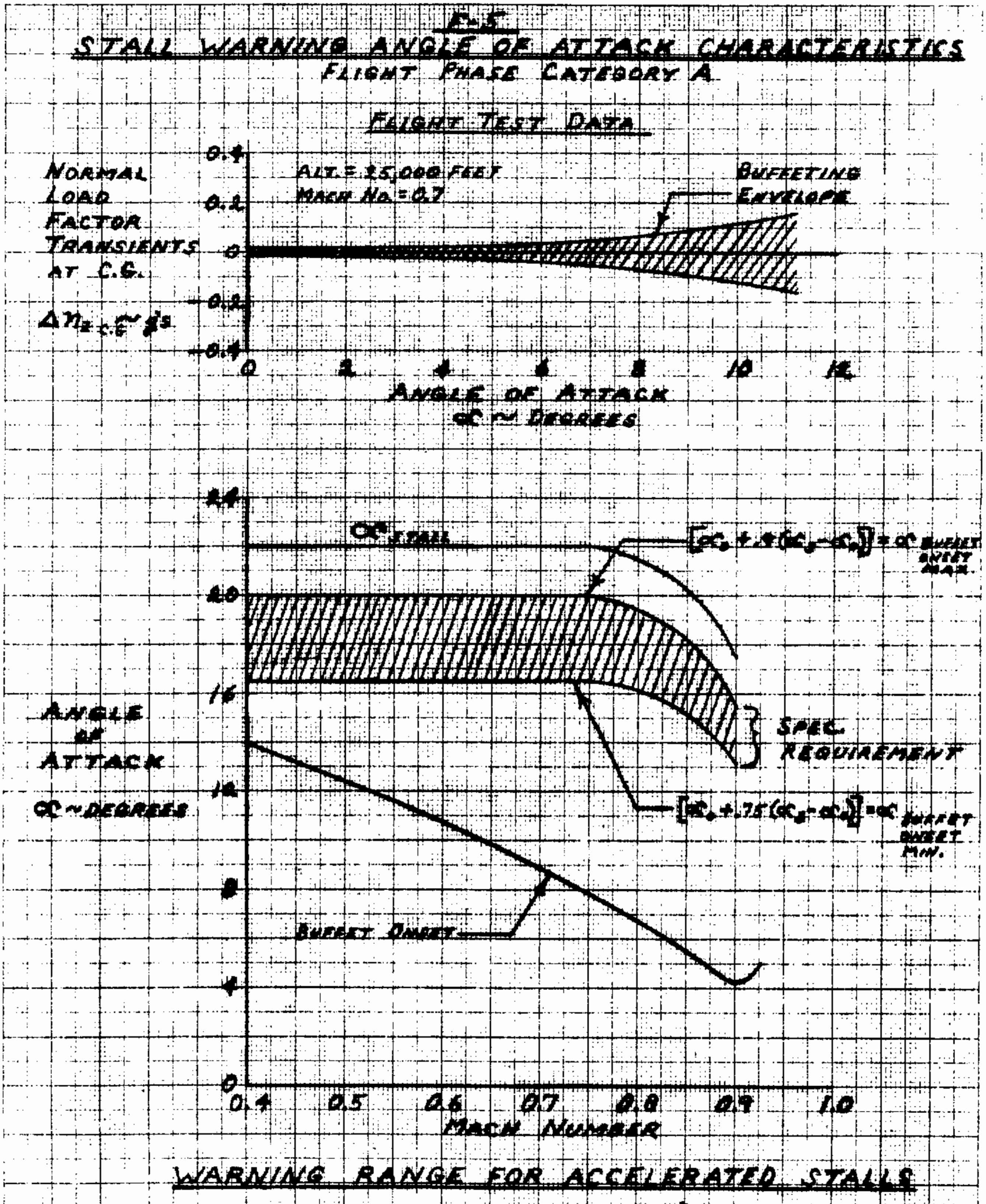


FIGURE 1 (3.4.2.2.2)

Requirement

Paragraph 3.4.2.3 Stall Characteristics. In the unaccelerated stalls of 3.4.2.1, the airplane shall not exhibit uncontrollable rolling, yawing, or downward pitching at the stall in excess of 20 degrees for Classes I, II and III, or 30 degrees for Class IV airplanes. It is desired that no pitch-up tendencies occur in unaccelerated or accelerated stalls. In unaccelerated stalls, mild nose-up pitch may be acceptable if no elevator control force reversal occurs and if no dangerous, unrecoverable, or objectionable flight conditions result. A mild nose-up tendency may be acceptable in accelerated stalls if the operational effectiveness of the airplane is not comprised and:

a. The airplane had adequate stall warning

b. Elevator effectiveness is such that it is possible to stop the pitch-up promptly and reduce the angle of attack, and

c. At no point during the stall, stall approach, or recovery does any portion of the airplane exceed structural limit loads.

The requirements apply to all stalls resulting from rates of speed reduction up to 4 knots per second. The stall characteristics will be considered unacceptable if a spin is likely to result.

Comparison

Figures 1 (3.4.2.3) through 8 (3.4.2.3) present flight test time history data from unaccelerated and accelerated stalls for a clean configuration, four- and five-store configurations. The accelerated stalls were performed using the wind-up turn and symmetrical pullup methods to full aft stick or n_L whichever came first. The straight flight unaccelerated stalls were performed also to full aft stick.

No divergent yawing, rolling, or downward pitching are exhibited at the stall and no pitchup, spin tendencies or dangerous flight conditions resulted. A favorable comparison with the paragraph requirements is exhibited. Table 1 (3.4.2.3) presents a summary of the flight test data.

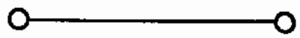
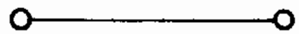
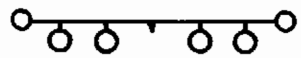
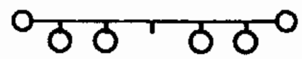
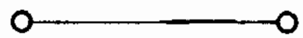
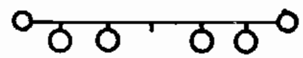
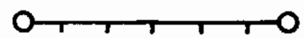
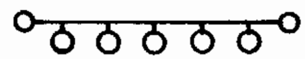
Resolution

None

Recommendation

None

F-5 FLIGHT TEST DATA SUMMARY

FIGURE	CONFIGURATION	MANEUVER	ALTITUDE (FEET)	M.N.	C.G. (% MAC)	FLIGHT PHASE	STABILITY AUGMENTER
1		1 g Stall	10,000 ft	---	15.65	CAT A	Off
2		1 g Stall	10,000 ft	---	15.52	CAT C	Off
3		1 g Stall	10,000 ft	---	18.10	CAT A	Off
4		1 g Stall	10,000 ft	---	14.91	CAT C	Off
5		Wind-up Turn	10,000 ft	0.6	22.9	CAT A	Off
6		Wind-up Turn	10,000 ft	0.6	16.32	CAT A	Off
7		Wind-up Turn	17,000 ft	0.95	13.81	CAT A	Off
8		Pull-up	5,000 ft	0.85	19.85	CAT A	Off

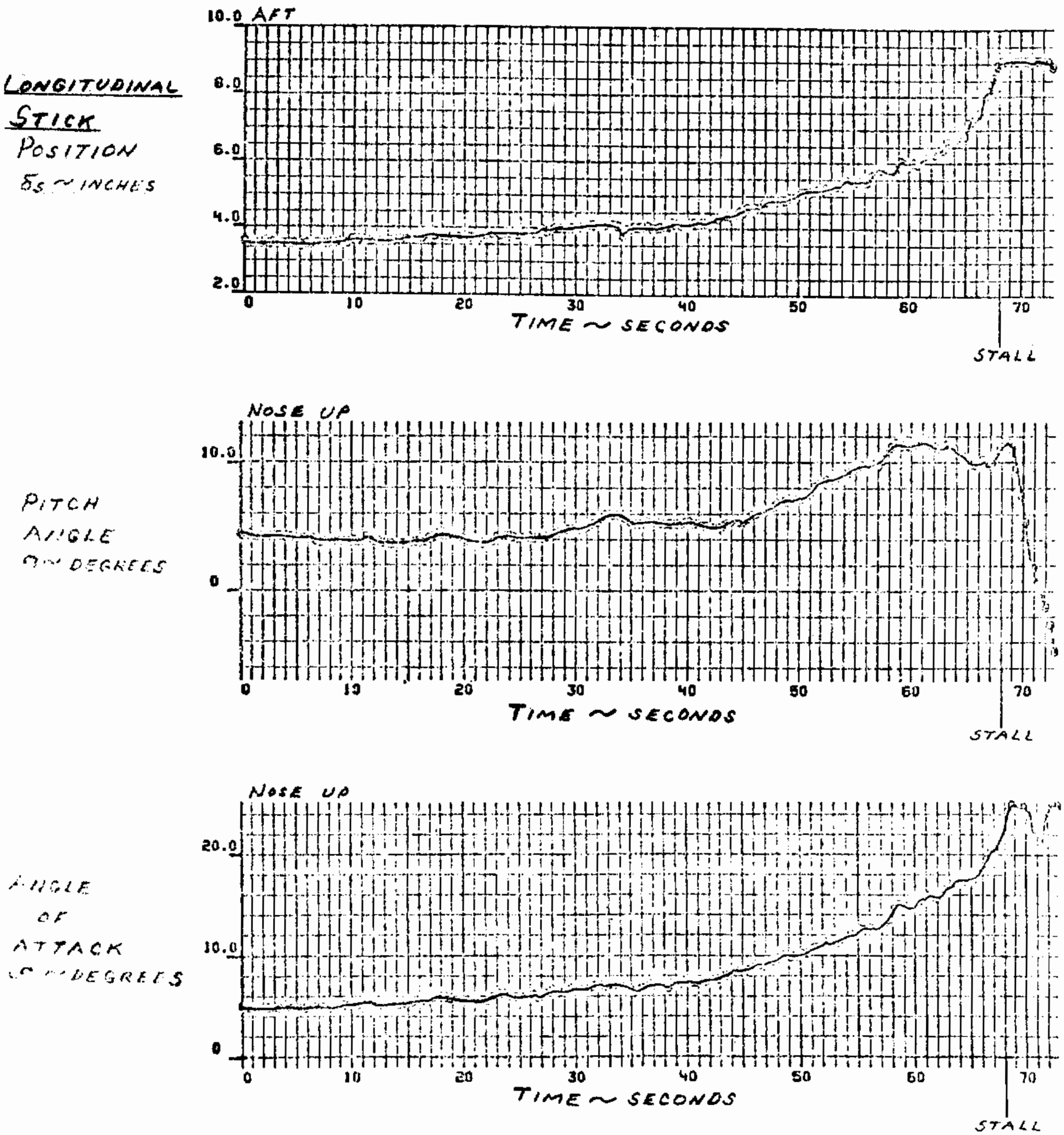
STALL CHARACTERISTICS

TABLE 1 (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL

CONFIGURATION	ALTITUDE	N_Z C.G.	WEIGHT	C.G. POS.
O———O	10,000 FT.	1.0 g.	14,170 LBS.	15.65 % MAC

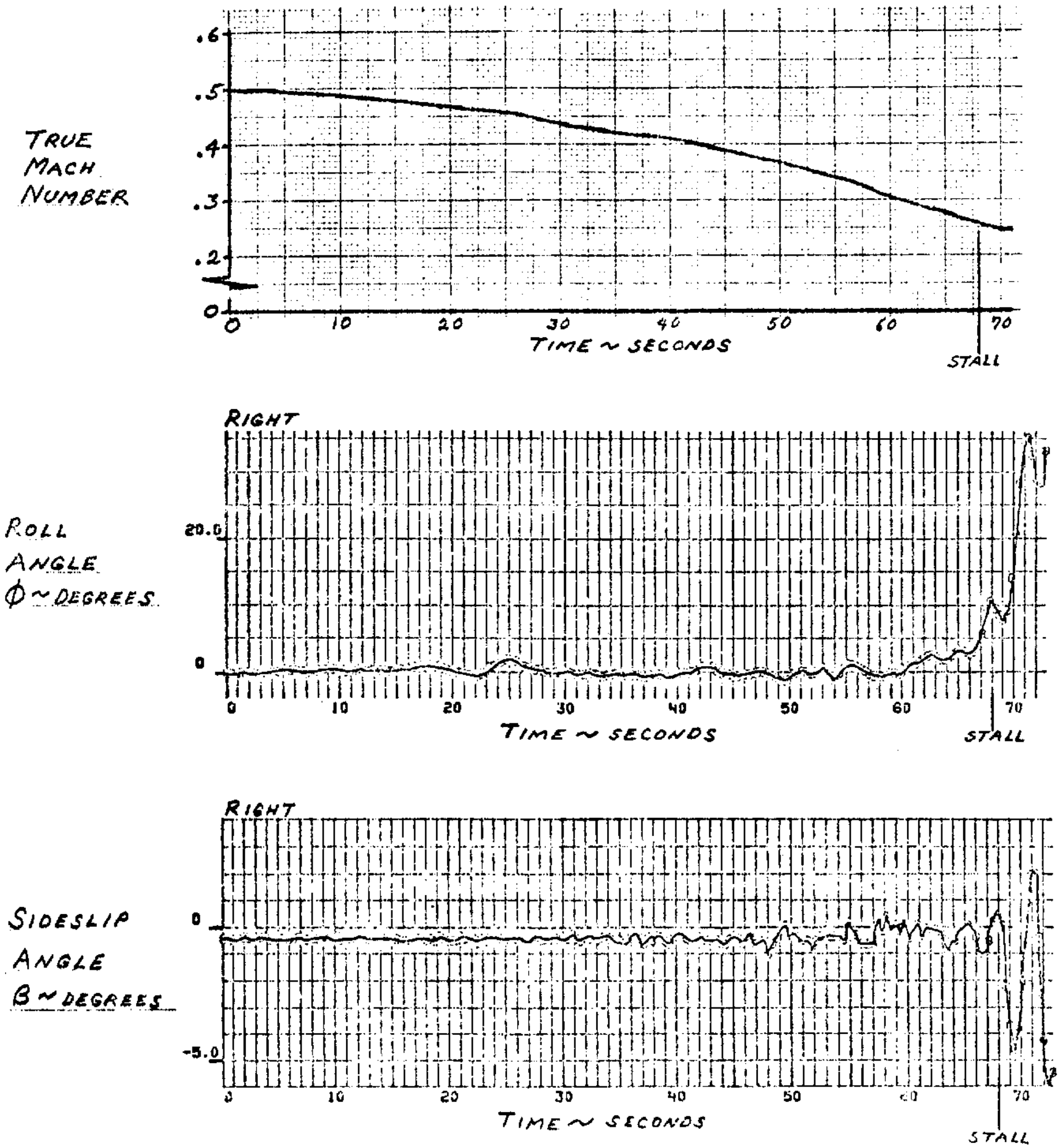
FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 1a (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL



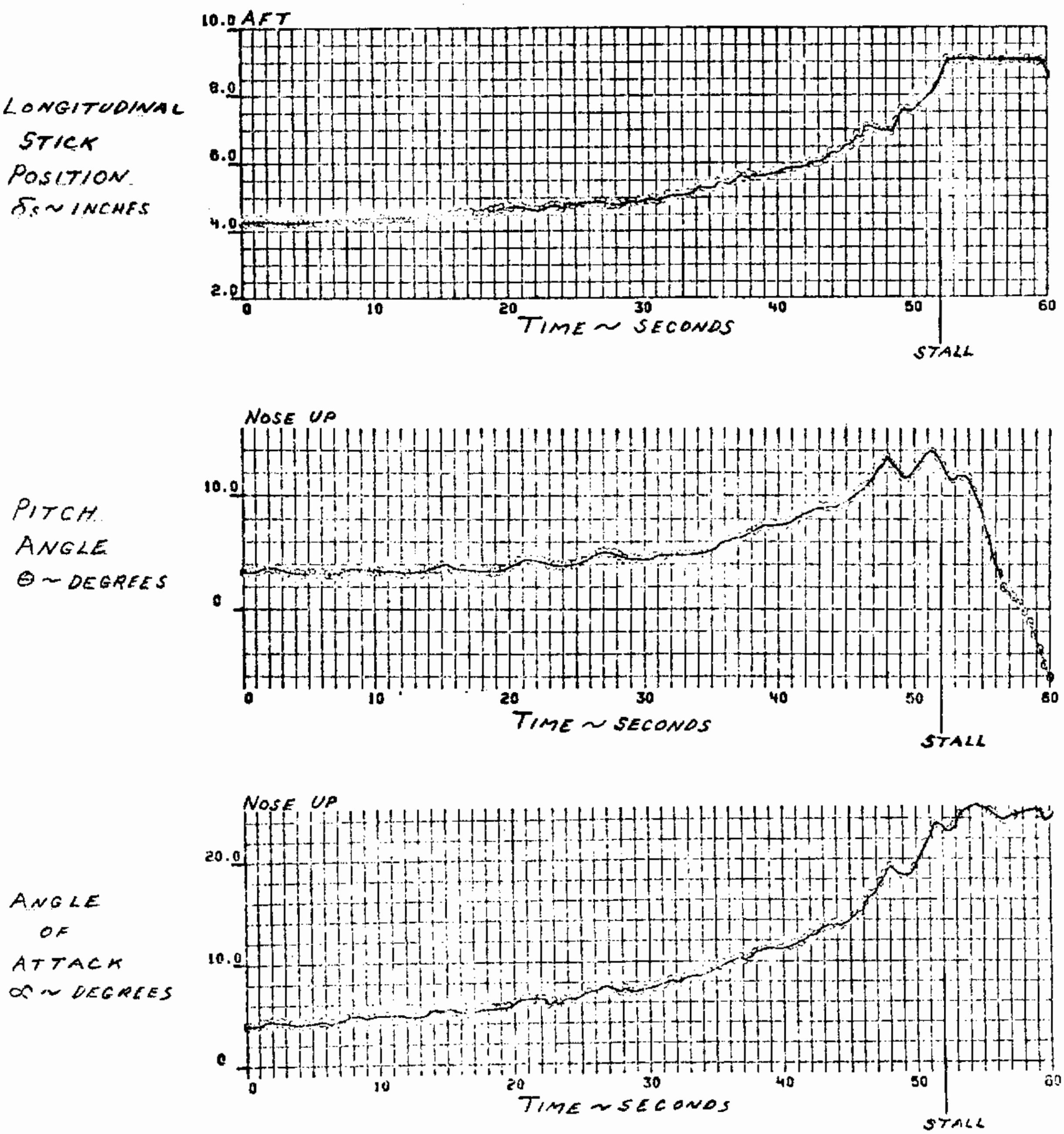
STALL CHARACTERISTICS

FIGURE 1l (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL

CONFIGURATION	ALTITUDE	N_Z C.G.	WEIGHT	C.G. POS.
O———O	10,000 FT.	1.0 g	13,920 LBS	15.52% MAC

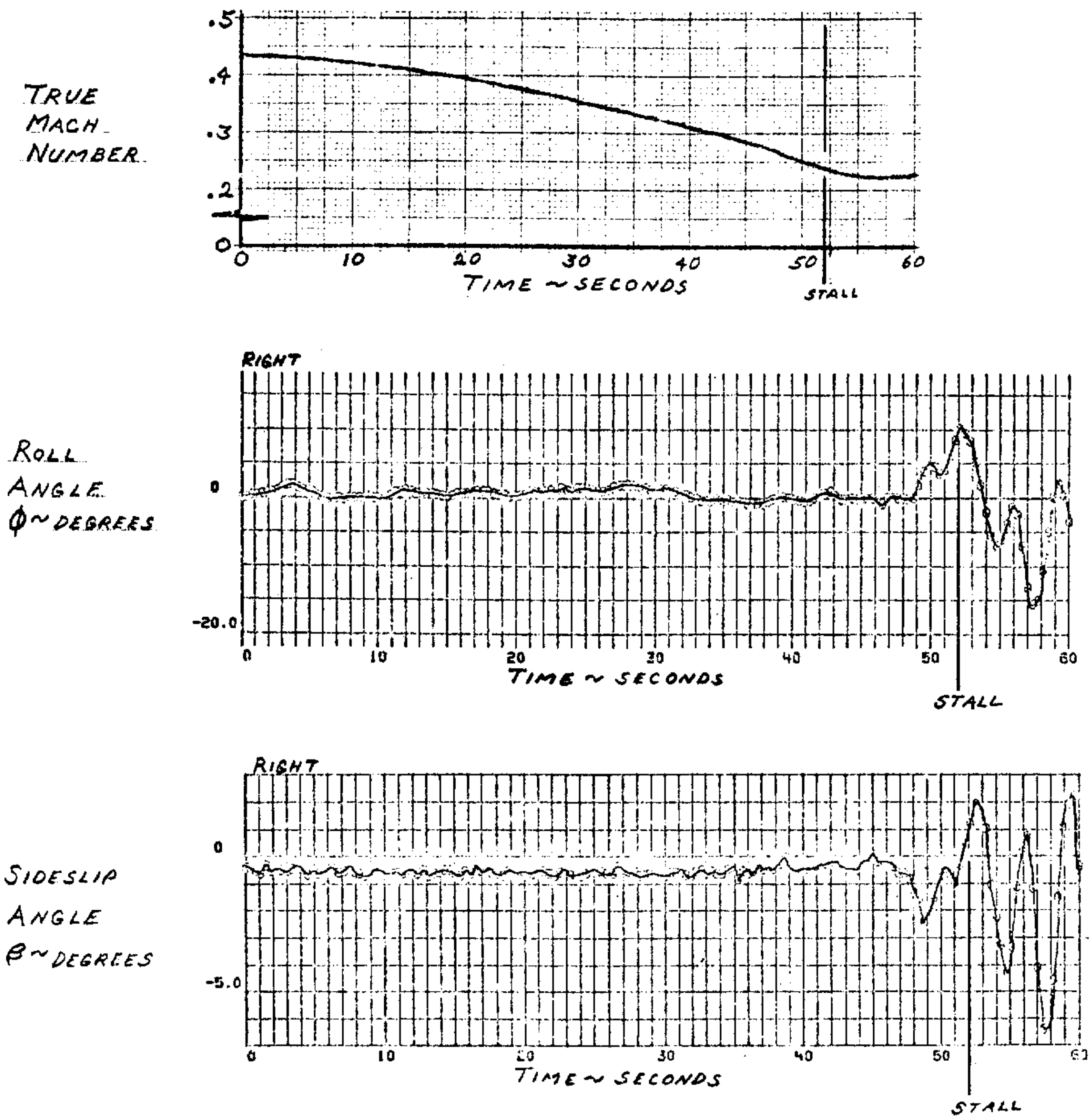
FLIGHT PHASE CATEGORY C
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 2a (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0 g STALL



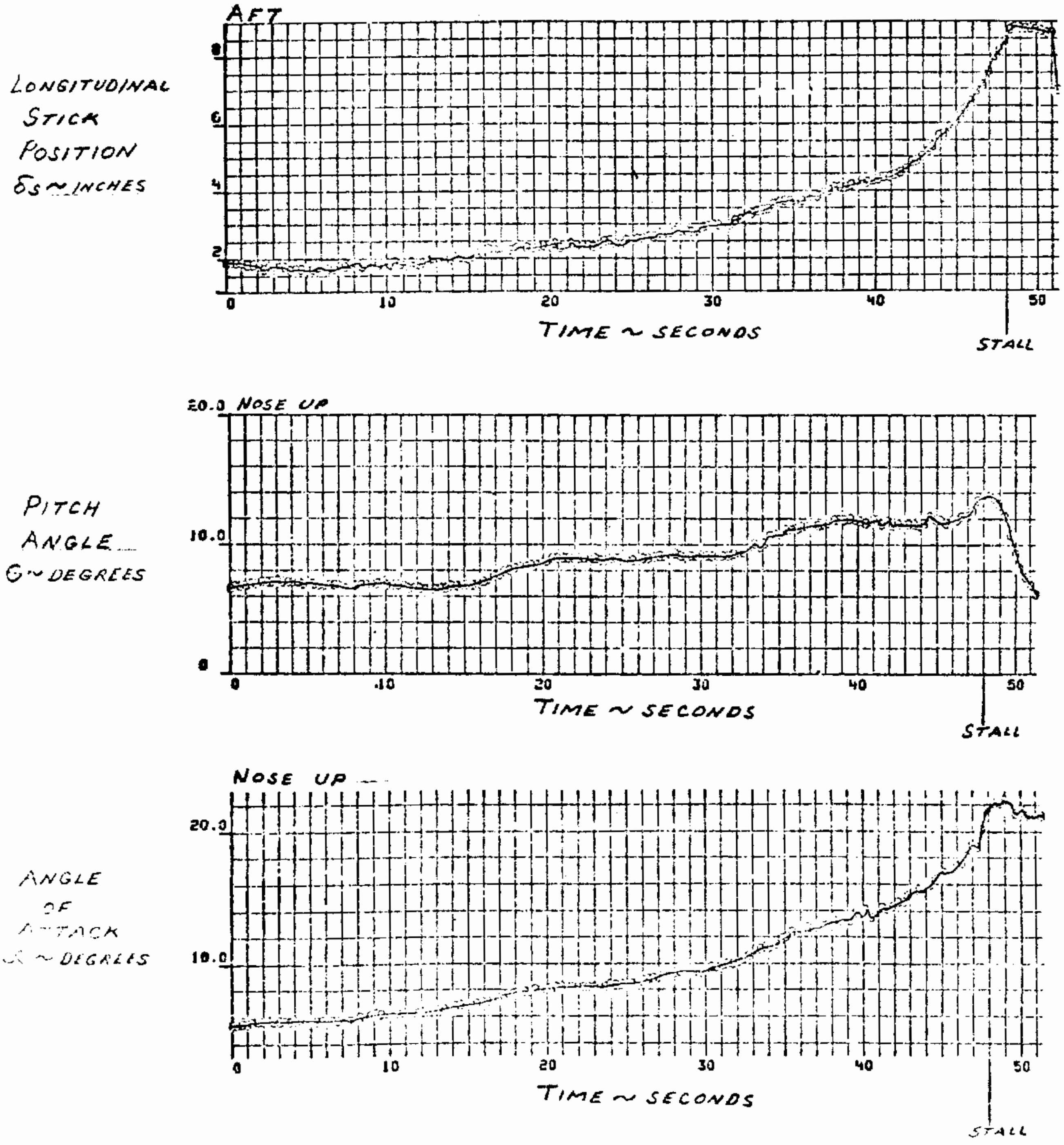
STALL CHARACTERISTICS

FIGURE 2l (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL

CONFIGURATION	ALTITUDE	N_Z C.G.	WEIGHT	C.G. POS
	10,000 FT.	1.0g	14,340 LBS	18.1% MAC

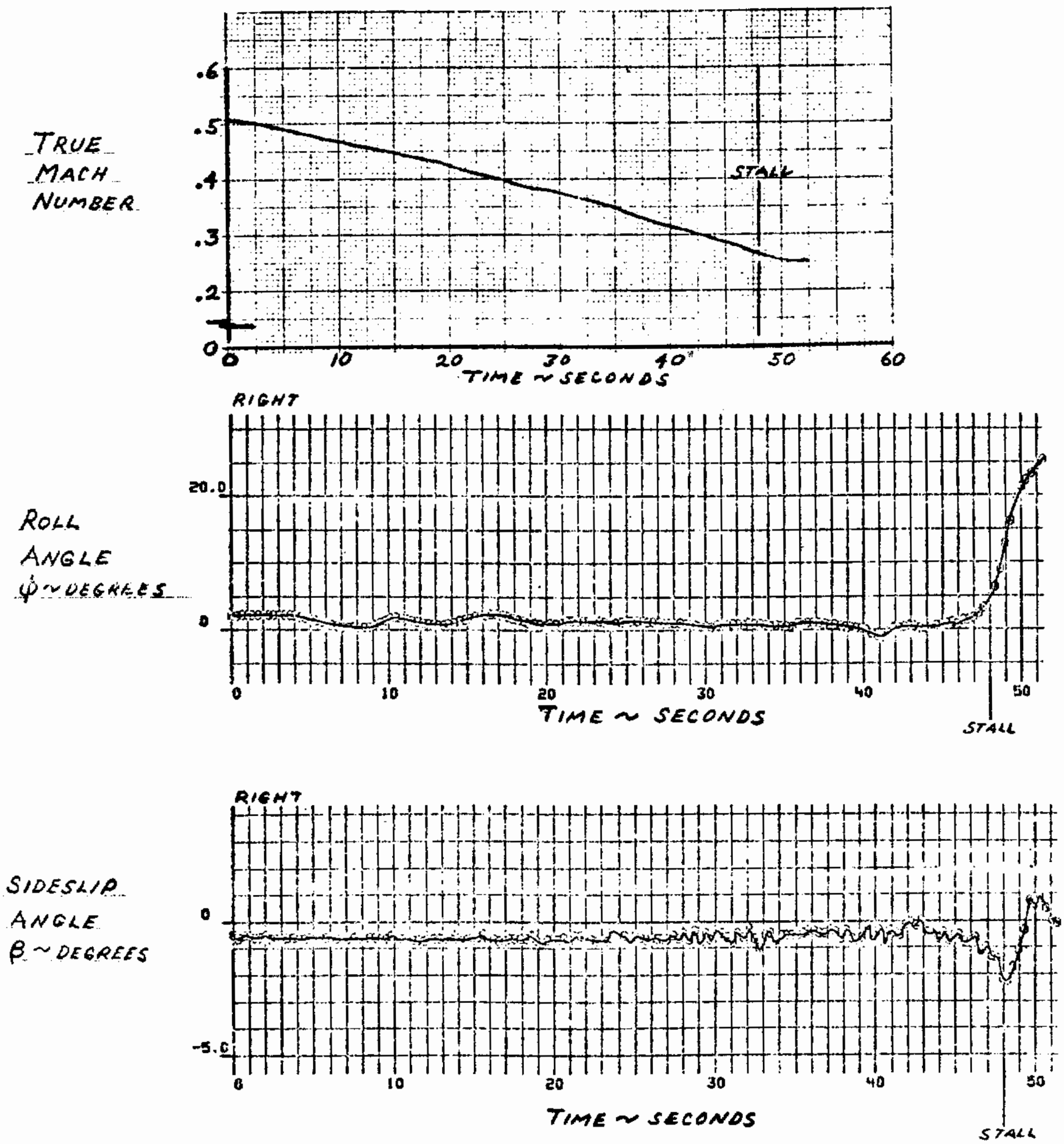
FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 3a (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0 g STALL



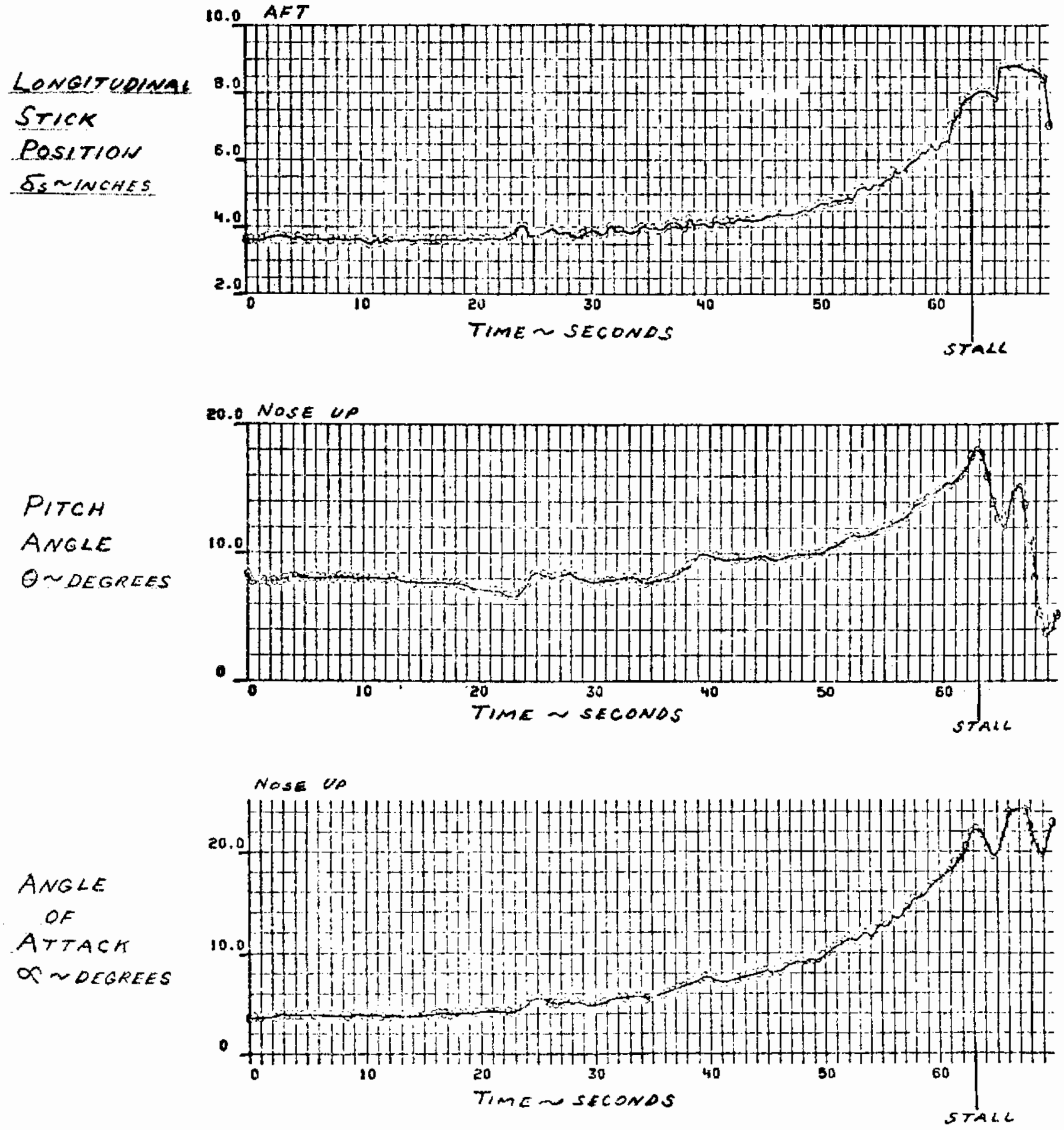
STALL CHARACTERISTICS

FIGURE 3l (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL

CONFIGURATION	ALTITUDE	N_Z c.g.	WEIGHT	C.G. POS.
	10,000 FT.	1.0g	12920 LBS.	14.91% MAC

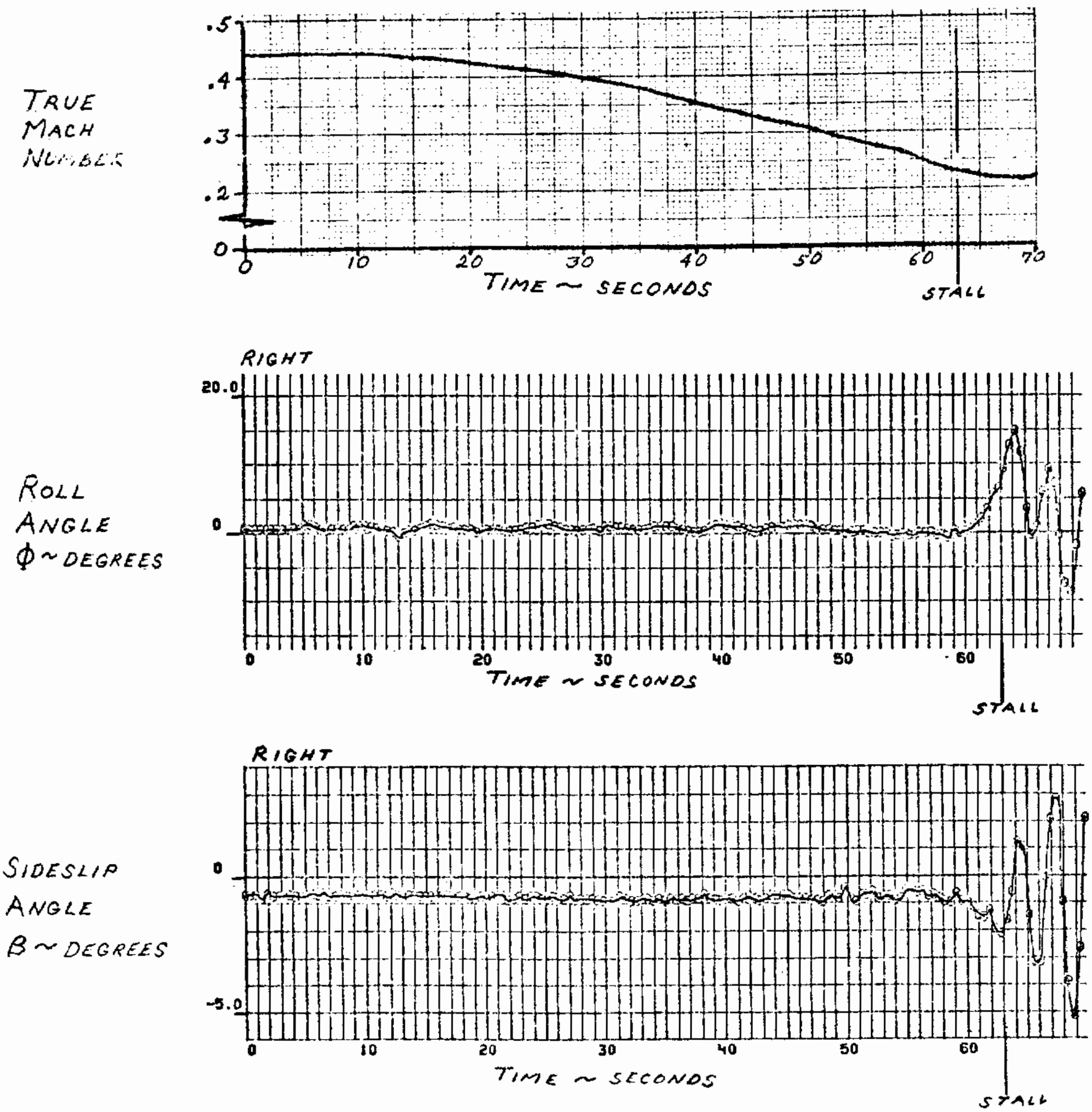
FLIGHT PHASE CATEGORY C
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 4a (3.4.2.3)

F-5 FLIGHT TEST DATA
TIME HISTORY OF A 1.0g STALL



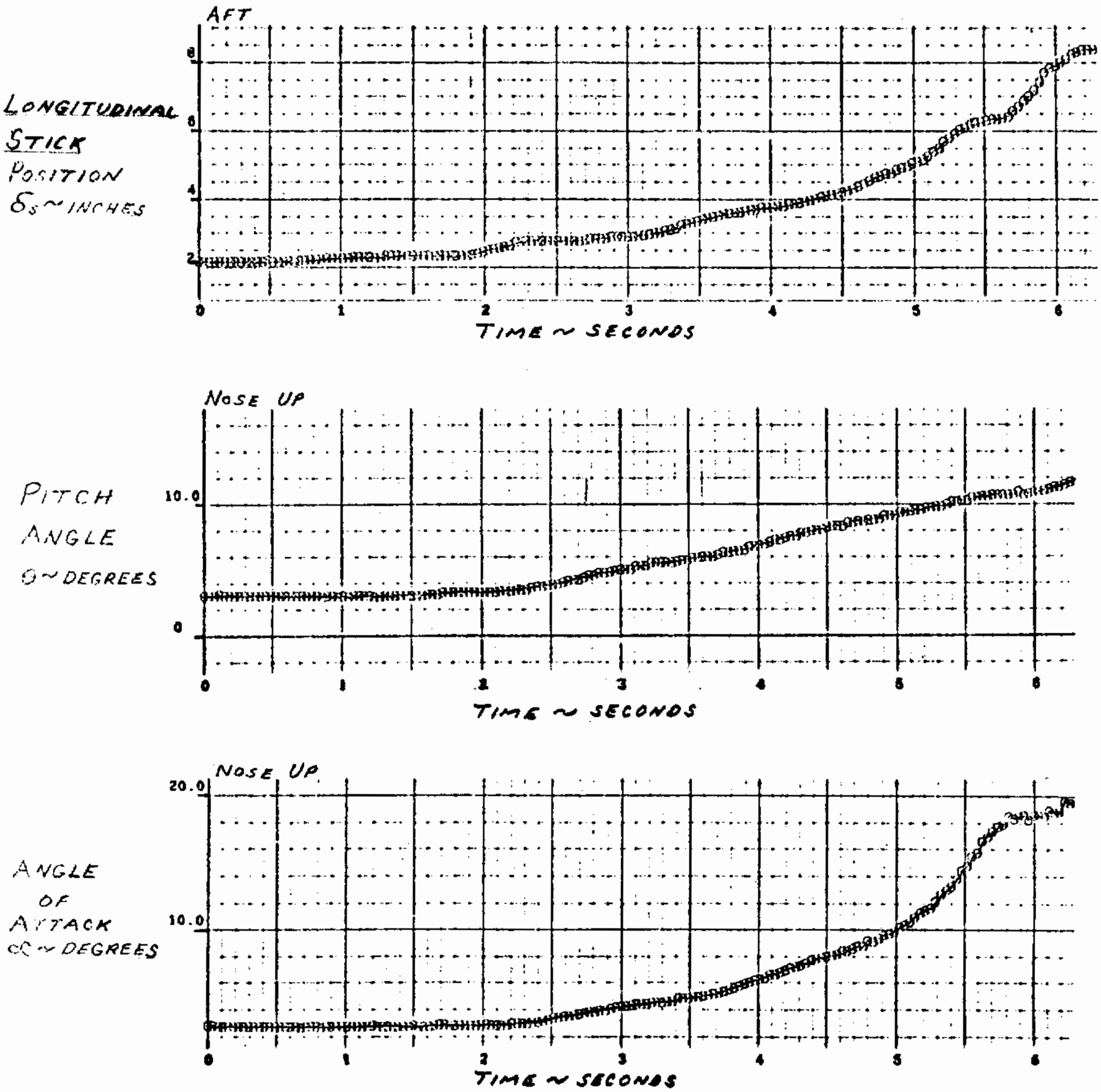
STALL CHARACTERISTICS

FIGURE 4l (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A WIND-UP TURN

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
o———o	10,000 FEET	0.6	12,790 LBS.	22.9 % MAC

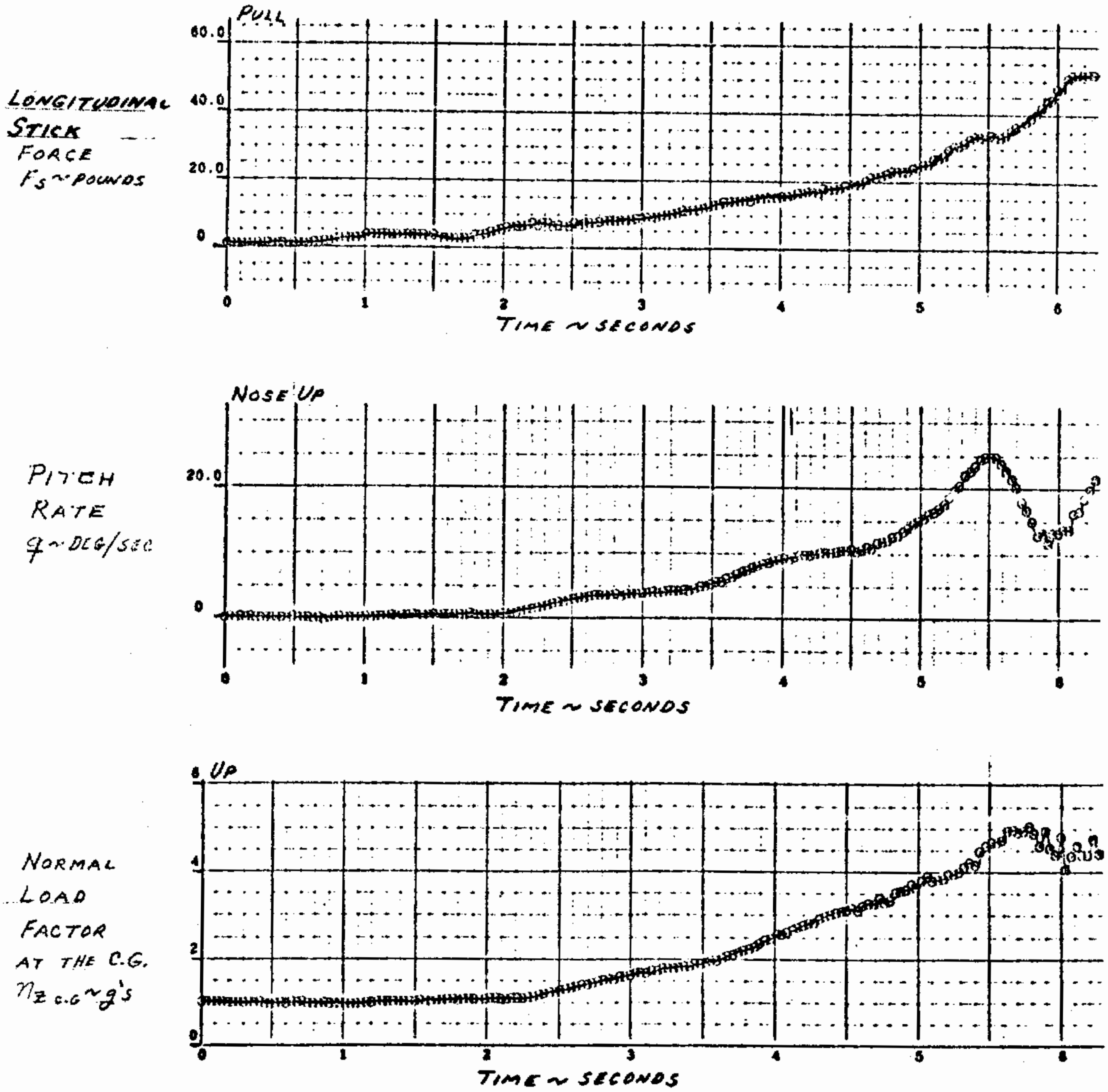
FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 5a (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A WIND-UP TURN



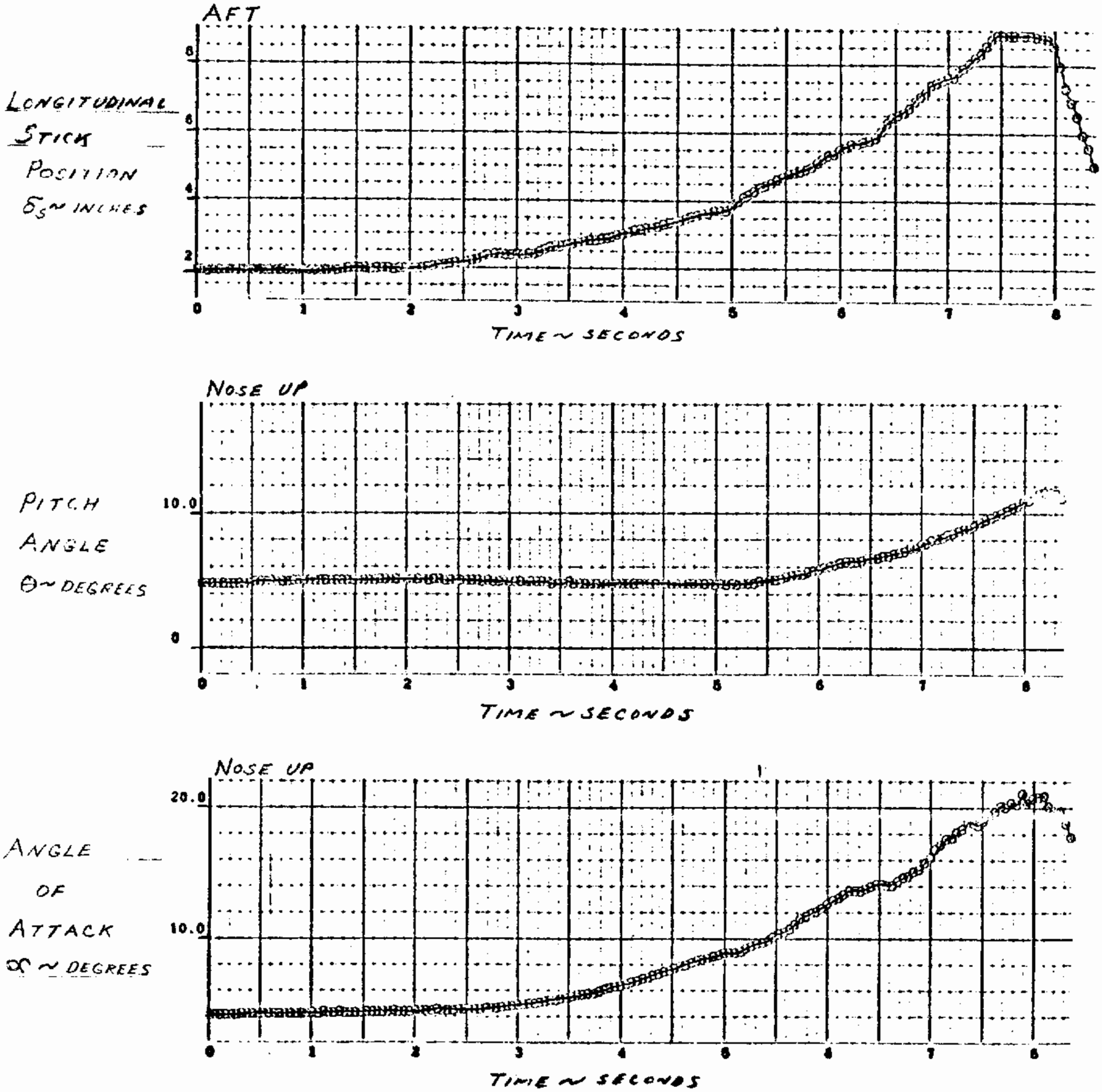
STALL CHARACTERISTICS

FIGURE 51 (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A WIND-UP TURN

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
	10,000 FEET	0.6	15,670 LBS.	16.32% MAC

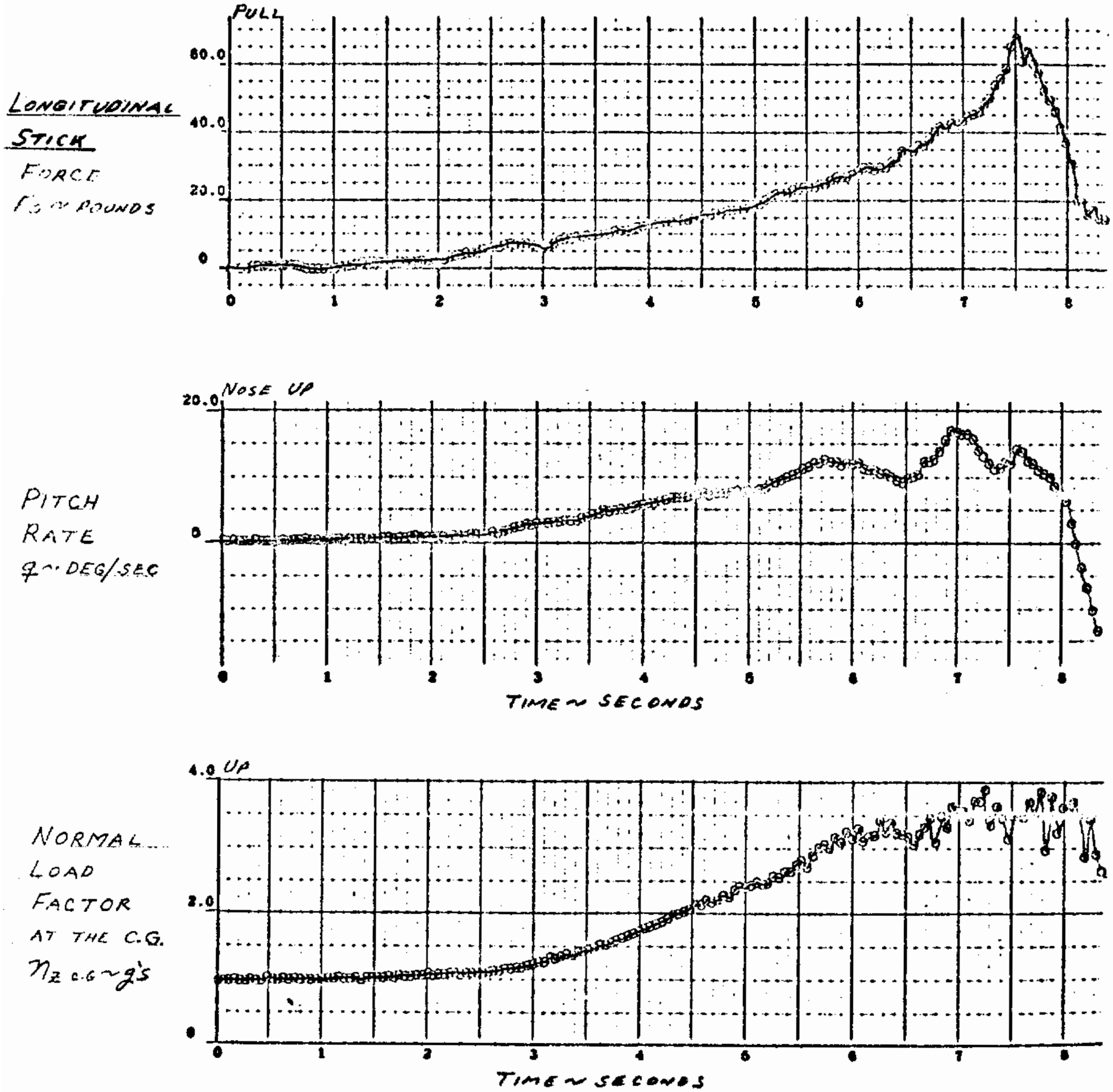
FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 6a (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A WIND-UP TURN



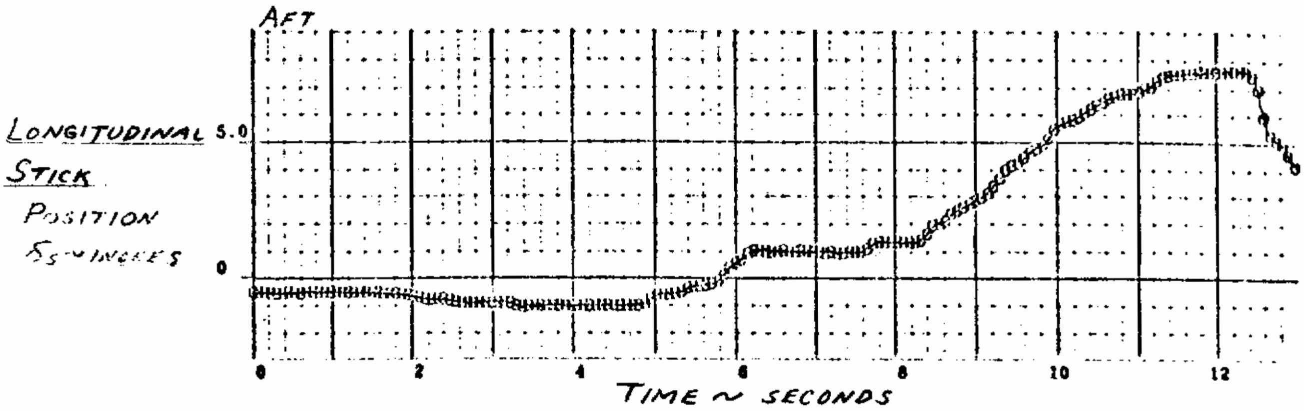
STALL CHARACTERISTICS

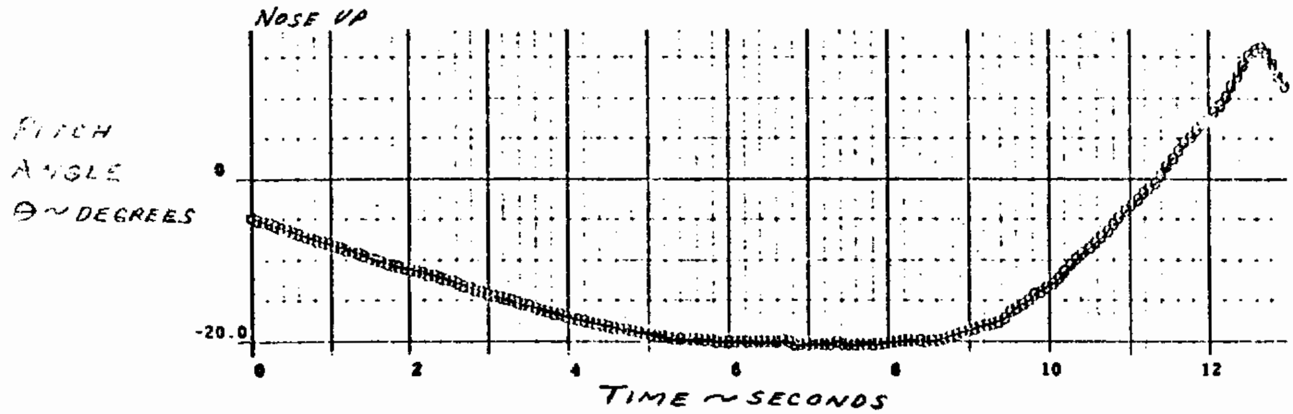
FIGURE 62 (3.4.2.3)

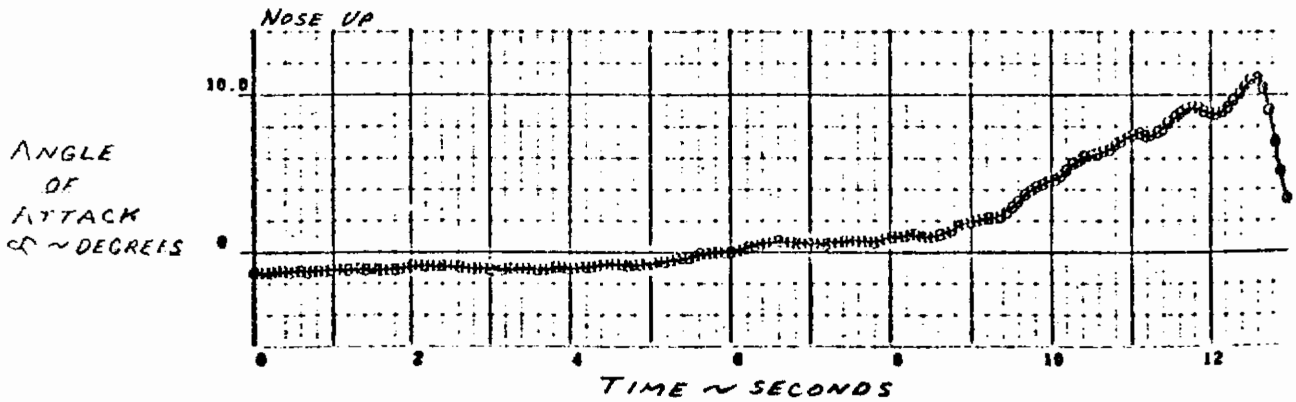
F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A SYMMETRICAL PULLUP

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
O-┬-┬-┬-┬-┬-┬-O	17,000 FEET	0.95	13,070 LBS.	13.81% MAC

FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



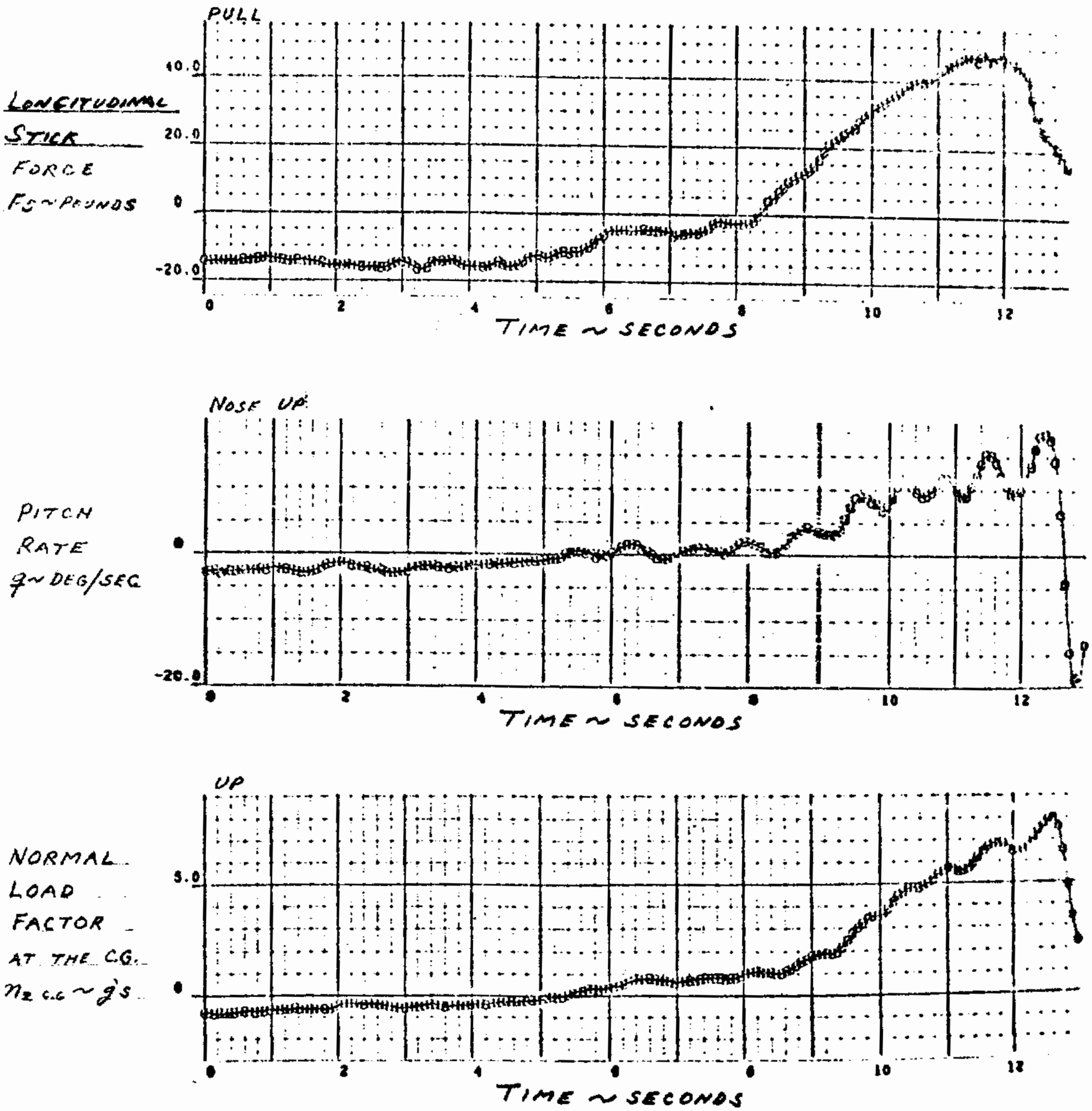




STALL CHARACTERISTICS

FIGURE 7a (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A SYMMETRICAL PULLUP



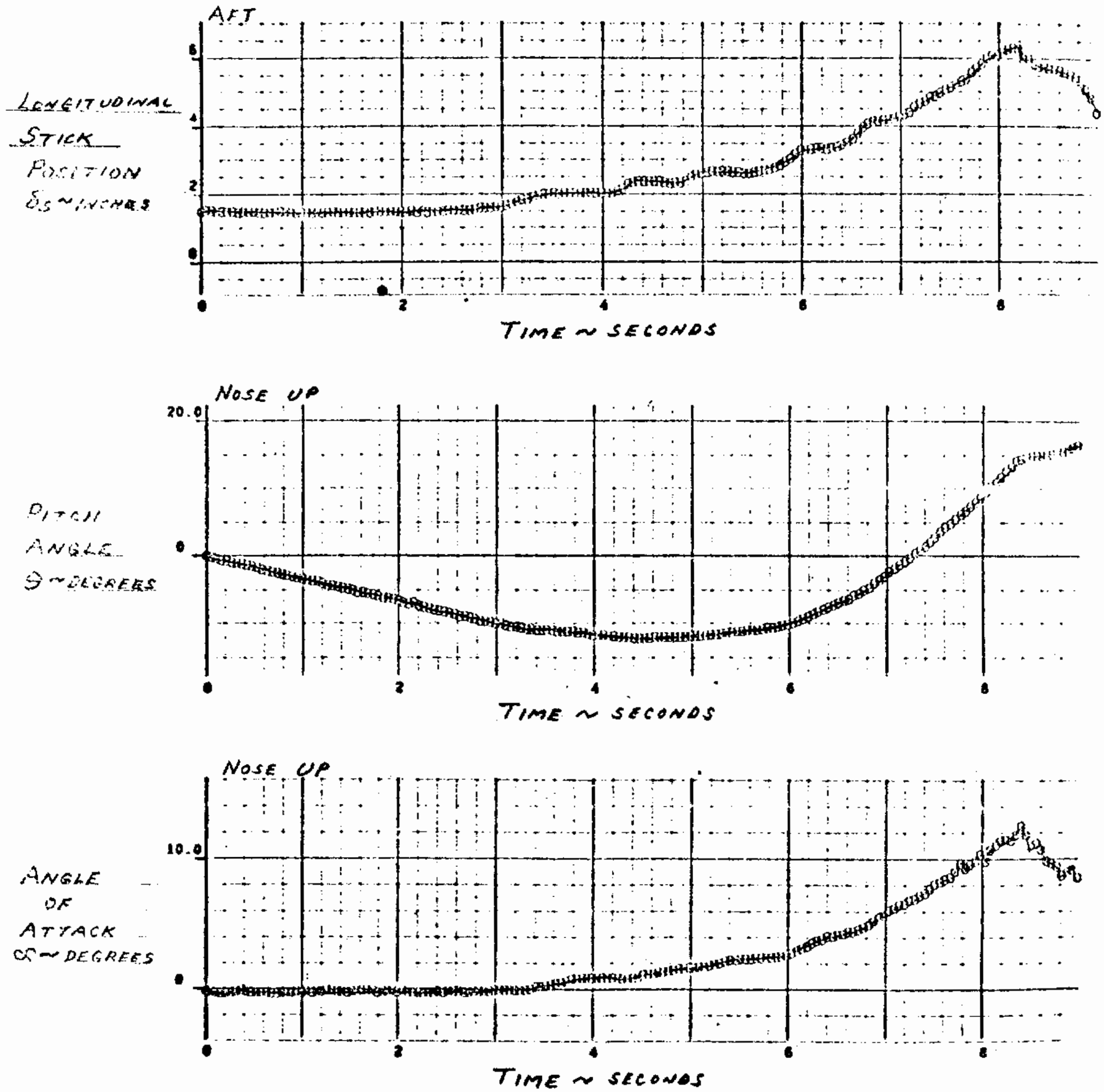
STALL CHARACTERISTICS

FIGURE 7b (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A SYMMETRICAL PULLUP

CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
	5,000 FEET	0.85	12,660 LBS.	19.85% MAC

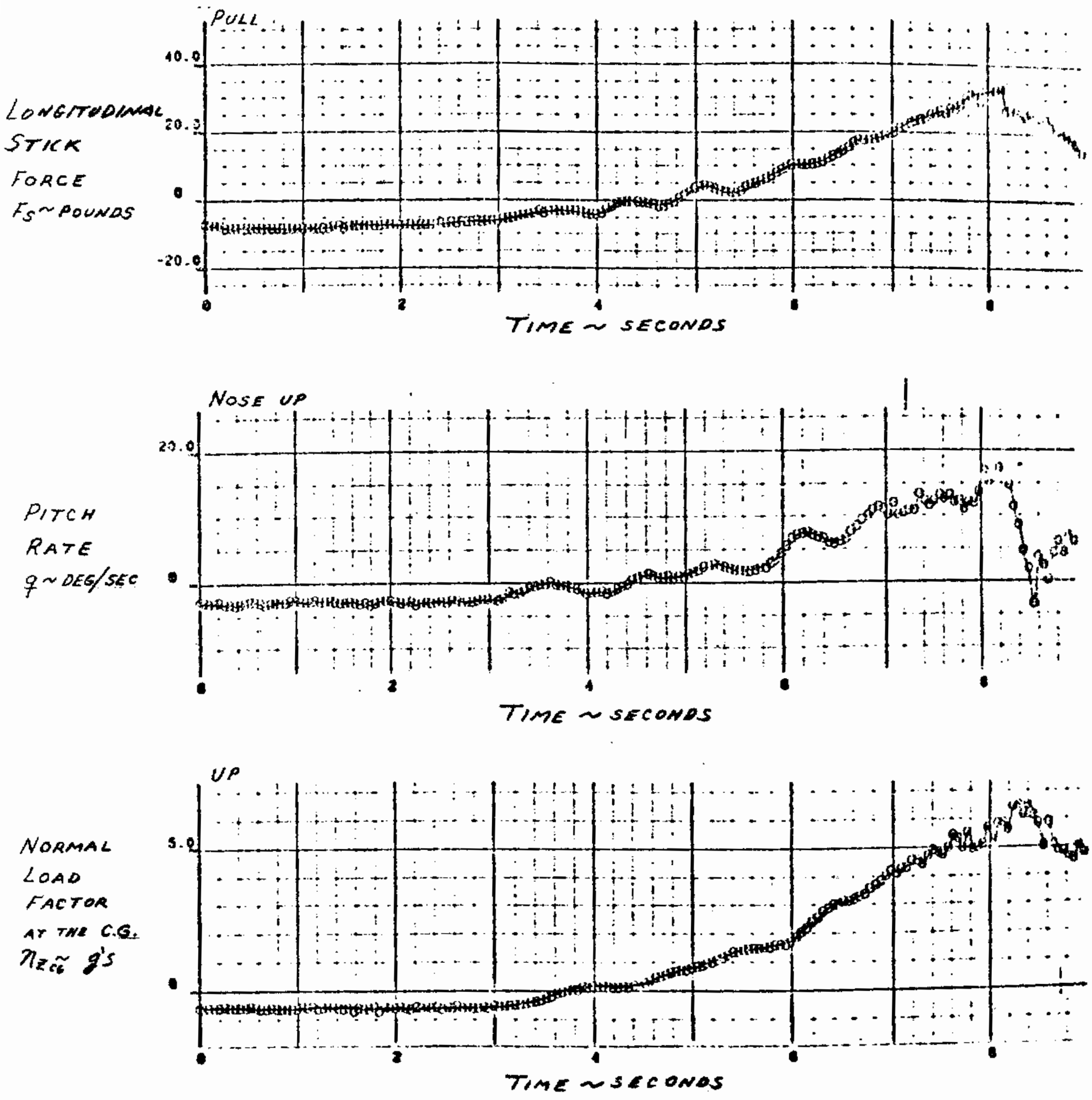
FLIGHT PHASE CATEGORY A
STABILITY AUGMENTERS OFF



STALL CHARACTERISTICS

FIGURE 8a (3.4.2.3)

F-5 ACCELERATED STALL FLIGHT TEST DATA
TIME HISTORY OF A SYMMETRICAL PULLUP



STALL CHARACTERISTICS

FIGURE 82 (3.4.2.3)

Requirement

Paragraph 3.4.2.4 Stall recovery and prevention. It shall be possible to prevent the complete stall by moderate use of the controls at the onset of the stall warning. It shall be possible to recover from a complete stall by use of the elevator, aileron, and rudder controls with reasonable forces, and to regain level flight without excessive loss of altitude or buildup of speed. Throttles shall remain fixed until speed has begun to increase when an angle of attack below the stall has been regained. In the straight-flight stalls of 3.4.2.1, with the airplane trimmed at a speed not greater than 1.4 V_S and with a speed reduction rate of at least 4.0 knots per second, elevator control power shall be sufficient to recover from any attainable angle of attack.

Comparison

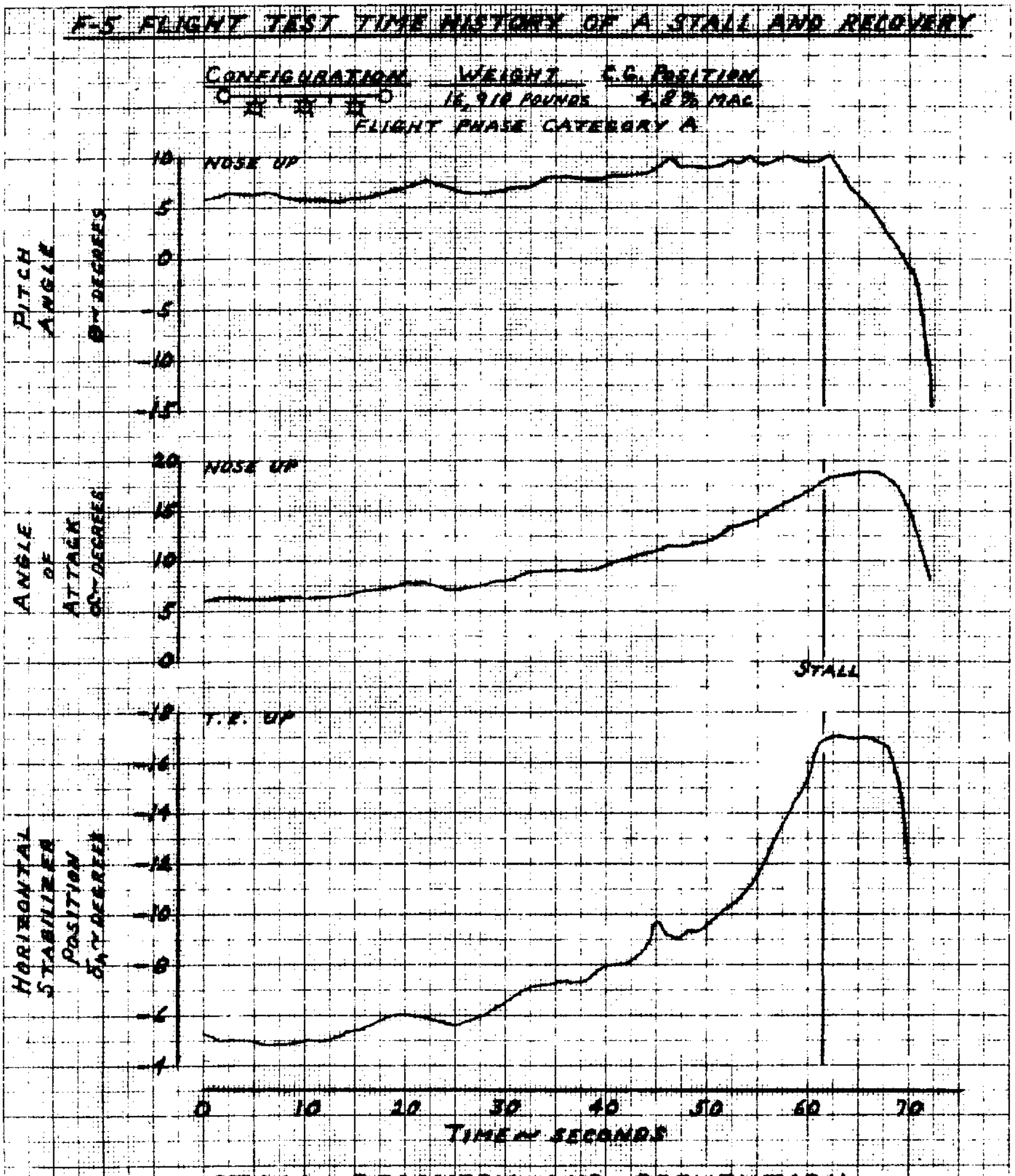
It is possible to prevent the complete stall of the F-5 airplane at the onset of stall warning by moderate use of the controls. A detailed discussion of F-5 stall warning characteristics is presented in paragraph 3.4.2.2. Figure 1 (3.4.2.4) presents a flight test time history of a 1.0 g stall and recovery. The data show the airplane to be out of the stalled attitude within 10 seconds after stalling. Minimum altitude loss and speed variation were realized during the recovery.

Resolution

In most aircraft, stall recovery is easier to accomplish at reduced or idle throttle settings. Since the three aerodynamic controls are allowed to be used in stall recovery, there is no apparent reason why throttle use is not allowed to aid recovery.

Recommendation

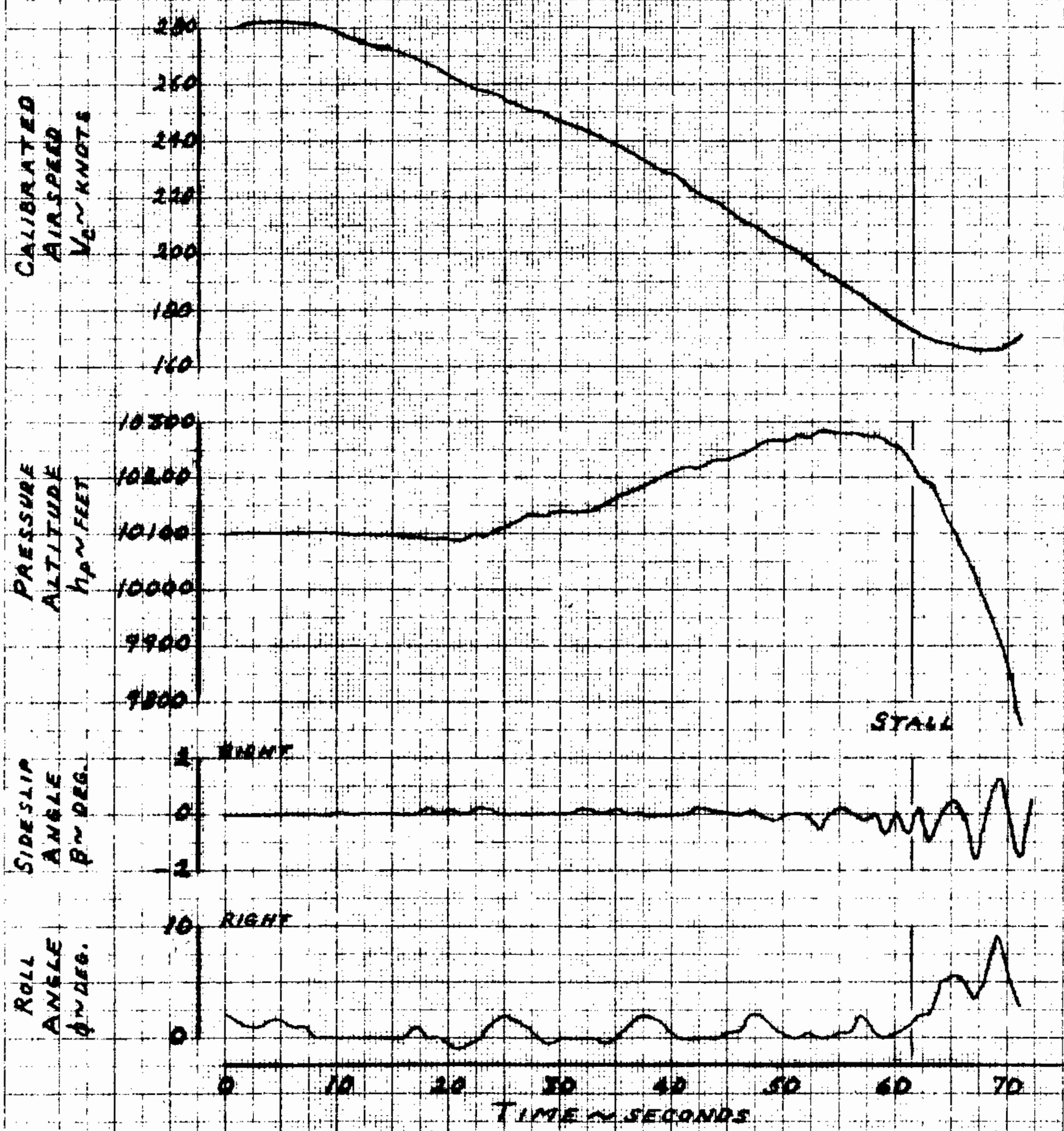
At the start of the third sentence, delete "Throttles shall remain fixed" and replace by "Throttles may be reduced."



STALL RECOVERY AND PREVENTION

FIGURE 1a (3.4.2.4)

F-5 FLIGHT TEST TIME HISTORY OF A STALL AND RECOVERY



STALL RECOVERY AND PREVENTION

FIGURE 14 (3.4.2.4)

Requirement

Paragraph 3.4.2.4.1 One-engine-out stalls. On multiengine airplanes, it shall be possible to recover safely from stalls with the critical engine inoperative. This requirement applies with the remaining engines at up to thrust for level flight at $1.4V_S$, but these engines may be throttled back during recovery.

Comparison

No flight test data are available for comparison of F-5 characteristics with the requirements of this paragraph. However, due to the close proximity of the engines to the plane of symmetry, the yawing moments induced by a one-engine-out condition is quite small and will not produce unacceptable yawing or rolling tendencies. The safe recovery from stalls will not be hampered by this condition. This requirement is acceptable.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.3 Spin recovery. If spin demonstration is required by MIL-S-25015 or MIL-D-8708, consistent prompt recoveries shall be possible from all modes of incipient and fully developed erect and inverted spins, using controls as required by the referenced specifications. If such controls include a special spin recovery device, that device shall satisfy the following addtional requirements: required pilot action shall be easy, consistent, and simple; the device shall be immediatley reusable for several spins on the same flight. Recovery control forces shall not exceed 250 pounds rudder, 75 pounds elevator, or 35 pounds aileron.

Comparison

The following is a synopsis of the historical background of both analytical and flight test programs performed on T-38 and F-5 aircraft.

T-38 AIRCRAFT

1. Preflight Test/Analyses

   a. Tests:

      (1) Vertical Wind Tunnel (Spin Model)

      (2) Free Flight Model (Catapult)

      (3) Hi-Attitude Static and Fuselage Nose Balance (W.T.)

      (4) Reynolds No. Effects and Effect of Rotary Motion (W.T.)

   b. Conclusions

      (1) Rather Flat Spin Mode: $\alpha = 70^{\circ} - 85^{\circ}$

      (2) Spin Rate: 170°/second

      (3) Spin Axis: c.g.

      (4) Entry: Above stall speed - abrupt up tail

      (5) Recovery: 1-1/2 to 3-1/2 turns

(6) Recovery Controls: Aileron with, up tail, rudder against

(7) Spin Recovery Chute of 241.2 ft^2 fist type was most effective.

(8) At high angle of attack the yawing moment is primarily from the forward fuselage.

(9) Reynolds number has a primary effect on yawing moment and side force, much less effect on all other derivatives.

(10) Critical Reynolds number is not affected by rotary motion

c. Test Analysis Report:

Bernard, A. V., "A Qualitative Analysis of the Spin Characteristics of the Northrop T-38 Trainer Airplane", Northrop Corporation, Aircraft Division, NOR-59-429, July 1959.

2. <u>Flight Test Phase</u>

a. Initial Demonstration of Recovery:

Very hard to develop a spin - had to use spin recovery chute

b. Analytic Predictions:

Six-Degree-of-Freedom IBM Program obtained from NASA and with NASA help established that -

(1) Control application required to promote or recover spin. Recovery controls were same as (1. -b. -6) from preflight analysis.

(2) Steady-state spin and recovery characteristics -

Oscillatory mode easily recoverable.

Flat mode required excessive number of turns to recover. This was not recognized in preflight tests which showed only 1-1/2 to 3-1/2 turns to recover.

(3) Grantham, William D., "Analytical Investigation of the Spin Characteristics of a Supersonic Trainer Airplane Having a 24° Swept Wing", Langley Research Center, Langley Station, Hampton, Va., NASA TM X-606, April 1962.

c. Follow-on Flight Demonstration:

Directed primarily toward demonstrating spin resistance. General entry found to be from high pitch rate maneuver - Pitch rate couples with roll rate to give yaw acceleration. Results indicated extreme difficulty to enter spin from aborted normal flight maneuvers. Also substantiated analytic predictions.

d. Final Report Documentation:

Hirsch, D. L., "Spin Characteristics of the T-38 Airplane YJ85-GE-5 Engines Installed", Northrop Corporation, Aircraft Division, NOR-61-151, November 1961.

e. Correlation of Results

SPIN TUNNEL	ANALYTIC 6DOF COMPUTER	FLIGHT TEST
$\alpha s/s$ 75^o-85^o	$\alpha s/s$ 75^o	$\alpha s/s$ 84^o
Wing Tilt $\pm 7^o$ (smooth)	$\pm 3^o \phi$ (smooth)	Oscillatory ϕ $\pm 10^o$ ϕ smooth
Spin Rate-160^o/sec	175^o/sec	160^o/sec
Spin Axis - c.g.	c.g.	c.g.
Sink Rate 236 Ft/Sec	260 Ft/Sec	280 Ft/Sec
Recovery-1-1/2 to 3-1/2 turns	Recovery is function of yaw rate	Rapid Recovery From Oscillatory Mode - Excessive number of turns (smooth) for recovery from flat mode
Recovery Control δ_r against δ_a with δ_h aft	As predicted δ_r against δ_a with δ_h aft	As predicted δ_r against δ_a with δ_h aft
No inverted mode	As predicted no inverted mode	As predicted no inverted mode
Entry - Full Up Tail at Max Rate Near Stall Velocity	No Entries Using Control Inputs	Difficult to enter - Entry controls as predicted from Catapult Tests: full-stick-deflection push-pull and hold

F-5 AIRCRAFT

1. Preflight Tests and Analyses

a. Tests:

High attitude wind tunnel test. Complete model buildup to yield data for rotary derivatives calculated based on conventional techniques established.

b. Analyses:

(1) Rotary derivatives calculated by conventional techniques established by NAA. Wykes, J.H., Casteel, G.R., Collins, R.A., "An Analytical Study of the Dynamics of Spinning Aircraft", WADC, TR-58-381, Parts I and II December 1958, Part III February 1960.

(2) Complete six-degree-of-freedom (6DOF) equations mechanized for IBM and Analog. Certain linearization had to be accomplished in terms of cross coupling aerodynamics because of computer limitation.

(3) A unique simple approach was employed in establishing possible spin entry boundaries. Angle of attack versus yaw rate boundaries were determined which, if passed through, could lead to a reported spin. Initial values of yaw rate, and angle of attack were set and the resulting motions of the aircraft were studied to determine the spin characteristics exhibited. See Figure 1 (3.4.3). The method is analogous to a spin tunnel program; however, many more combinations of yaw rate and angle of attack can be obtained in a shorter time span and much more precise control of the initial conditions can be maintained.

Analyses were conducted to establish the effect of F-5 external store configurations on the spin entry, steady state spin and controls for spin recovery. The recovery spin chute sizing was also accomplished. Results indicated the no-external-store F-5 and T-38 were similar. The analyses were completed prior to flight test and used to choose critical configurations and flight conditions to be demonstrated.

2. Flight Test

Based on analyses, the clean F-5 and the most directionally unstable empty centerline tank configurations with the lowest I_x and I_y (roll and yaw moments of inertia) were flight demonstrated. Flight testing was primarily directed toward demonstrating spin resistance rather than recovery. Results indicated that spin resistance of the F-5 was the same as that of the T-38. The addition of a centerline tank did not change the resistance to spin but did accentuate the poststall gyrations. One spin was developed, which indicated good correlation with the predicted entry envelope.

3. Final Report Documentation

Titiriga, A., et al, "F-5 Spin Susceptibility Investigation", Northrop Corporation, Aircraft Division, NOR-65-33, December 1964.

CF-5 (CANADIAN) AIRCRAFT

1. Analysis

High-attitude wind tunnel tests were conducted to determine the aerodynamic effects of a reconnaissance nose. Six-degree-of-freedom analyses were then conducted to determine the effects of aerodynamic and inertia changes between the CF-5 and F-5 aircraft. Based upon the results of this study, it was recommended, and accepted by Canada, that the changes were either negligible or in a direction to be less critical and that no flight testing was required.

2. Final Report Documentation

Titiriga, A., "The Analytical Investigation of the Spin Characteristics of the CF-5 Aircraft", Northrop Corporation, Aircraft Division, NOR-67-43, March 1967.

NF-5 (NETHERLANDS) AIRCRAFT

1. Analysis

High-attitude wind tunnel tests were conducted to determine the effect of maneuver flap deflection and larger wing pylon fuel tanks. Six-degree-of-freedom analysis showed that although some frequency changes occurred during the spin mode, the resulting motion was very similar to that of the F-5. It was recommended that no spin flight tests be conducted and it was so accepted by the Netherlands Air Force.

2. Final Report Documentation

Kandalaft, R. N., "Analytical Investigation of the Spin Characteristics of the NF-5 Aircraft", Northrop Corporation, Aircraft Division, NOR-67-153, September 1968.

F-5 ANALYTICAL SPIN SENSITIVITY STUDY

1. Analysis

Analog computer 6 DOF runs were conducted to isolate the effects of aerodynamic and inertia changes on the F-5 spin characteristics. The results indicated the following characteristics:

2. Spin Boundary Effects

a. Cm_{α}, $C_{n_{\beta}}$, Cl_{β}, and C_{n_r} are most significant in determining spin boundary characteristics.

b. Increasing inertia along the wing tends to make the entry boundary more remote.

c. Changes in pitch inertia isolated further the steady-state spin regions.

Trends established for the above are shown in Figure 2 (3.4.3).

3. Spin Entry

On the computer, using F-5 data, spin entry with controls cannot be achieved. Initial values of yaw rate and angle of attack are utilized to map the spin region. The basic F-5 data were altered to investigate the effects on spin entry, with the following results:

a. If the margin between full-up tail trim angle of attack and the angle of attack for spin entry was reduced to less than 10 degrees, then spins could be entered with normal control inputs.

b. If adverse yaw due to aileron was introduced at α's greater than 20 degrees, then spins could be obtained with a larger margin between α_{trim} and α for spin entry.

c. Losses in directional stability in conjunction with decreases in pitching moments resulted in nonrecoverable flat spins.

4. Final Report Documentation

Titiriga, A., et al, "F-5 Spin Sensitivity Study (Criteria Establishment and Comments Relative to Military Spin Specification)", Northrop Corporation, Aircraft Division, FMR-69-7, May 1969.

Resolution

The F-5 and T-38 airplanes are highly resistant to spin entry. No spins were reported during the service lives of these two airplanes which so far have spanned over 3 million hours. However, in a fully developed spin, these airplanes will be difficult to recover. Hence, disagreement between the F-5 characteristics and the requirements of this paragraph does exist.

A resolution of this disagreement is presented in the form of comments on MIL-S-25015 and appears as a recommendation to this paragraph.

Recommendation

The following are comments on revision to MIL-S-25015 and are directed solely to Class IV airplanes. The experience gained from the T-38 and F-5 spin programs is the basis for these comments.

1. General Requirements

The specification directs attention to the recovery from spins. It is suggested that the requirements be separated into the following:

Phase I	Spin Resistance
	a. analytic
	b. flight demonstration
Phase II	Spin Recovery (oscillatory)
	a. analytic
	b. flight demonstration
Phase III	Spin Recovery (fully developed)
	a. analytic
	b. flight demonstration

Analyses shall include, but not be limited to:

1. Definition of critical flight conditions for spin entry
2. Definition of critical configurations for spin entry
3. Definition of recovery controls as a function of configuration and spin mode.

2. Configurations

The effects of configuration differences in terms of aerodynamic and inertial characteristics shall be determined analytically prior to flight demonstration. Analytically is defined as but not limited to:

a. 6-degree-of-freedom motion analysis
b. Wind tunnel tests
c. Free flight model tests

It is suggested that correlation of at least two of the above be accomplished.

Analyses shall include investigation of the following items. General trends are presented for use as a guide.

a. C.G. position - In general, aft limit c.g. positions result in the highest trim angle of attack and are considered the most critical.

b. Longitudinal Stability - Although, in general, this is covered under Item a, the lowest static margin condition at stall angles of attack is considered the most critical.

c. Lateral Stability - Large negative values of C_{ℓ_β} in the body axis affect the steady-state spin and spin recoveries. These configurations should be investigated.

d. Directional Stability - Configurations exhibiting the least directional stability at the highest trim angles of attack show least resistance to spin entry and should be investigated.

e. Adverse Yaw - Yaw due to roll control can affect the entry, incipient, steady-state spin and recovery characteristics. Partial as well as full roll authority inputs should be investigated. Extreme adverse yaw at angles of attack near stall is considered the most critical.

f. Rotary Derivatives - In general, the least damping or most pro-spin rotary derivatives are considered the most critical.

g. Moments of Inertia - It is considered that the yaw coupling moments of inertia are the most significant. Configurations which yield the highest value of $\left(\frac{I_x - I_y}{I_z}\right)pq$ are considered the most critical.

h. Stability Augmentation - Some common motion feedbacks can actually trigger loss of control. For example, depending upon the configuration, roll rate feedback to ailerons or spoilers can actually reverse roll damping. (The F-5 airplane has no stability augmentation in the axis).

3. Analyses

Analyses shall be conducted leading to definition of the flight test demonstration program. These analyses shall be documented and approved by the procuring activity prior to initiation of flight test.

4. Flight Test

The flight test demonstration program shall be conducted in two phases:

Phase I	Spin resistance and recovery from poststall gyration
Phase II	Spin recovery from fully developed spins

If no fully developed spins are encountered in the spin resistance flight program, and if all poststall gyrations and/or incipient oscillatory spins were or could be shown to be recoverable, there is no requirement to demonstrate recovery from fully developed spins (PhaseII).

5. Phase I - Flight Demonstration

Spin resistance flight tests shall be approved by the procuring activity prior to initiation of testing and shall include, but not be limited to, the conduction of the maneuvers given in Table 1 (3.4.3). These maneuvers are also applicable to the analysis section requirements.

Maneuvers	Configurations	No. of Entries	
Zooms	Critical	1	30^{o} climb to stall, full back stick, (power as required)
		1	60^{o} climb to zero airspeed controls trimmed (power as required)
		1	60^{o} climb to 150K, full fwd. stick* applied at 150K (idle power)
		1	90^{o} climb to zero airspeed controls trimmed (power as required)
		1	90^{o} climb to 150K, full aft stick* applied at 150K (idle power)
Wind-up Turns (Landing Config)		1	Turn to left, full aft stick to stall*, no rudder input
		1	Turn to left, full aft stick to stall*, bottom rudder, top aileron
		1	Turn to right, full aft stick to stall*, no rudder input
		1	Turn to right, full aft stick to stall, bottom rudder, top aileron
Wind-up Turns (Cruise Config)		2	Left and right wind-up turns to limit load factor or C_L max*, then reduce power to stall aircraft, no rudder input
		2	Left and right wind-up turns to limit load factor or C_L max*, then reduce power to stall aircraft, hold bottom rudder and top aileron
Immelmanns			Normal Immelmann maneuvers to be performed with following restrictions:

SPIN SUSCEPTIBILITY FLIGHT TEST PROGRAM MANEUVERS

TABLE 1 (3.4.3)

Maneuvers	Configurations	No. of Entries	
Immelmanns (cont'd)		2	Airspeed at top of maneuver approximately 200 K (left and right rollout)
		2	Airspeed at top of maneuver approximately 100 K (left and right rollout)
1g Stalls		2	Full aft stick, no rudder or aileron (power as required)
		2	Full aft stick, rudder and aileron to hold heading and wings level (power as required)

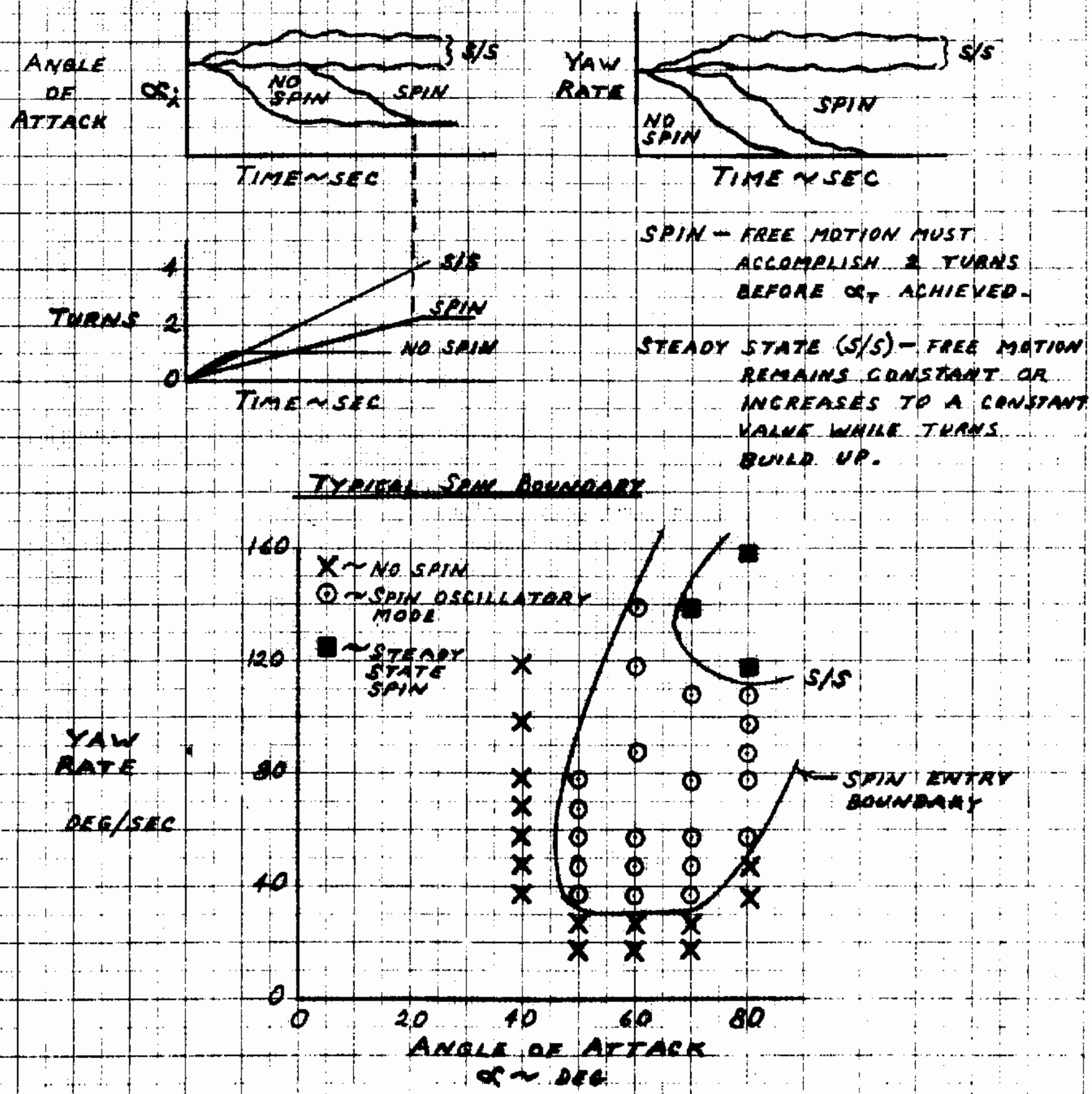
*Horizontal tail input shall be at maximum rate from trim position.

NOTE: 1) All control inputs shall be held in that position for a prolonged time.
2) These maneuvers are not selected short of what will spin the F-5.

SPIN SUSCEPTIBILITY FLIGHT TEST PROGRAM MANEUVERS

TABLE 1 (3.4.3)
(continued)

F-5A ANALYTICAL SPIN STUDY



SPIN RECOVERY

FIGURE 1 (3.4.3)

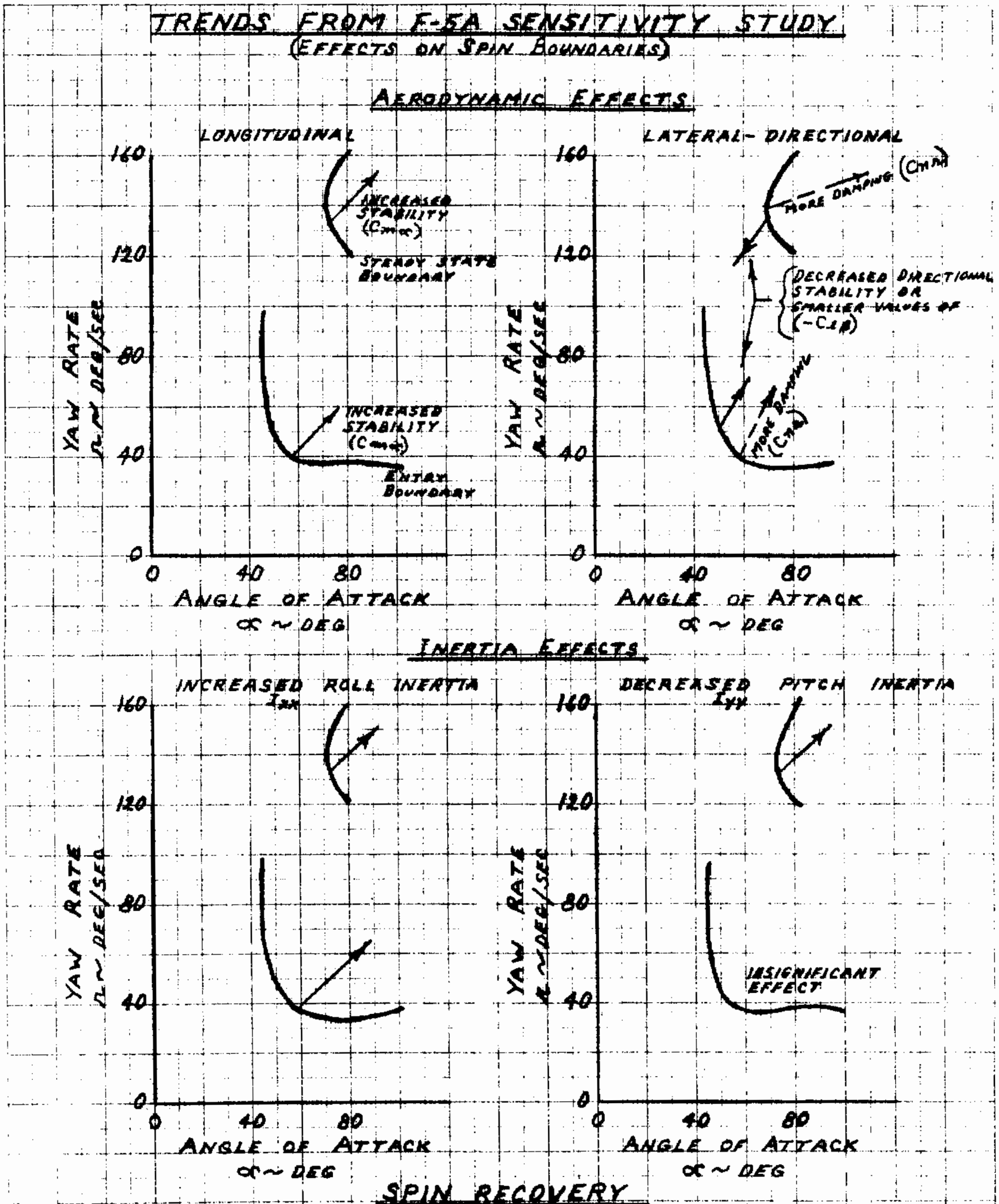

TRENDS FROM F-5A SENSITIVITY STUDY
(EFFECTS ON SPIN BOUNDARIES)
AERODYNAMIC EFFECTS
LONGITUDINAL
INCREASED STABILITY
STEADY STATE BOUNDARY
ENTRY BOUNDARY
LATERAL - DIRECTIONAL
MORE DAMPING
DECREASED DIRECTIONAL STABILITY OR SMALLER VALUES OF
YAW RATE
ANGLE OF ATTACK
INERTIA EFFECTS
INCREASED ROLL INERTIA
DECREASED PITCH INERTIA
INSIGNIFICANT EFFECT
SPIN RECOVERY
0
40
80
120
160

FIGURE 2 (3.4.3)

Requirement

Paragraph 3.4.4 Roll-pitch-yaw coupling. For Class I and IV airplanes in rudder-pedal-free, elevator-control-fixed, maximum-performance rolls through 360 degrees, entered from straight flight or from turns, pushovers, or pullups ranging from 0g to 0.8 n_L, the resulting yaw or pitch motions and sideslip or angle of attack changes shall neither exceed structural limits nor cause other dangerous flight conditions such as uncontrollable motions or roll autorotation. During combat-type maneuvers involving rolls through angles up to 360 degrees, the yawing and pitching shall not be so severe as to impair the tactical effectiveness of the maneuver. These requirements define Level 1 and Level 2 operation. For Class II and Class III airplanes, these requirements apply in rolls through 120 degrees.

Comparison

F-5 roll coupling flight test results were used to compare F-5 characteristics with the requirements of this paragraph. These roll-pitch-yaw coupling tests consisted of 360 degree rolls using maximum aileron deflections with maximum rates of control inputs and roll entry normal load factors ranging from 0g to 2/3 n_L.

The data obtained from these flight tests were plotted as Roll Entry Normal Load Factor versus Peak Normal Load Factor experienced during the roll. These data are presented in Figures 1 (3.4.4) through 4 (3.4.4). Since flight test data were not available for load factors above 2/3 n_L, the data were linearly extrapolated to 0.8 n_L for comparison with the paragraph requirements.

In general, these data indicate that if the 360 degree rolls were entered at 0.8 n_L, the resulting peak normal load factor would, for some flight conditions, slightly exceed the structural limit. However, it is not expected to induce dangerous flight conditions such as uncontrollable motions or roll autorotation. Figures 5 (3.4.4) through 7 (3.4.4) present typical time histories of these maneuvers. These data indicate that the yawing and pitching resulting from the maneuvers are moderate and the tactical effectiveness is not impaired.

Resolution

Two basic concepts have to be considered regarding this paragraph. These are the purpose and the consequence of the requirement. If the purpose is to design a combat type airplane (Class IV) that is superior, then its restrictions in roll (a primary air combat maneuver) have to be minimized. The consequence then will be that the requirement for such a design will have to be sufficiently severe to permit roll entry at very high angles of attack such as at the left boundary ($C_{L_{max}}$) of the V-n diagram. In this region, 360 degree rolls are conducted to hold and maintain high angles of attack for a prolonged period of time. This produces much-needed drag that pilots, in air combat situations, required for deceleration without heading change. This is done to preclude overshooting a threat airplane during tracking.

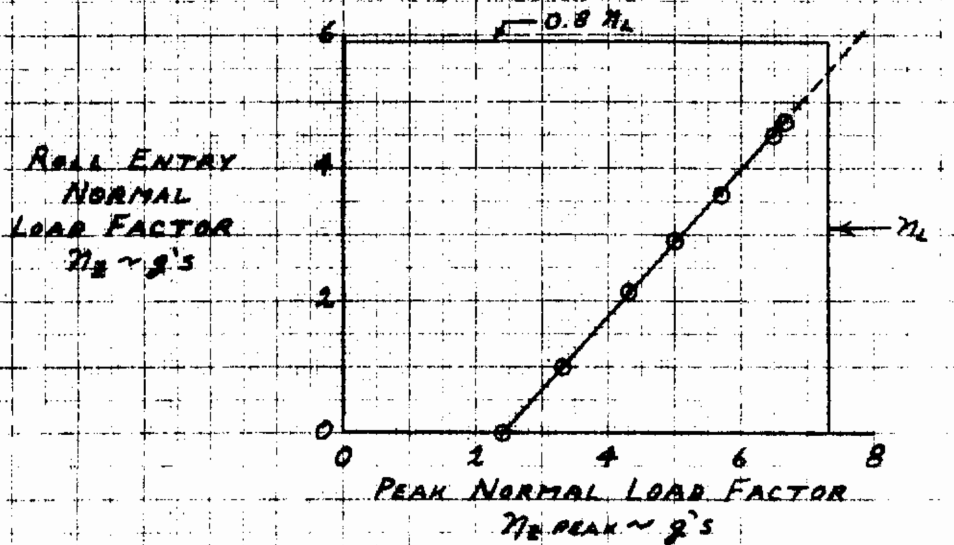
As currently written, the requirement places more emphasis on structural limits than on resulting dangerous flight conditions due to roll at high angles of attack, especially in the $C_{L_{max}}$ region where the structural limits are hardly ever exceeded. Yet, at the $C_{L_{max}}$, for some airplanes, if roll is attempted, spin or uncontrollable motions will result. Consequently, these airplanes become restricted in this region and lack excellence in air combat capability.

Recommendation

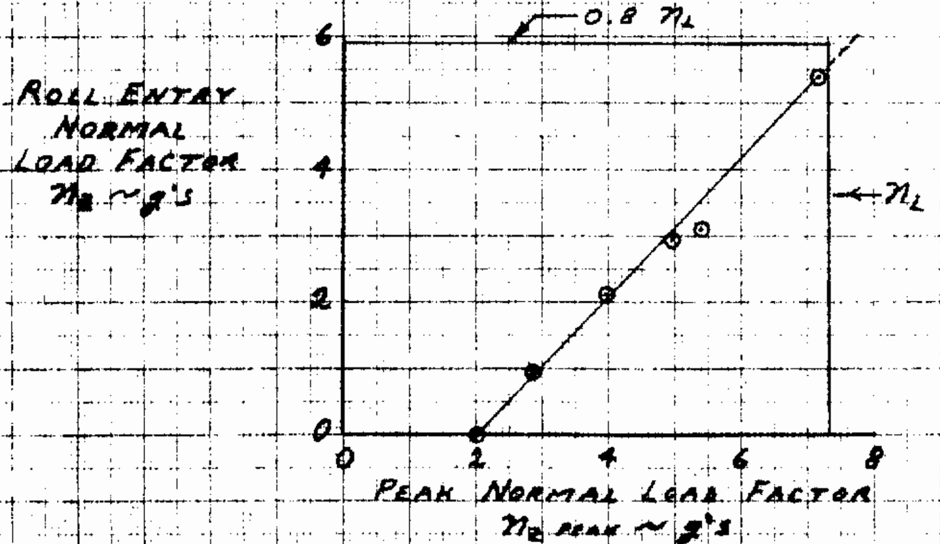
Replace "0.8 n_L" by "$C_{L_{max}}$ or 0.8 n_L, whichever occurs first," and after autorotation add "or spin."

F-5 FLIGHT TEST ROLL COUPLING RESULTS
360 DEGREE ROLLS (MAXIMUM AILERON)
FLIGHT PHASE CATEGORY A

C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
23.0% MAC		10,000 FEET	0.8



C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
22.5% MAC		20,000 FEET	0.9

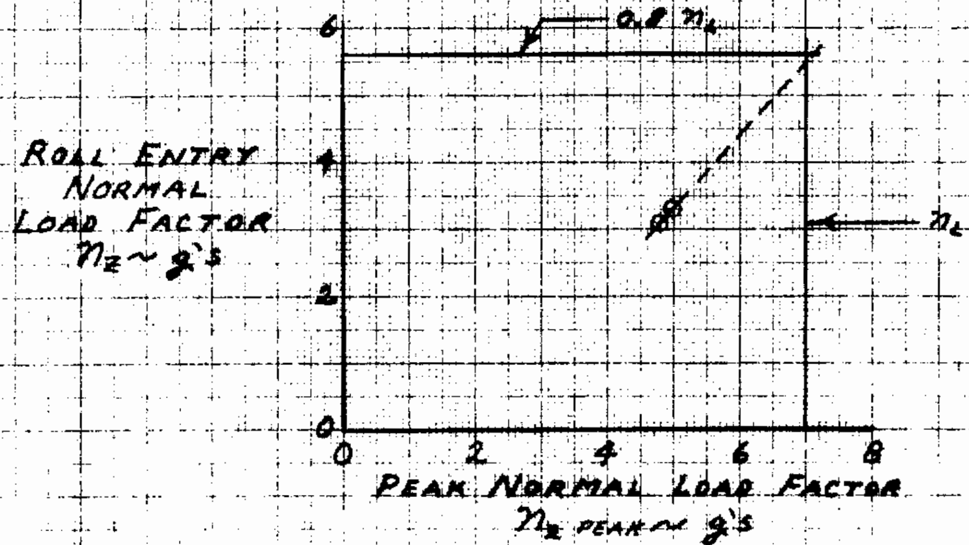


ROLL-PITCH-YAW COUPLING

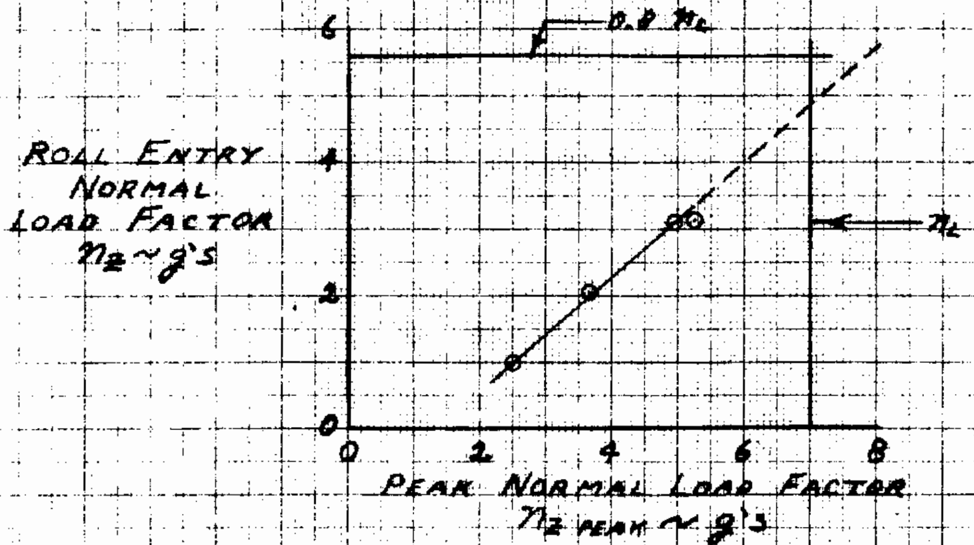
FIGURE 1 (3.4.4)

F-5 FLIGHT TEST ROLL COUPLING RESULTS
360 DEGREE ROLLS (MAXIMUM AILERON)
FLIGHT PHASE CATEGORY A

C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
23.0% MAC		20,000 FEET	0.8



C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
22.5% MAC		20,000 FEET	0.9

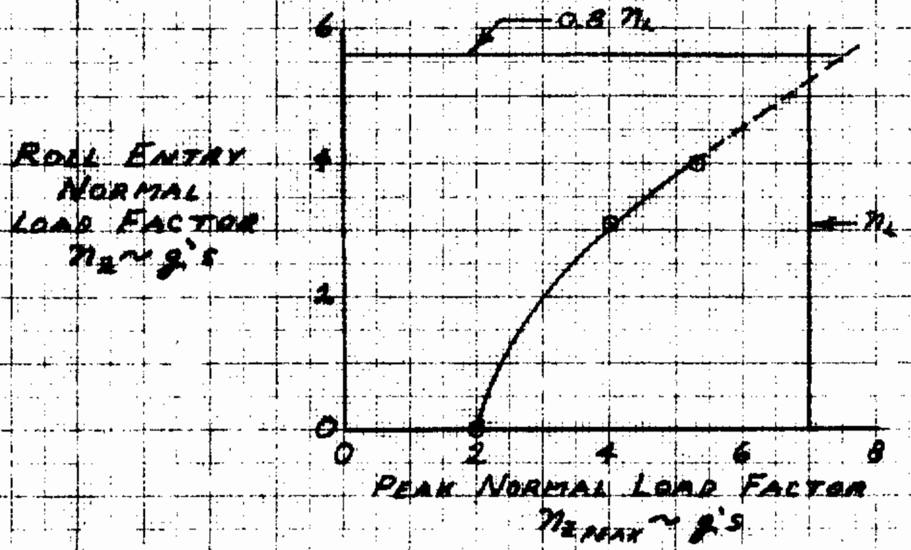


ROLL-PITCH-YAW COUPLING

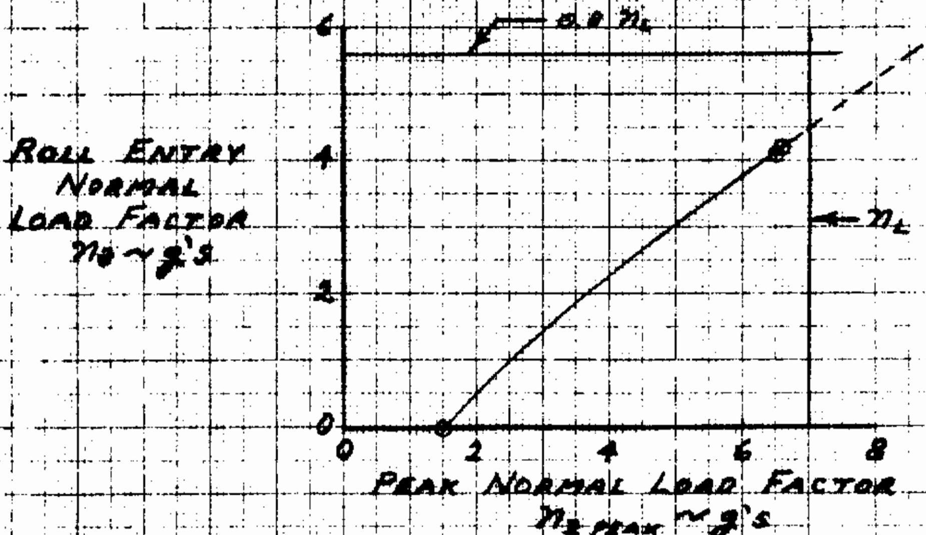
FIGURE 2 (3.4.4)

F-5 FLIGHT TEST ROLL COUPLING RESULTS
360 DEGREE ROLLS (MAXIMUM AILERON)
FLIGHT PHASE CATEGORY A

C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
22.0 % MAC		10,000 FEET	0.8



C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
22.0% MAC		20,000 FEET	0.9

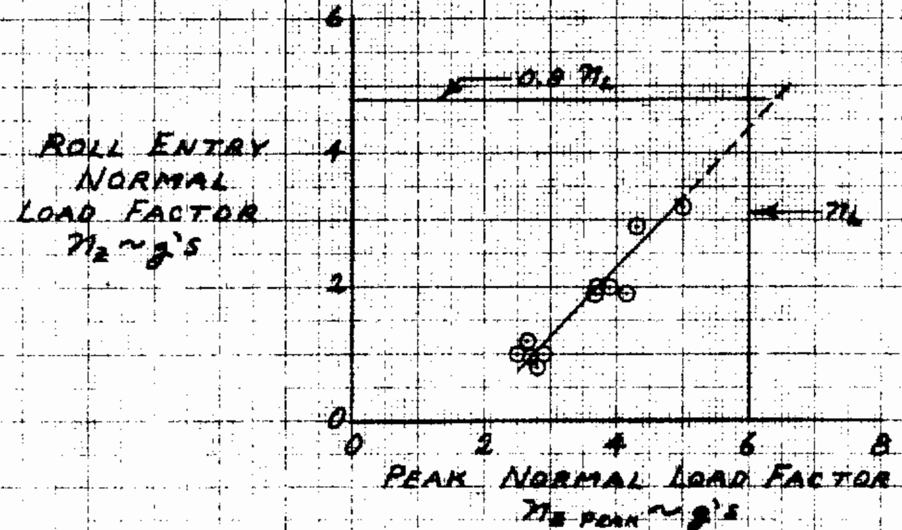


ROLL-PITCH-YAW COUPLING

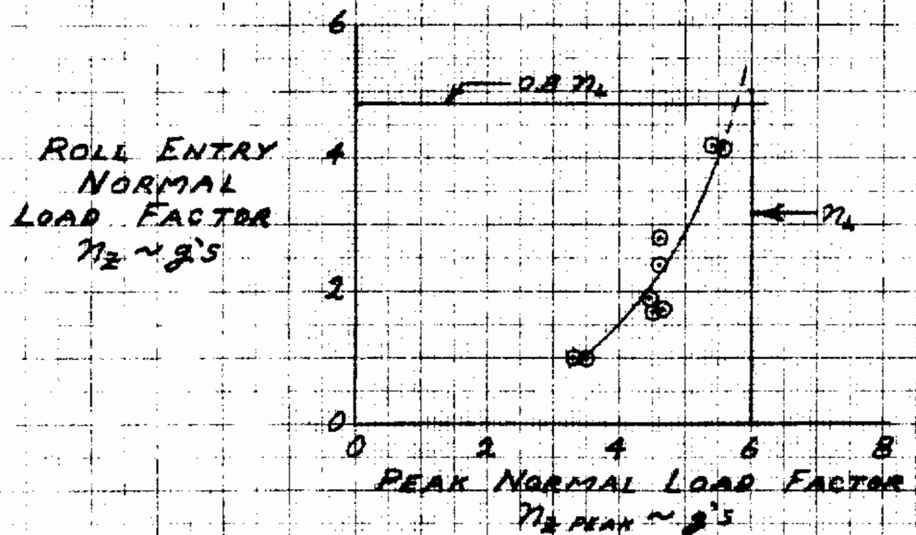
FIGURE 3 (3.4.4)

F-5 FLIGHT TEST ROLL COUPLING RESULTS
360 DEGREE ROLLS (MAXIMUM AILERON)
FLIGHT PHASE CATEGORY A

C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
22.5% MAC		10,000 FEET	0.8



C.G. POSITION	CONFIGURATION	ALTITUDE	MACH NUMBER
20.2% MAC		10,000 FEET	0.8

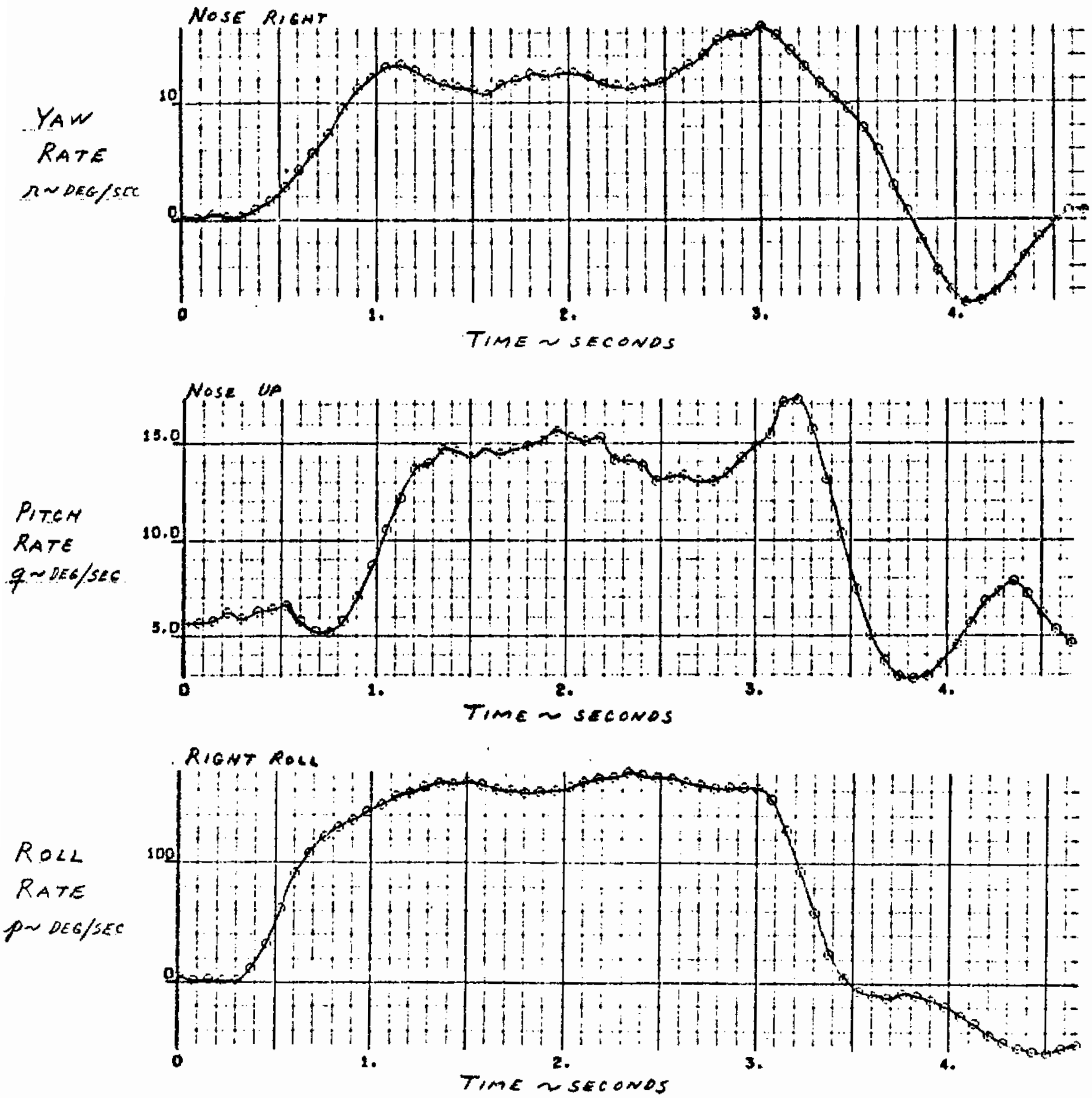


ROLL-PITCH-YAW COUPLING

FIGURE 4 (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL

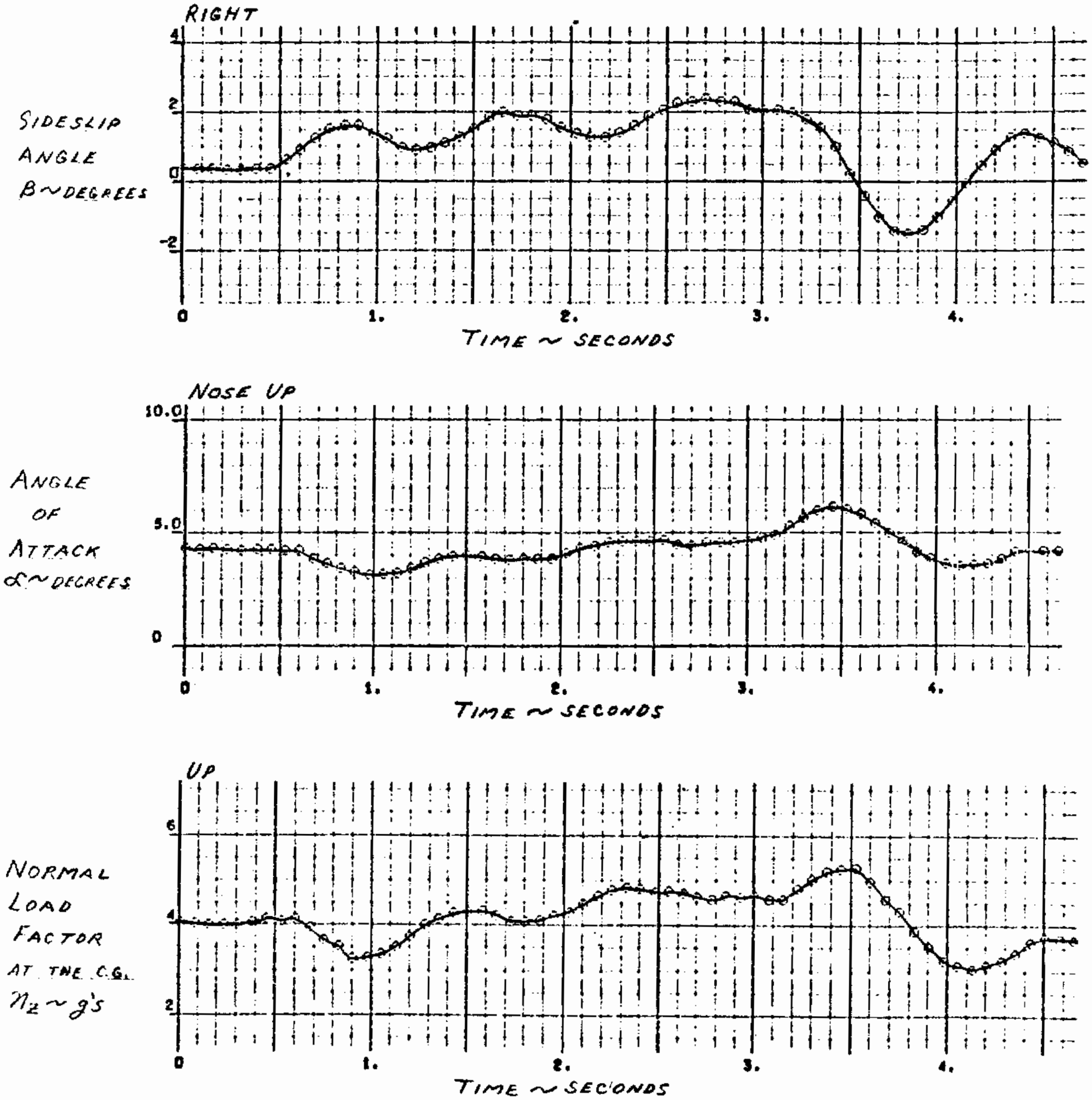
CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
	10,000 FT.	0.82	11,450 LBS.	22.4% MAC



ROLL-PITCH-YAW COUPLING

FIGURE 5a (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL

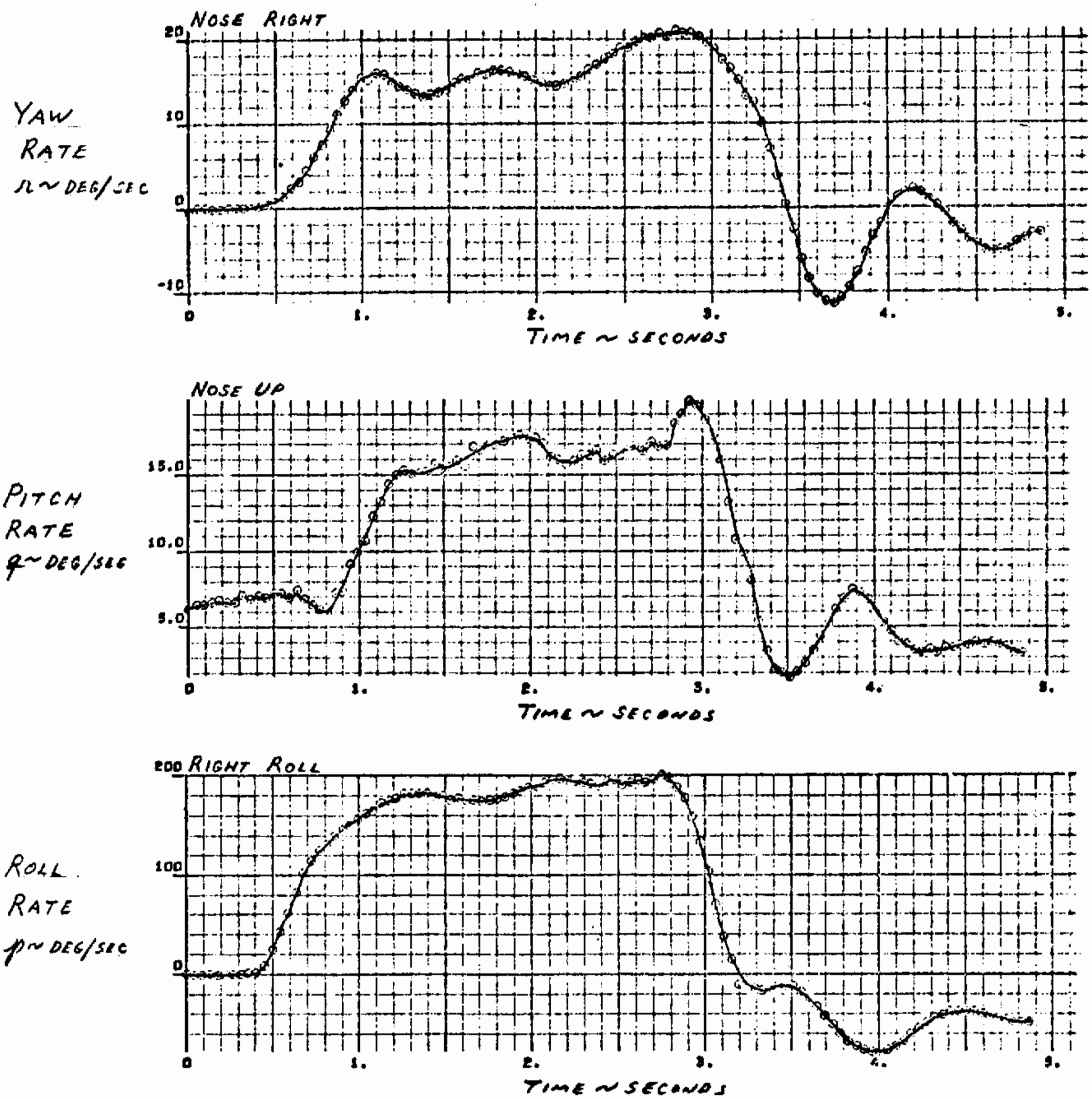


ROLL-PITCH-YAW COUPLING

FIGURE 5b (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL

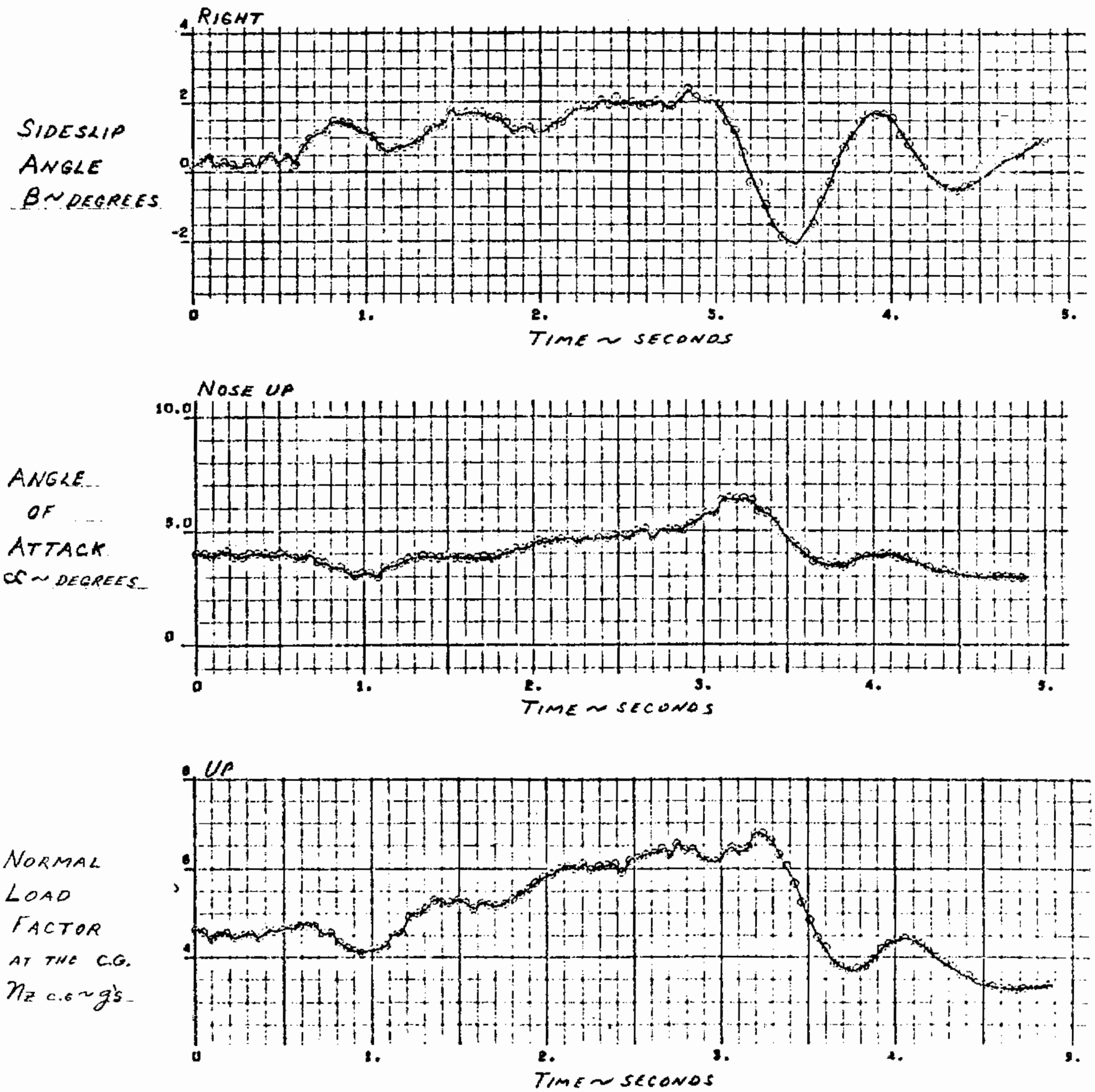
CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
⊢——⊣	10,000 FT.	0.85	11,480 LBS.	22.6% MAC



ROLL-PITCH-YAW COUPLING

FIGURE 6a (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL

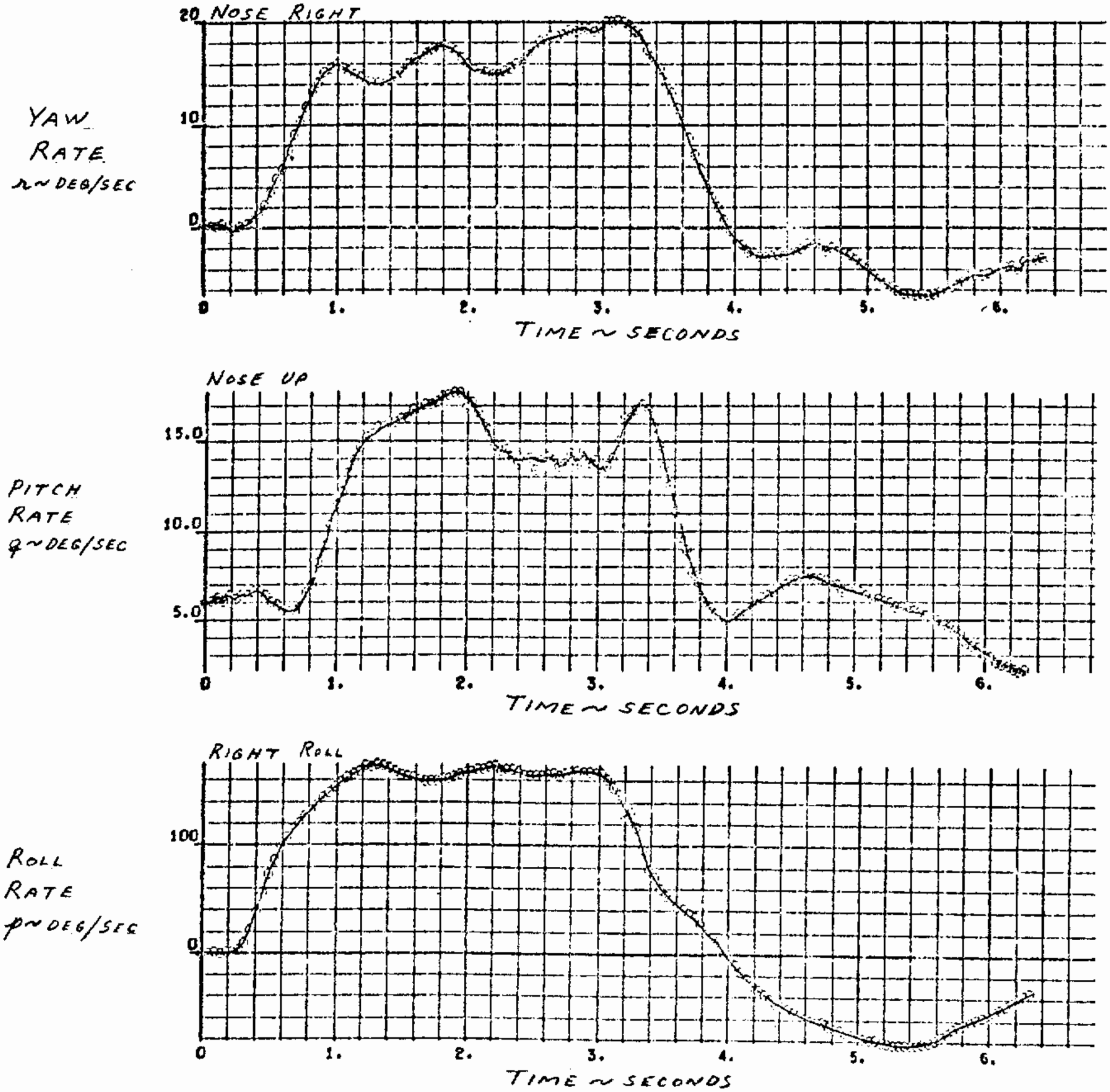


ROLL-PITCH-YAW COUPLING

FIGURE 6.b (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL

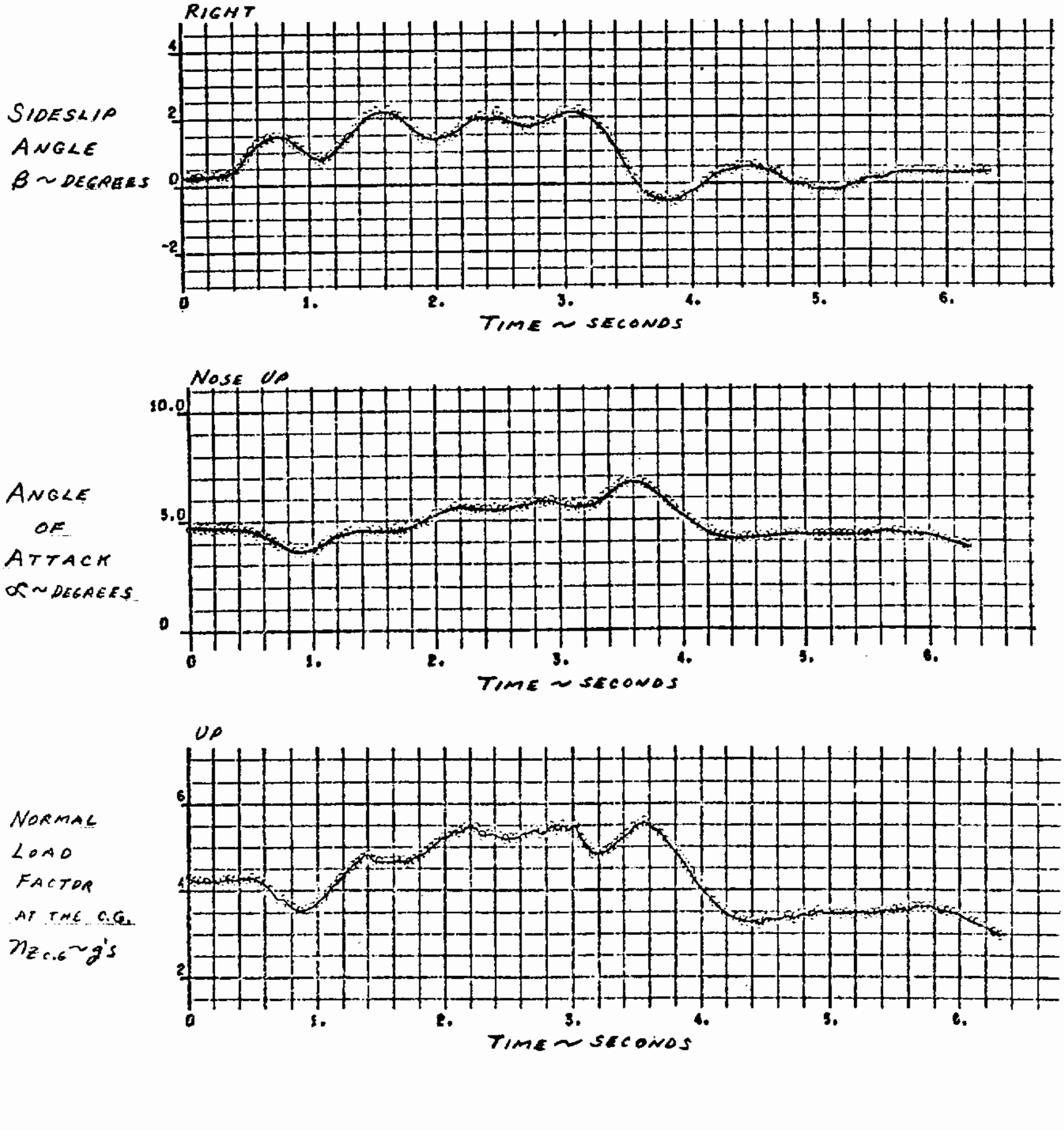
CONFIGURATION	ALTITUDE	MACH NUMBER	WEIGHT	C.G. POSITION
	10,000 FT.	0.81	11,570 LBS.	19.8 % MAC



ROLL-PITCH-YAW COUPLING

FIGURE 7a (3.4.4)

F-5 FLIGHT TEST TIME HISTORY OF A 360 DEGREE ROLL



ROLL-PITCH-YAW COUPLING

FIGURE 7b (3.4.4)

Requirement

Paragraph 3.4.5 Control harmony. The elevator and aileron force and displacement sensitivities and breakout forces shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

Comparison

The F-5 exhibits control harmony in agreement with this paragraph. Breakout forces are 2 pounds for the horizontal tail control and 1 pound for the aileron control. The linkages are arranged such that the longitudinal stick motion produces no aileron motion and the lateral stick motion produces no horizontal tail motion. The stick force sensitivities are 7 pounds per inch for the horizontal tail control and 3 pounds per inch for the aileron control.

The force and displacement sensitivities were optimized during the flight test program. Stick-to-surface ratios are nonlinear for both aileron and horizontal tail. This is done to optimize the variations with airspeed of airplane responses to control inputs.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.5.1 Control force coordination. The cockpit control forces required to perform maneuvers which are normal for the airplane should have magnitudes which are related to the pilot's capability to produce such forces in combination. The following control force levels are considered to be limiting values compatible with the pilot's capability to apply simultaneous forces:

Type Control	Elevator	Aileron	Rudder
Center-stick	50 pounds	25 pounds	175 pounds
Wheel	75 pounds	40 pounds	175 pounds

Comparison

The F-5 utilizes full power controls with artificial feel for all three primary control systems; i.e., horizontal tail, ailerons and rudder. The control forces for all normal maneuvers are equal to or less than the limiting values of this paragraph.

Resolution

None

Recommendation

None

4. QUALITY ASSURANCE

This section of the specification has been reviewed and is considered to be very adequate.

The F-5 airplane design, development, and flight testing spanned the time period of 1962 to 1965. Flying qualities demonstration was conducted in compliance with the prevailing military specification.

The quality assurance specification was unlike the present one; nevertheless; the F-5 design and test conditions guidelines very nearly resembled those of the present specification.

5. PREPARATION FOR DELIVERY

This section has never been applicable to this specification. Yet, in all revisions subsequent to the first writing of the specification, this section reappeared with the phrase "not applicable."

Since this section has not served any obviously meaningful purpose, it is suggested that it be deleted from the specification.

6. NOTES

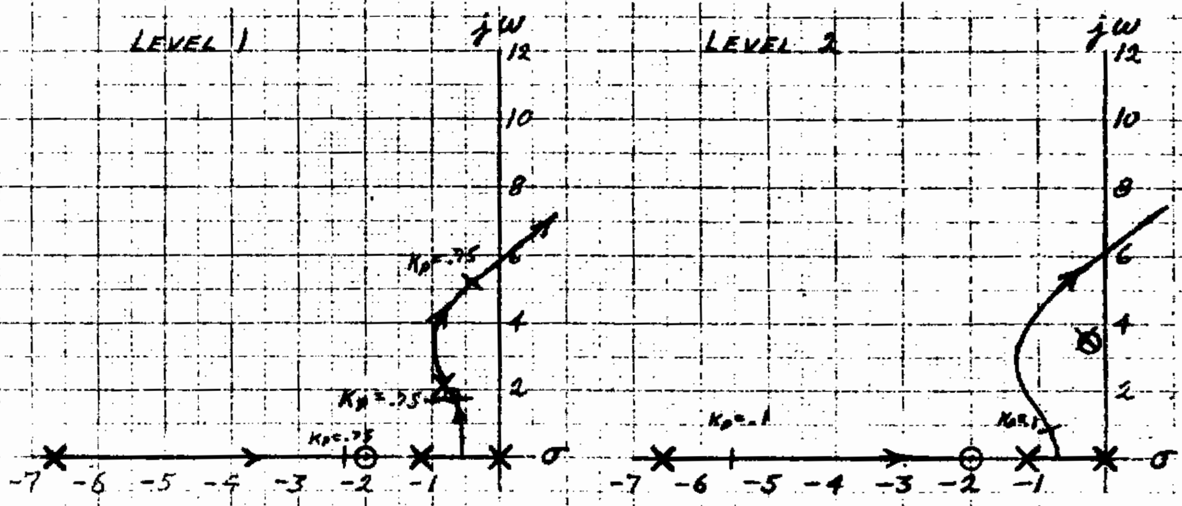
This section of the specification has been reviewed and is considered to be adequate except for the definition of "n" and "n/α" which appear in Section 6.2.5.

The specification defines "n" as "normal acceleration or normal load factor, measured at the c.g." Normal acceleration and normal load factor are interchanged freely in the definition and throughout the specification. The axis system is not defined, such as, normal to the flight path axis or to the body axis of the airplane.

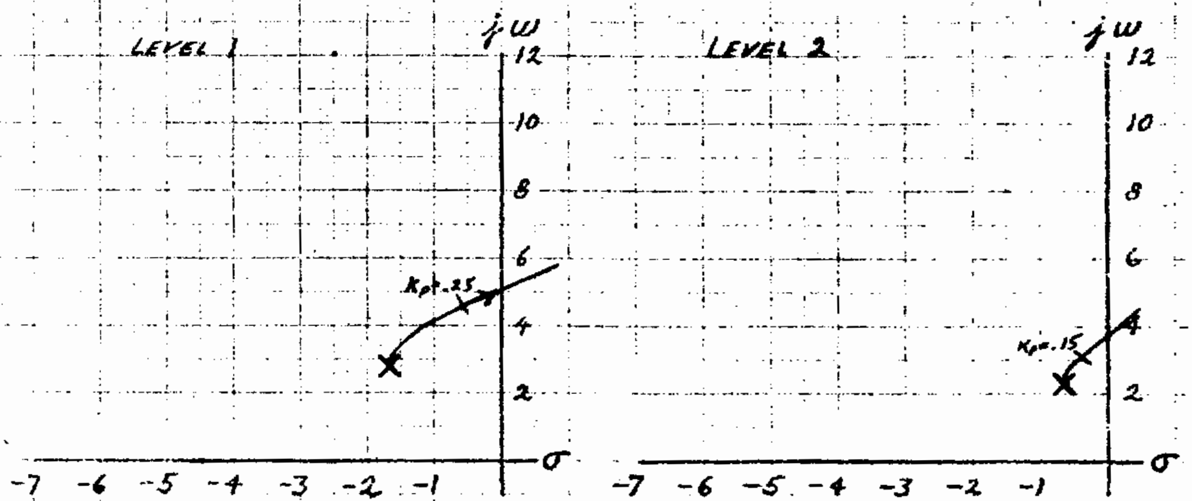
It is suggested that "n" be defined uniformly as normal load factor in the body axis system of the airplane at the c.g.

CONFIGURATION	MN	ALTITUDE	FLIGHT PHASE
¤——¤	0.8	30,000 FEET	CATEGORY A

ROOT LOCUS FOR ROLL ANGLE TRACKING, $T_L = .5$



ROOT LOCUS FOR PITCH ANGLE TRACKING, $T_L = .5$

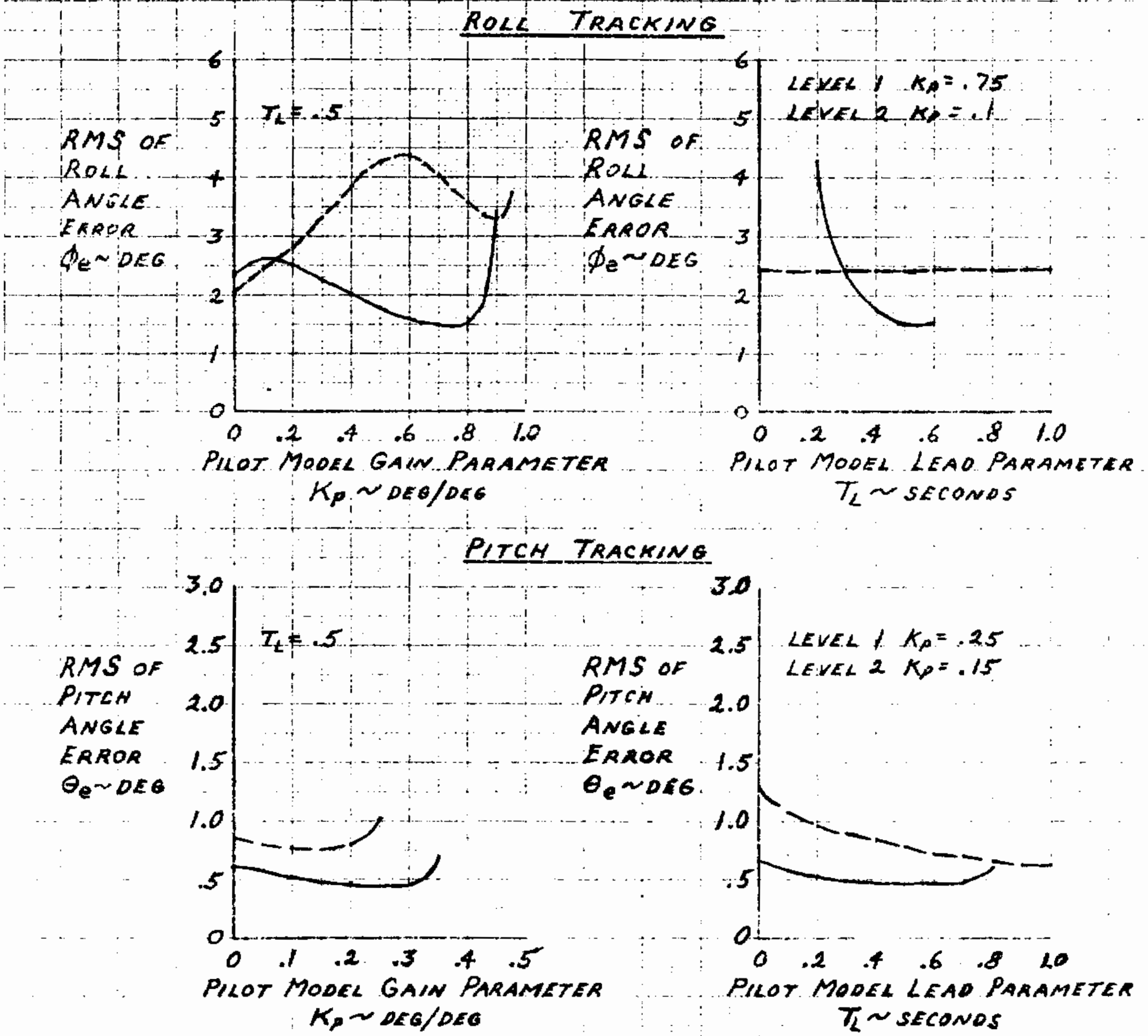


APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 41 (3.7.5)

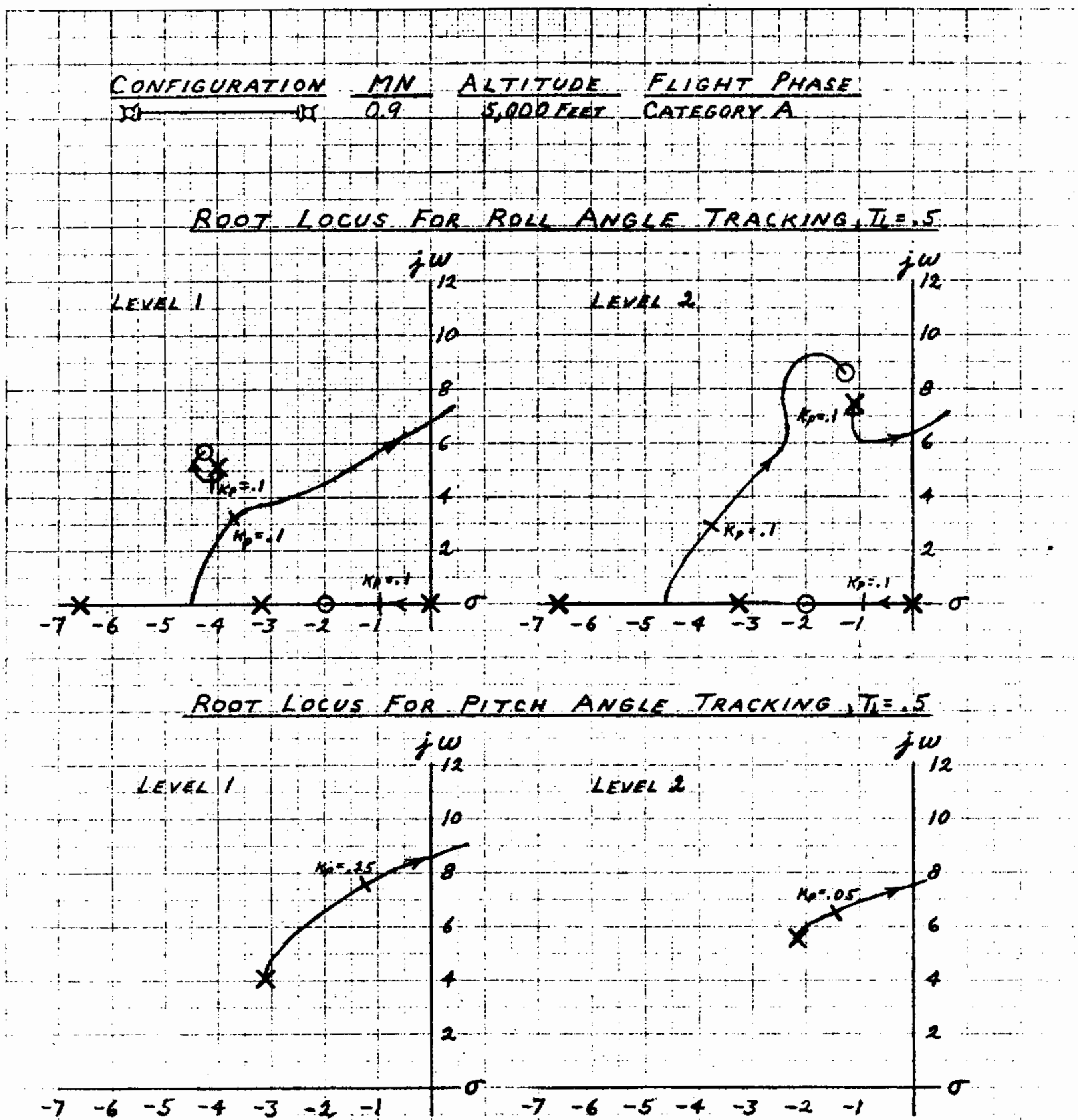
PILOT LEAD AND GAIN VARIATION

CONFIGURATION	MN	ALTITUDE	FLIGHT PHASE	LEGEND
	0.8	30,000 FEET	CATEGORY A	—— LEVEL 1 – – – LEVEL 2



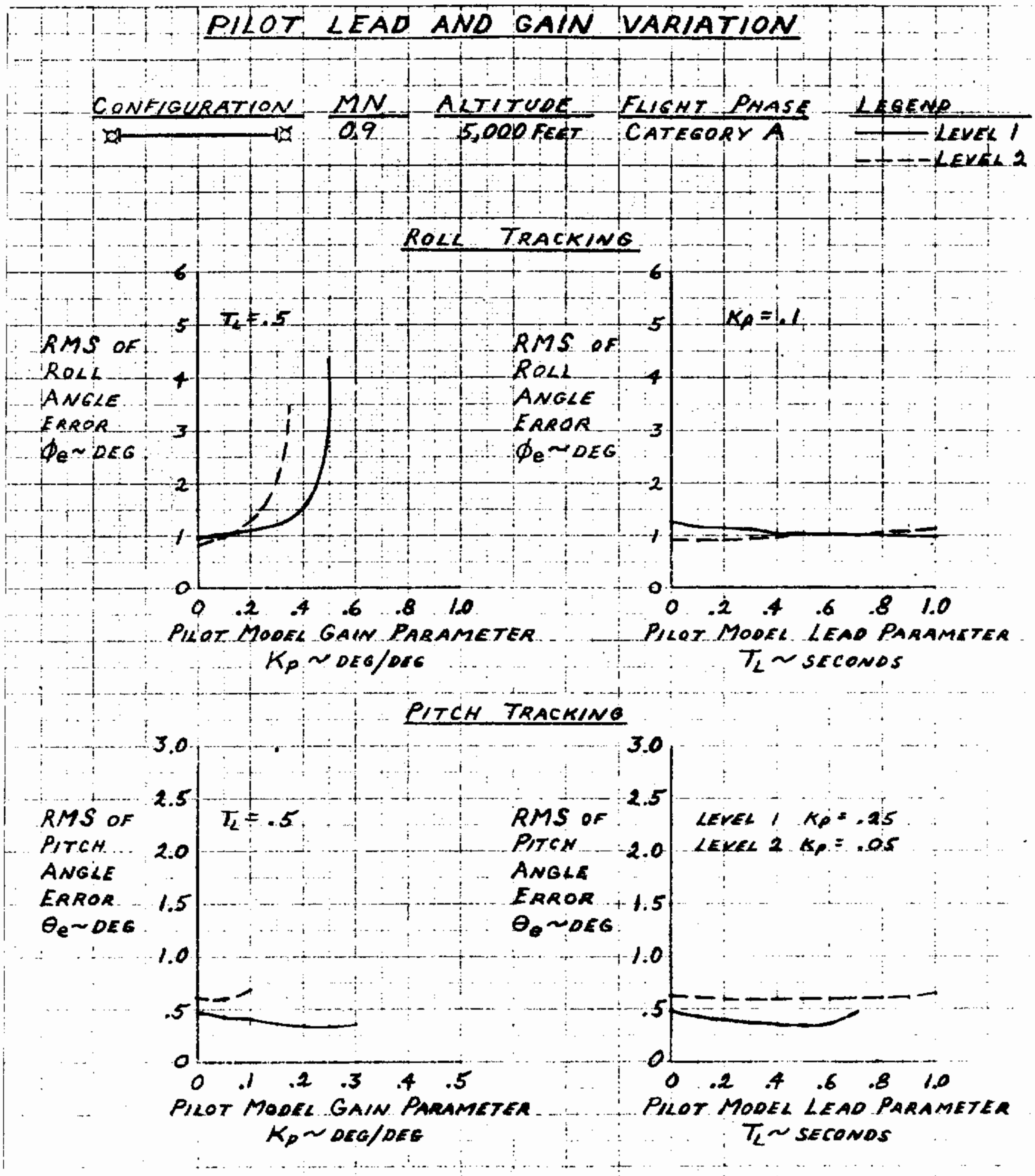
APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 4a (3.7.5)



APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 3l (3.7.5)

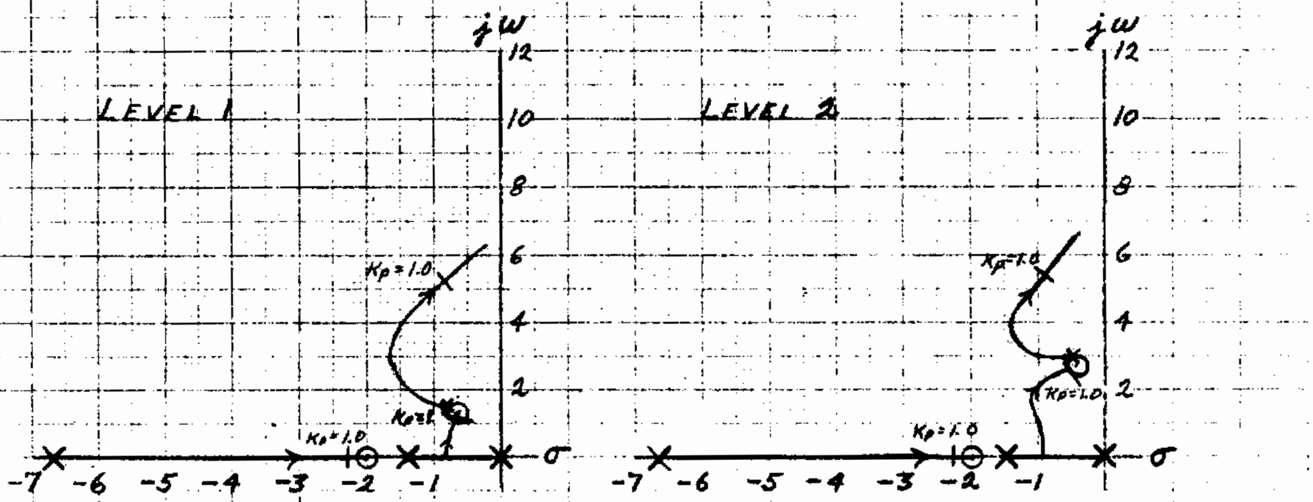


APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

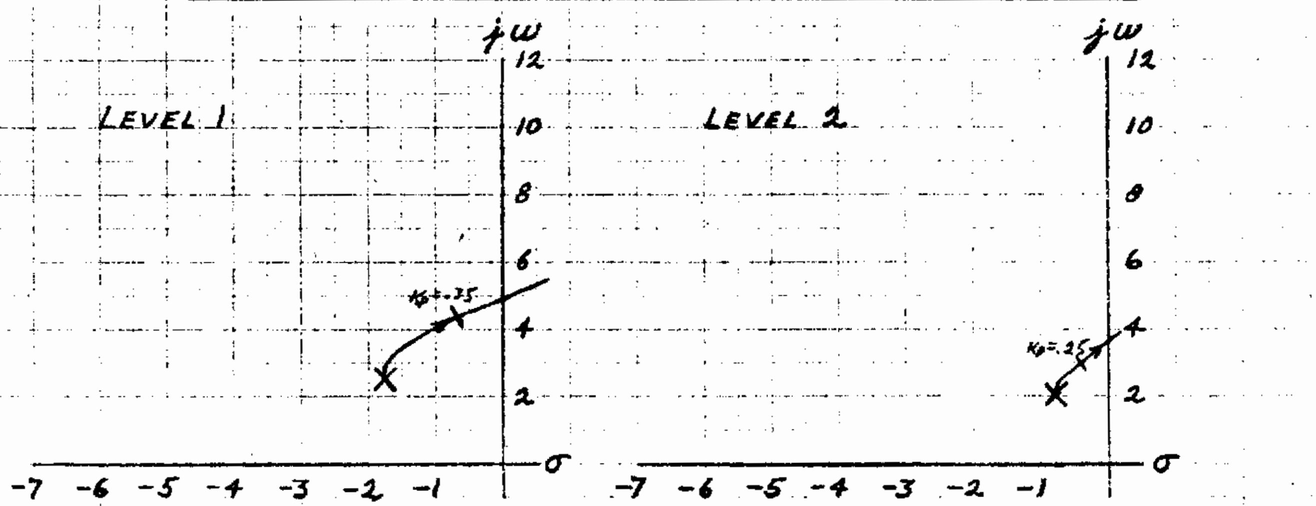
FIGURE 3a (3.7.5)

CONFIGURATION	MN	ALTITUDE	FLIGHT PHASE
	0.4	5,000 FEET	CATEGORY C

ROOT LOCUS FOR ROLL ANGLE TRACKING, $T_L = .5$

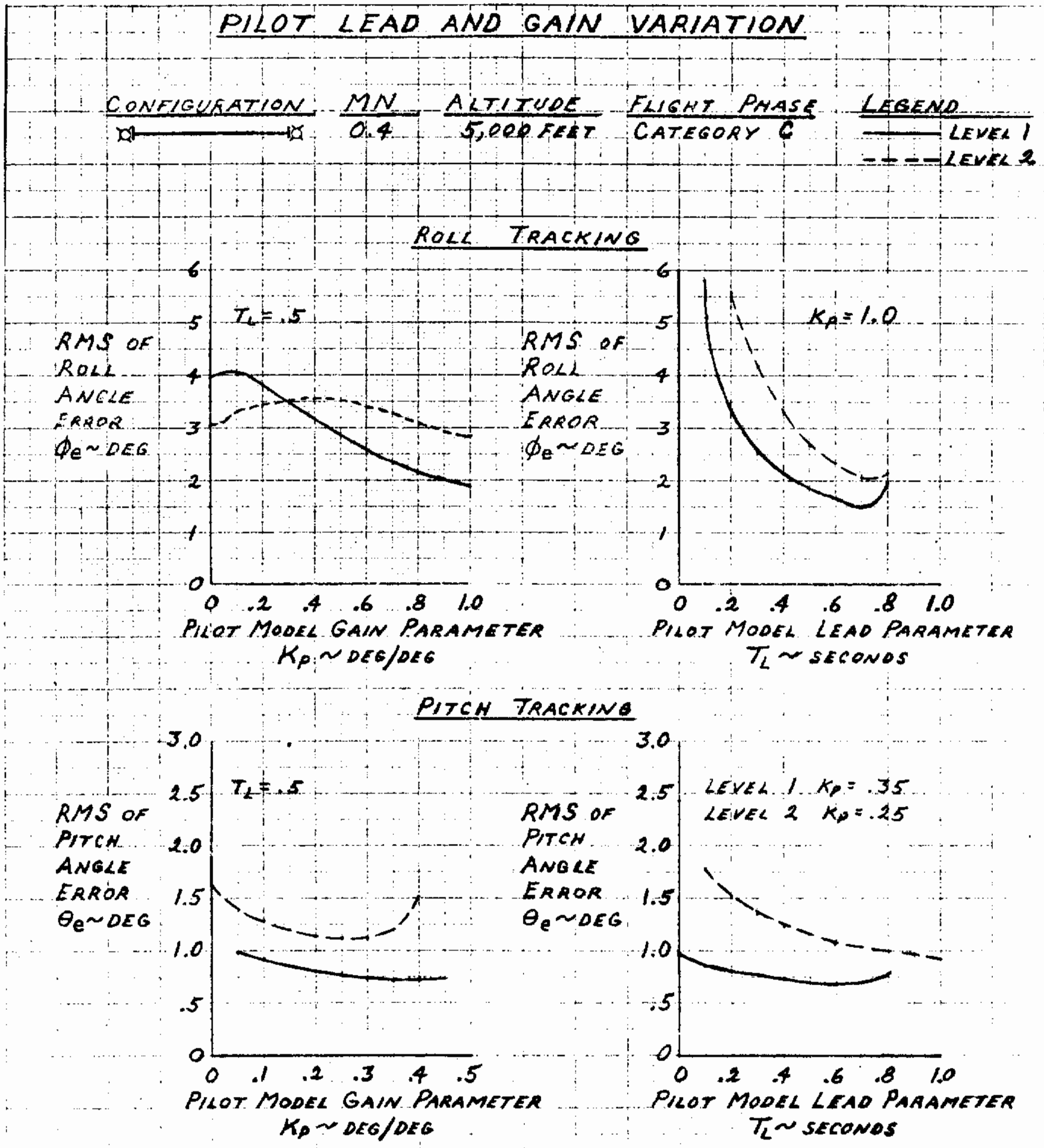


ROOT LOCUS FOR PITCH ANGLE TRACKING, $T_L = .5$



APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

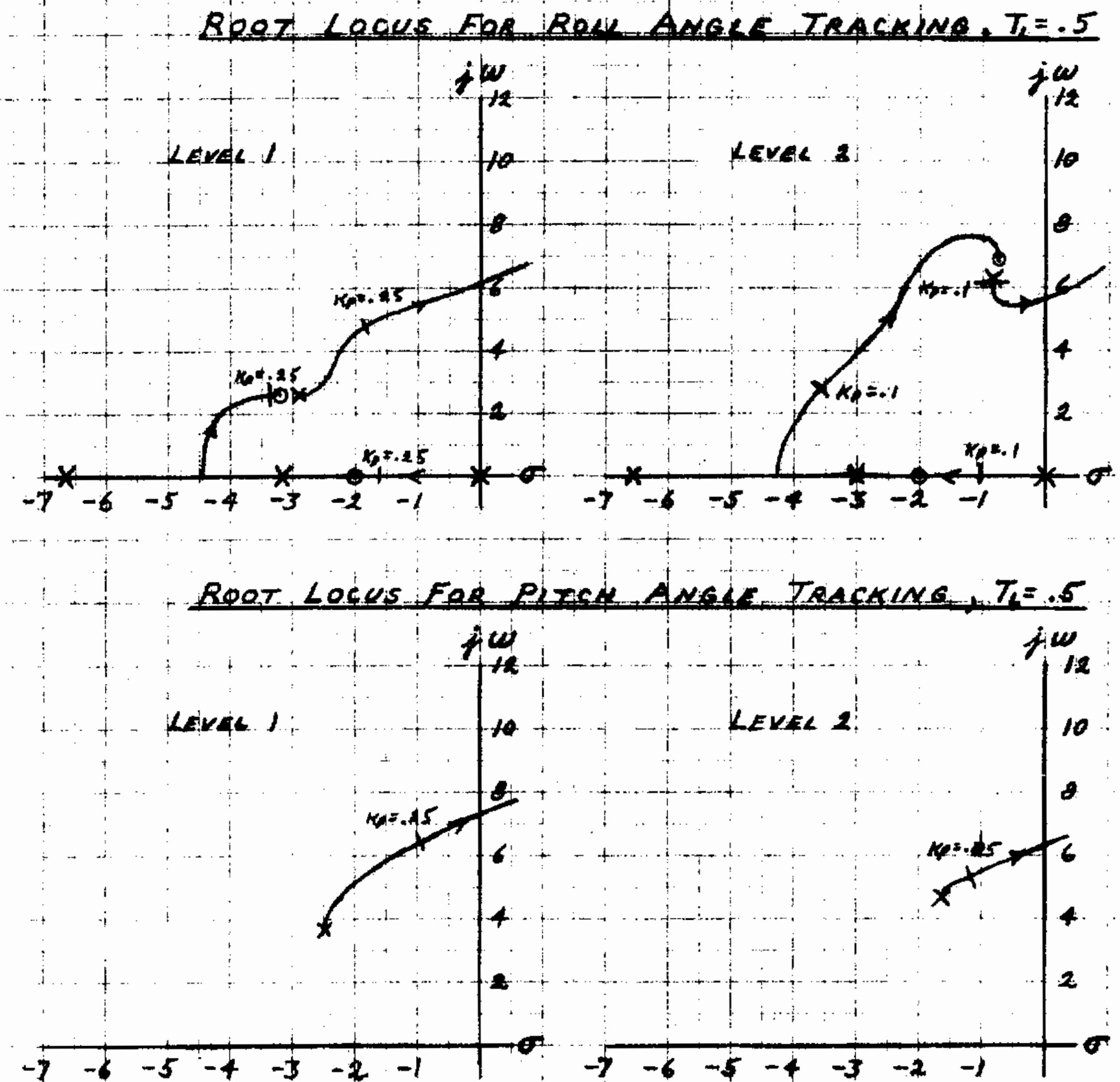
FIGURE 26 (3.7.5)



APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 2a (3.7.5)

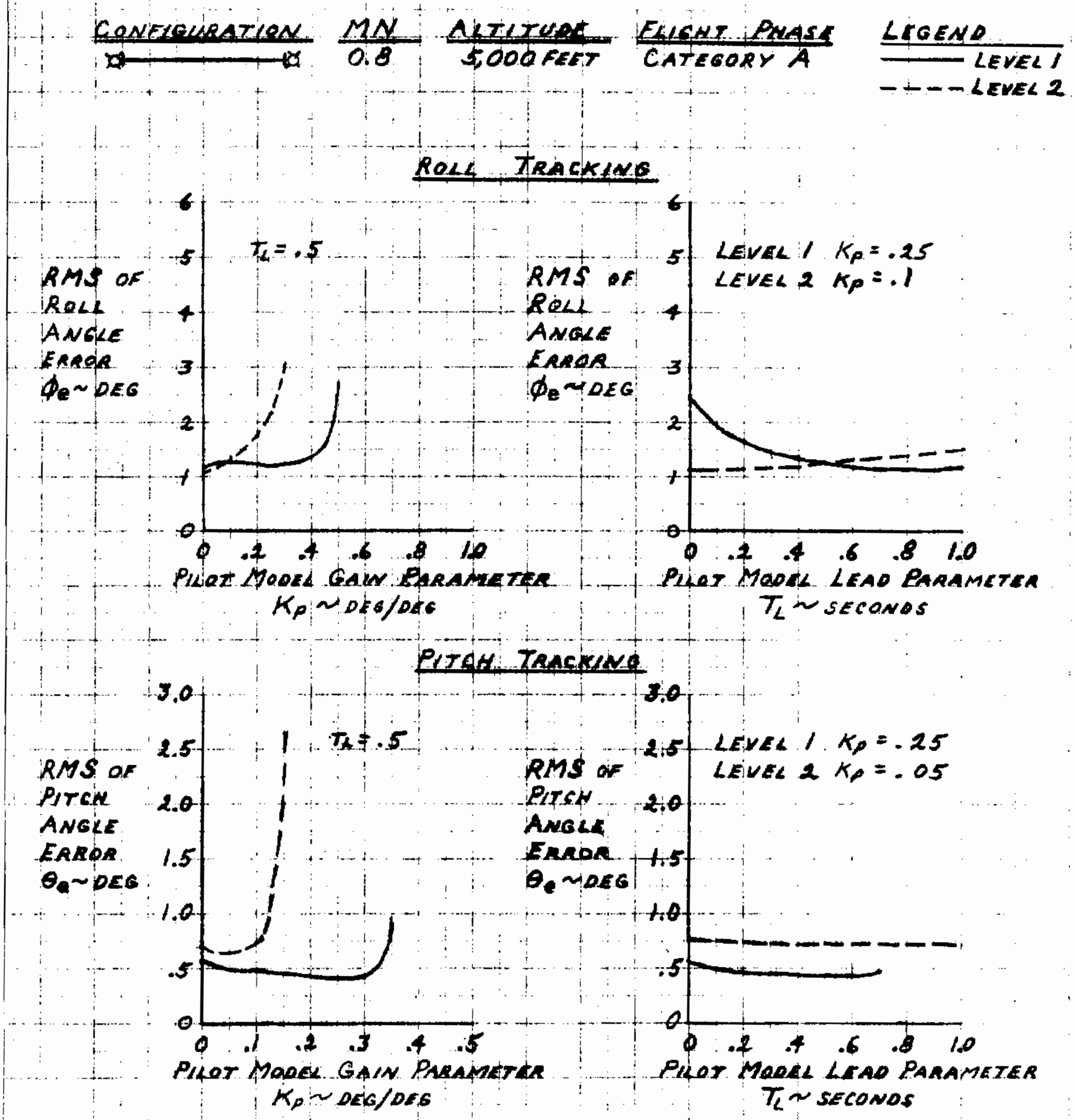
CONFIGURATION	MN	ALTITUDE	FLIGHT PHASE
	0.8	5,000 FEET	CATEGORY A



APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 16 (3.7.5)

PILOT LEAD AND GAIN VARIATION



APPLICATION OF THE TURBULENCE MODELS IN ANALYSES

FIGURE 1a (3.7.5)

To implement this recommendation it is suggested that representative aircraft, that are presently flying, which are known to exhibit outstanding, good, or poor characteristics in calm air command tracking and/or turbulent air horizon tracking in some flight region, be analyzed to provide a background basis from which a practical turbulence criterion can be devised. Until the criterion is established, all new designs should be required to submit this type of analysis for consideration. Such a requirement would assure general awareness of the problem.

For pitch tracking, with operative pitch damper, a pilot gain of 0.25 doubles the short-period frequency but reduces the damping ratio by seventy percent. With inoperative pitch damper, a low 0.05 gain increases the frequency, but again reduces the damping ratio.

Figures 2a (3.7.5) and 2b (3.7.5) present similar results for M = 0.4, Altitude = 5000 feet, Category C. The pilot gain plots indicate that the pilot can reduce the rms roll and pitch errors. The root locus plots again indicate the loss in damping ratio and increase in frequency with pilot gain.

Figures 3a (3.7.5) and 3b (3.7.5) present the results for M = 0.9, Altitude = 5000 feet, Category A. The results indicate that the pilot cannot reduce the roll error and can only achieve a small reduction in pitch error. With operative yaw damper, the roll error root locus plot indicates a different manner of approaching instability for large values of pilot gain.

Figures 4a (3.7.5) and 4b (3.7.5) give results for M = 0.8, Altitude = 30,000 feet, Category A. For roll tracking, with operative yaw damper, the pilot can appreciably reduce the roll error if he will provide 0.5 second lead and sufficient gain. With inoperative yaw damper he cannot improve the stick fixed value. For pitch tracking, the pilot can reduce the pitch error slightly.

Resolution

Because the present paragraph 3.7, Atmospheric disturbances, does not specify a criterion for flying in turbulence, the typical F-5 results presented above have no basis for comparison or need for resolution.

Recommendation

It is recommended that a flying qualities analytical procedure be specified based on the analysis technique whose results are illustrated above. In particular, it is recommended that the comparison analysis for command tracking in calm air be conducted simultaneously. The results can then be easily combined. This combined procedure would nullify the possibility that new airplanes might be designed which would meet the required turbulence specification but possibly at the expense of a reduction in calm air tracking capability.

only the pertinent root-locus branches, which become unstable, are shown. For example, augmentation and phugoid characteristics are not shown.

For the roll angle tracking cases, the bank angle (error) to side gust excitation transfer function denominator roots together with the stable Pade' denominator root are denoted by small crosses (i.e., poles). Similarly, the bank angle to side gust excitation transfer function complex numerator and the pilot model numerator leadterm are denoted by small circles (i.e., zeros).

The spiral mode real pole is always slightly stable. The Pade' denominator negative real pole is always plotted at (-2/0.3) or -6.7. The roll mode real pole location varies between -3.3 and -1.2 depending upon the flight condition. The pilot model lead zero is always plotted at -2.0. The Dutch Roll pole and roll angle error numerator zero are both located in the complex plane depending upon their respective damping ratio and positive undamped frequency values. Although the pole to zero closures vary depending on the flight condition considered, the value of pilot model gain K_p selected is noted by a short bar on each root locus branch.

For the pitch angle tracking cases, only the pitch angle (error) to vertical gust excitation of the short period denominator pole has been plotted as it is pertinent relative to where and how the model pilot, closed-loop becomes unstable. The selected pilot model gain is noted by a small bar on the root locus branch.

Figure 1a (3.7.5) presents calculated rms roll and pitch tracking results for M = 0.8, Altitude = 5000 feet, Category A. For roll tracking the upper left-hand plot indicates that even if the pilot uses a near-optimum T_L value of 0.5 second, he cannot reduce the roll angle error below the stick-fixed value. For Level 1, a reasonable value for pilot gain is approximately 0.25 and for Level 2, the very small value of 0.1 second is appropriate. For these two pilot gain values, the upper right-hand plot verifies that 0.5 second was a reasonable assumption for the pilot lead value used in the calculations for the left-hand plot.

For pitch tracking, the lower left-hand plot indicates that with 0.5 second lead the pilot can reduce the rms pitch error value if he uses a gain of 0.25, approximately, for Level 1. For Level 2 a slight improvement is obtained with 0.05 gain. The lower right-hand plots verify that 0.5 second lead was reasonable.

Figure 1b (3.7.5) presents root locus plots which indicate the pilot gain values selected for the results presented in Figure 1a. These plots indicate the stability margin that is obtained and the shift in damped natural frequency and total damping that occur as the pilot gain is varied. For roll tracking, with operative yaw damper, a pilot gain of 0.25 reduces the Dutch Roll damping ratio by sixty percent and doubles the damped frequency. With inoperative damper, a small 0.1 gain slightly increases the low Dutch Roll damping ratio and slightly reduces the frequency.

The model pilot transport delay in bank angle was 0.3 second, represented by the first order Pade' approximation

$$\exp(-0.3s) = \frac{-\left(s - \frac{2}{0.3}\right)}{\left(s + \frac{2}{0.3}\right)}$$

In pitch the model pilot transport delay, also represented by the Pade' approximation, was taken to be 0.45 second. The variations of closed loop performance and system characteristics with pilot parameters are presented as Figures 1 (3.7.5) through 4 (3.7.5). These figures provide the basis for the discussion of flying in turbulence with the F-5 airplane in a clean configuration with wing tip missiles.

Four level flight conditions were analyzed with damper augmentation operative and inoperative. In each case, the airplane was assumed to be flying in 10 fps rms turbulence and the model pilot's task was to attempt to hold zero bank and pitch angles; i.e., horizon tracking in turbulence. The airplane random response motions in bank angle and pitch angle caused by 10 fps rms gusts can be summarized by the root-mean-square (rms) angular error relative to the horizon.

A nondimensional pilot gain parameter, K_p value of zero, corresponds to a stick-fixed condition whereas finite K_p values imply the model pilot senses the angular error and moves his control stick to correct this error. The model pilot transport delay constant accounts for his reaction delay time, and the lead parameter value is indicative of the pilot's experience and aggressiveness.

The results for each flight condition (for Levels 1 and 2) are presented on two separate pages, Application of the Turbulence Models in Analyses, Figure "a" and Figure "b". For both figures "a" and "b", the roll and pitch tracking results are plotted on the upper and lower portion of each page, respectively.

The left-hand side of the "a" figure presents the rms variation of roll and pitch angle error in degrees versus increasing values of the nondimensional pilot model gain K_p, but for a constant T_L value of 0.5 second. The right-hand side of the "b" figure presents justification for selection of the constant 0.5 second, T_L value by indicating the rms angular error variation with various T_L values for a fixed K_p value. The Level 1 and 2 fixed K_p values, selected from the left-hand plots, usually correspond to the optimum value unless some low, finite value was arbitrarily used because no optimum was indicated.

The "b" figure presents constant 0.5 second T_L value root locus plots which indicate how the pertinent open-loop characteristics were changed as the closed-loop pilot model gain K_p was increased from zero to an unstable value, indicated by one of the root locii branches crossing the imaginary axis. For clarity, The turbulence velocities u_g, v_g, w_g, p_g, q_g, and r_g are then applied to the airplane equations of motion through the aerodynamic terms. For longitudinal analyses u_g, w_g, and q_g gusts should be employed. For lateral-directional analyses v_g, p_g, and r_g should be used. The gust velocity components u_g, v_g, and w_g shall be considered mutually independent (uncorrelated) in a statistical sense. However, q_g is correlated with w_g, and r_g is correlated with v_g. The rolling velocity gust p_g is statistically independent of all the other gust components.

Comparison

The effects of atmospheric disturbances on the F-5 and T-38 flying qualities are summarized in references 9 to 16. Because various methods of analysis have been under development, only the results from the most recent approach as explained in references 15 and 16 will be presented and discussed here.

The Background Information and Users Guide to MIL-F-8785B suggests closed-loop analysis as a method for incorporating the specific turbulence models into an assessment of the effect of turbulence on ride and flying qualities. To determine the suitability of this approach, the Air Force has sponsored two research contracts with Northrop, "Airplane Flying Characteristics in Turbulence", Contract F33615-70-C-1156 and "Flying Qualities Prediction and Evaluation in Turbulence", Contract F33615-71-C-1076. These programs have developed a new approach to pilot-vehicle analysis of flying qualities in calm air and in turbulence. The accuracy and applicability of the flying qualities prediction has been assessed by the large-amplitude moving-base simulation documented in references 15 and 16.

Although the method can be applied to multiloop pilot tasks such as heading and flight-path-angle control, only bank-angle and pitch-angle attitude-hold tasks are considered. Also, the identical procedure can be used to assess the command tracking characteristics in calm air. However, only the tracking characteristics in turbulence are presented here.

The rigid-body aircraft dynamics were represented by equations of motion for all six degrees of freedom laterally and longitudinally. The Dryden form of the v, r, w and q gusts was used where the v and w gusts were each scaled to 10 fps rms. Operative yaw and pitch damper augmentation represents Level 1 cases. Inoperative augmentation represents Level 2 cases. The bank and pitch pilot models were transfer functions, Y_{p_ϕ} and Y_{p_θ}. These expressions contained the model pilot gain parameter, K_p, the pilot lead parameter, T_L, and a pilot transport delay parameter, τ. The parameters K_p and T_L were schematically varied to provide the airplane bank and pitch response data.

Requirement

Paragraph 3.7.5 Application of the turbulence models in analyses. The gust velocities shall be applied to the airplane equations of motion through the aerodynamic terms only, and the direct effect of the gust on the aerodynamic sensors shall be included when such sensors are part of the airplane augmentation system. When using the discrete model, all significant aspects of the penetration of the gust by the airplane shall be incorporated in the analyses. Application of the continuous random model depends on the range of frequencies of concern in the analyses of the airframe. When structural modes are significant, the exact distribution of the gust velocities over the airframe should be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional functions only of x, but w_g shall be considered two-dimensional, a function of both x and y, for the evaluation of aerodynamic forces and moments.

When structural modes are not significant, airframe rigid-body responses may be evaluated by considering uniform gust immersion along with linear gradients of the gust velocities. The uniform immersion is accounted for by u_g, v_g, and w_g defined at the airplane center of gravity. The angular velocities due to the turbulence are equivalent in effect to the airplane angular velocities. These angular velocities are defined (precisely at very low frequencies only) as follows:

$$p_g = -\frac{\partial w_g}{\partial y}$$

$$-\dot{\alpha}_g = q_g = +\frac{\partial w_g}{\partial x}$$

$$v_g = -\frac{\partial v_g}{\partial x}$$

$$\Phi_{p_g}(\Omega) = \frac{\sigma_w^2}{L_w} \frac{0.8\left(\frac{\pi L_w}{4b}\right)^{1/3}}{1+\left(\frac{4b}{\pi}\Omega\right)^2}$$

$$\Phi_{q_g}(\Omega) = \frac{\Omega^2}{1+\left(\frac{4b}{\pi}\Omega\right)^2} \Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = \frac{\Omega^2}{1+\left(\frac{3b}{\pi}\Omega\right)^2} \Phi_{v_g}(\Omega) \quad \text{where } b = \text{wing span}$$

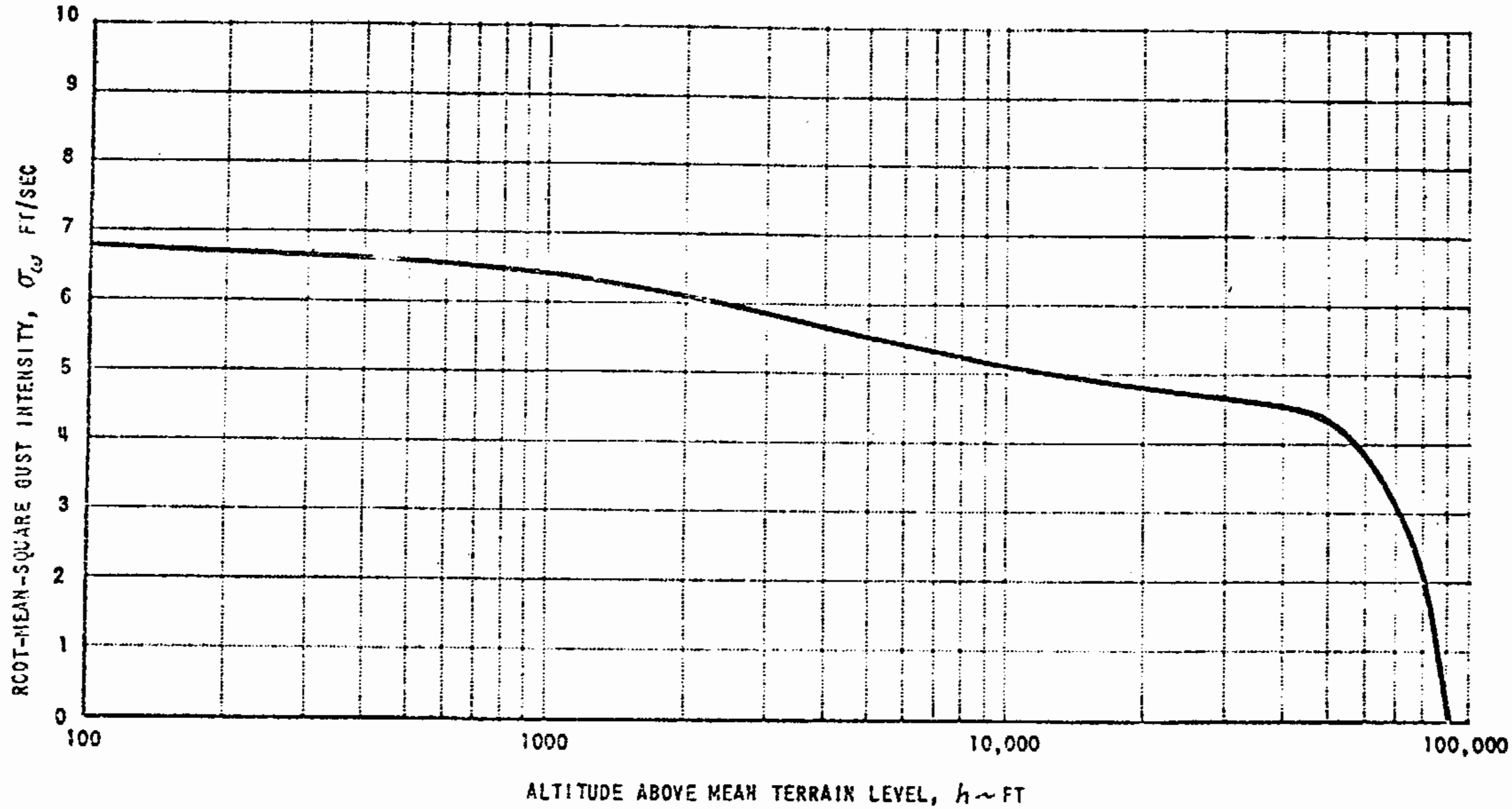


FIGURE 8. Intensity for Clear Air Turbulence

Requirement

Paragraph 3.7.4.1 Thunderstorm turbulence (von Karman scales). The scales for thunderstorm turbulence using the von Karman form are $L_u = L_v = L_w = 2500$ feet.

Paragraph 3.7.4.2 Thunderstorm turbulence (Dryden scales). The scales for thunderstorm turbulence using the Dryden form are $L_u = L_v = L_w = 1750$ feet.

Comparison

None

Resolution

None

Recommendation

Figure 8, Intensity for Clear Air Turbulence, might be replaced by analytic expressions. The present curve can be satisfactorily approximated using three straight-line segments or two straight-line segments and, for the higher altitude values, a parabolic segment.

Requirement

Paragraph 3.7.3 Scales and intensities (clear air turbulence). The root-mean-square intensity σ_w for clear air turbulence is defined on figure 8 as a function of altitude. The intensities σ_u and σ_v may be obtained using the relationships

$$\frac{\sigma_u^2}{L_u^{2/3}} = \frac{\sigma_v^2}{L_v^{2/3}} = \frac{\sigma_w^2}{L_w^{2/3}} \quad \text{(von Karman form)}$$

$$\frac{\sigma_u^2}{L_u} = \frac{\sigma_v^2}{L_v} = \frac{\sigma_w^2}{L_w} \quad \text{(Dryden form)}$$

The scales for clear air turbulence are defined in 3.7.3.1 and 3.7.3.2 as a function of altitude. The altitude shall be defined consistently with any applicable terrain models specified in the contract. For those Flight Phases involving climbs and descents, a single set of scales and intensities based on an average altitude may be used. If an average set of scales and intensities is used for Category C Flight Phases, it shall be based on an altitude of 500 feet.

Paragraph 3.7.3.1 Clear air turbulence (von Karman scales). The scales for clear air turbulence using the von Karman form are:

Above h = 2500 feet: $L_u = L_v = L_w = 2500$ feet
Below h = 2500 feet: $L_w = h$ feet
$L_u = L_v = 184\, h^{1/3}$ feet

Paragraph 3.7.3.2 Clear air turbulence (Dryden scales). The scales for clear air turbulence using the Dryden form are:

Above h = 1750 feet: $L_u = L_v = L_w = 2500$ feet
Below h = 1750 feet: $L_w = h$ feet
$L_u = L_v = 145 h^{1/3}$ feet

Paragraph 3.7.4 Scales and intensities (thunderstorm turbulence). The root-mean-square intensities σ_u, σ_v, and σ_w are all equal to 21 feet per second for thunderstorm turbulence. The scales for thunderstorm turbulence are defined in 3.7.4.1 and 3.7.4.2. These values are to be used when evaluating the airplane's controllability in severe turbulence, but need not be considered for altitudes above 40,000 feet.

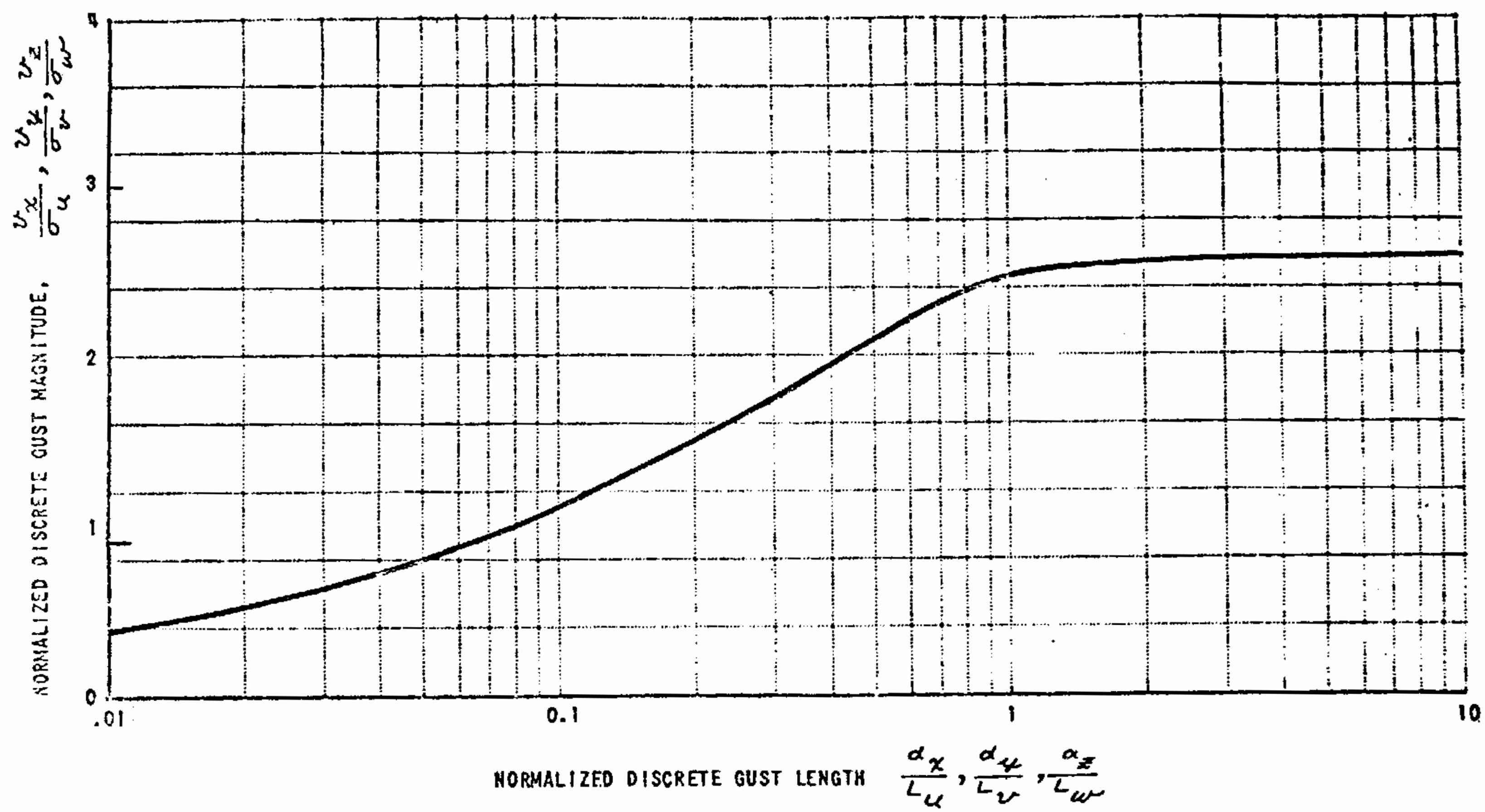


FIGURE 7. Magnitude of Discrete Gusts

Recommendation

Figure 7, Magnitude of Discrete Gusts, might be replaced by an analytic expression. For normalized discrete gust lengths greater than one, use a normalized discrete gust magnitude of 2.7; and for normalized discrete gust lengths equal to or less than one, use a logarithmic expression.

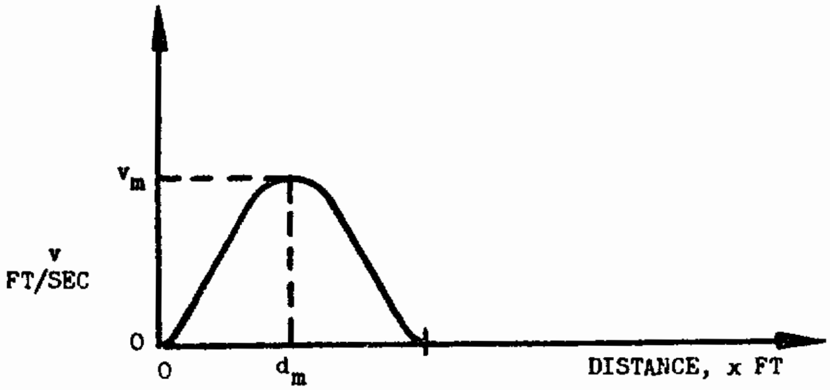
To be consistent with MIL-F-9490C (USAF) dated 13 March 1964, "Flight Control Systems", a velocity of 40 feet per second might be specified for analytical evaluation of the airplane's controllability and degradation of augmented vehicle damping ratios due to a discrete gust.

The discrete gust transient analysis should be extended to include a statistical dynamics analysis which would allow correlation with the continuous turbulence spectra results. By taking the Fourier integral of the present "1-cosine" shape, a simple expression is obtained which defines an equivalent continuous turbulence spectra filter. The subsequent analysis procedure is identical to that using the continuous random model spectra. A direct correlation of results is provided because the same form of statistical parameter is obtained.

Requirement

Paragraph 3.7.2.3 Discrete model. The discrete turbulence model may be used for any of the three gust-velocity components. The discrete gust has the "1 - cosine" shape:

$$
\begin{aligned}
v &= 0 &&, \; x < 0 \\
&= \frac{v_m}{2}\left(1 - \cos \frac{\pi x}{d_m}\right) &&, \; 0 \leq x \leq 2d_m \\
&= 0 &&, \; x > 2d_m
\end{aligned}
$$



Several values of d_m shall be used, each chosen so that the gust is tuned to each of the natural frequencies of the airplane and its flight control system (higher-frequency structural modes may be excepted). The magnitude v_m shall then be chosen from figure 7. The parameters L and σ to be used with figure 7 are the Dryden scales and intensities from 3.7.3 or 3.7.4 for the velocity component under consideration.

Comparison

None

Resolution

None

Requirement

Paragraph 3.7.2.1 Continuous random model (von Karman form). The von Karman form of the spectra for the turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \sigma_u^2 \frac{2L_u}{\pi} \frac{1}{[1 + (1.339 L_u \Omega)^2]^{5/6}}$$

$$\Phi_{v_g}(\Omega) = \sigma_v^2 \frac{L_v}{\pi} \frac{1 + \frac{8}{3}(1.339 L_v \Omega)^2}{[1 + (1.339 L_v \Omega)^2]^{11/6}}$$

$$\Phi_{w_g}(\Omega) = \sigma_w^2 \frac{L_w}{\pi} \frac{1 + \frac{8}{3}(1.339 L_w \Omega)^2}{[1 + (1.339 L_w \Omega)^2]^{11/6}}$$

Paragraph 3.7.2.2 Continuous random model (Dryden form). The Dryden form of the spectra for the turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \sigma_u^2 \frac{2L_u}{\pi} \frac{1}{1 + (L_u \Omega)^2}$$

$$\Phi_{v_g}(\Omega) = \sigma_v^2 \frac{L_v}{\pi} \frac{1 + 3(L_v \Omega)^2}{[1 + (L_v \Omega)^2]^2}$$

$$\Phi_{w_g}(\Omega) = \sigma_w^2 \frac{L_w}{\pi} \frac{1 + 3(L_w \Omega)^2}{[1 + (L_w \Omega)^2]^2}$$

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.7.2 Turbulence models. Where feasible, the von Karman form shall be used for the continuous random turbulence model, so that the flying qualities analyses will be consistent with the comparable structural analyses. When no comparable structural analysis is performed or when it is not feasible to use the von Karman form, use of the Dryden form will be permissible. In general, both the continuous random model and the discrete model shall be used. The scales and intensities used in determining the gust magnitudes for the discrete model shall be the same as those used in the Dryden continuous random model.

Comparison

None

Resolution

None

Recommendation

To obtain uniformity between analyses and moving base simulator results, a high and low frequency filter cutoff should be specified. A second order roll-off in both cases seems appropriate. The high frequency break might correspond to a wave length somewhere between ten centimeters to eight times the fuselage length or wing span. The low frequency break might correspond to a period of 20 seconds or a frequency of 0.3 radians per second.

Requirement

Paragraph 3.7 Atmospheric disturbances

Paragraph 3.7.1 Use of turbulence models. Paragraphs 3.7.2 through 3.7.5 specify a continuous random turbulence model and a discrete turbulence model that shall be used in analyses to determine compliance with those requirements of this Specification that refer to 3.7 explicitly, to assess:

a. The effect of turbulence on the flying qualities of the airplane;

b. The ability of a pilot to recover from the effects of discrete gusts.

Comparison

None

Resolution

None

Recommendation

It is recommended that the purpose of the analysis be defined, preceding the present words of the paragraph, because this is the initial subparagraph of a new section.

The following is a suggested wording.

"The purpose of this paragraph is to present and define a complete method of analysis for investigation of atmospheric disturbance effects on an airplane's flying and ride qualities. The subparagraphs present the model pilot, closed-loop description".

Requirement

Paragraph 3.6.5 Direct normal-force control. Use of devices for direct normal-force control shall not produce objectionable changes in attitude for any amount of control up to the maximum available. This requirement shall be met for Levels 1 and 2.

Comparison

No direct normal force control device is used on the F-5.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.4 Auxiliary dive recovery devices. Operation of any auxiliary device intended solely for dive recovery shall always produce a positive increment of normal acceleration, but the total normal load factor shall never exceed 0.8 n_L, controls free.

Comparison

No auxiliary device is used on the F-5 solely for dive recovery. However, the speed brakes may be used for speed control in dive recovery. Pitch transients compensation is provided by a mechanical interconnect between the speed brakes and the horizontal tail control system, providing a series input independent of the pilot.

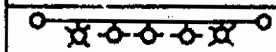
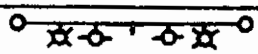
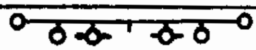
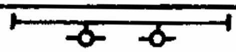
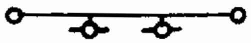
Resolution

None

Recommendation

None

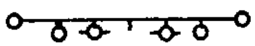
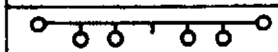
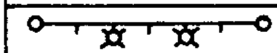
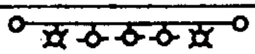
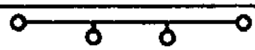
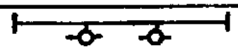
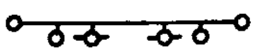
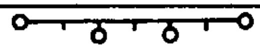
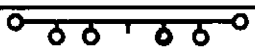
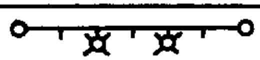
F-5 FLIGHT TEST DATA

CONFIGURATION	GROSS WT. (LBS.)	TEST C.G. (% MAC)	AFT C.G. LIMIT (% MAC)	INITIAL CONDITION					CONFIG. CHANGE	PARAMETER HELD CONSTANT	MAX. PUSH FORCE (lbs.)	MAX. PULL FORCE (lbs.)	Within Spec. Req.
				ALT. (1000 ft.)	V_c (knots)	GEAR POS.	FLAP POS.	Power Setting					
	17,940	16.3	21.0	10.0	232	Up	Dn	PFLF	Flaps Up	Rate of Climb	6.2	4.6	Yes
	12,620	23.9	26.0	9.8	195	Up	Dn	PFLF	Flaps Up	Rate of Climb	4.5	4.5	Yes
	15,520	21.4	21.0	2.3	244	Up	Dn	PFLF	Flaps Up	Rate of Climb	4.8	7.9	Yes
	12,310	22.7	22.0	5.1	245	Up	Dn	PFLF	Flaps Up	Rate of Climb	6.4	3.3	Yes
	12,890	19.9	21.0	2.3	246	Up	Dn	PFLF	Flaps Up	Rate of Climb	1.7	8.3	Yes
	11,130	19.9	22.0	35.0	303	Up	Up	MRP	MRP→Idle	Altitude	3.6	5.0	Yes
	11,060	20.0	22.0	35.3	306	Up	Up	MRP	MRP→Max	Altitude	3.8	7.8	Yes

PITCH TRIM CHANGES

TABLE 1b (3.6.3.1)

F-5 FLIGHT TEST DATA

CONFIGURATION	GROSS WT. (LBS.)	TEST C.G. (% MAC)	AFT C.G. LIMIT (% MAC)	INITIAL CONDITION ALT. (1000 ft.)	V_c (knots)	GEAR POS.	FLAP POS.	Power Setting	CONFIG. CHANGE	PARAMETER HELD CONSTANT	MAX. PUSH FORCE (lbs.)	MAX. PULL FORCE (lbs.)	Within Spec. Req.
	12,480	22.5	22.0	10.1	213	Up	Up	PFLF	Gear Down	Altitude	18.0	0.0	No
	12,420	22.7	22.0	10.0	213	Dn	Up	PFLF	Flaps Dn.	Altitude	8.9	14.5	No
	12,190	23.0	22.0	5.2	244	Up	Up	PFLF	Flaps Dn.	Altitude	7.5	3.9	Yes
	13,750	21.8	21.5	4.9	247	Dn	Up	PFLF	Flaps Dn.	Altitude	12.0	4.5	No
	13,060	23.6	23.0	5.0	242	Dn	Up	PFLF	Flaps Dn.	Altitude	8.3	13.9	No
	13,900	19.3	21.0	10.1	241	Dn	Up	PFLF	Flaps Dn.	Altitude	9.5	12.0	No
	13,080	22.7	22.5	5.1	243	Dn	Up	PFLF	Flaps Dn.	Altitude	5.8	0.0	Yes
	12,060	19.5	21.0	5.0	240	Dn	Up	PFLF	Flaps Dn.	Altitude	3.6	5.2	Yes
	13,170	21.9	22.0	10.0	152	Dn	Dn	PFLF	PFLF→Idle	Speed	2.0	0.0	Yes
	15,380	23.0	22.0	10.0	179	Dn	Dn	PFLF	PFLF→Max	Altitude	4.7	10.0	Yes
	14,230	22.6	22.0	10.0	189	Dn	Dn	Max	Gear Up	Rate of Climb	16.5	2.0	No
	13,670	22.9	22.0	9.9	191	Up	Dn	Max	Flaps Up	Rate of Climb	6.2	3.1	Yes
	14,640	21.2	22.5	4.6	240	Up	Dn	PFLF	Flaps Up	Rate of Climb	6.9	5.2	Yes
	15,560	21.8	21.5	2.4	240	Up	Dn	PFLF	Flaps Up	Rate of Climb	4.4	4.7	Yes
	15,160	23.0	23.0	2.2	234	Up	Dn	PFLF	Flaps Up	Rate of Climb	4.6	5.0	Yes

PITCH TRIM CHANGES

TABLE Ia (3.6.3.1)

TABLE XIV. Pitch Trim Change Conditions

	Flight Phase	Initial Trim Condition: Altitude	Speed	Landing Gear	High-lift Devices & Wing Flaps	Thrust	Configuration Change	Parameter to be held constant
1	Approach	$h_{o_{min}}$	Normal pattern entry speed	Up	Up	TLF	Gear down	Altitude and airspeed*
2				Up	Up	TLF	Gear down	Altitude
3				Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude and airspeed*
4				Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude
5				Down	Down	TLF	Idle thrust	Airspeed
6			$V_{o_{min}}$	Down	Down	TLF	Extend approach drag device	Airspeed
7				Down	Down	TLF	Takeoff thrust	Airspeed
8	Approach		$V_{o_{min}}$	Down	Down	TLF	Takeoff thrust plus normal clean-up for wave-off (go-around)	Airspeed
9	Takeoff			Down	Take-off	Take-off thrust	Gear up	Pitch attitude
10			Minimum flap-retract speed	Up	Take-off	Take-off thrust	Retract high-lift devices and wing flaps	Airspeed
11	Cruise and air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Idle thrust	Pitch attitude
12				Up	Up	MRT	Actuate deceleration device	
13				Up	Up	MRT	Maximum augmented thrust	
14			Speed for best range	Up	Up	TLF	Actuate deceleration device	

*Throttle setting may be changed during the maneuver

Notes:
- Auxiliary drag devices are initially retracted, and all details of configuration not specifically mentioned are normal for the Flight Phase.
- If power reduction is permitted in meeting the deceleration requirements established for the mission, actuation of the deceleration device in #12 and #14 shall be accompanied by the allowable power reduction.

Recommendation

It is recommended that an experimental study be conducted to determine if other parameters in conjunction with control forces might not be better utilized as controlling criteria for pitch trim change requirements. For example, "g" transients, airplane/control response, natural frequency, damping and attitude change contribute to the acceptability by pilots of airplane response to configuration changes. Moving-base simulation can be conducted to evaluate all these items with the pilot in the loop. The results will contribute quantitatively towards either validating completely the existing requirements or substantiating new requirements.

Requirement

Paragraph 3.6.3.1 Pitch trim changes. The pitch trim changes caused by operation of secondary control devices shall not be so large that a peak elevator control force in excess of 10 pounds for center-stick controllers or 20 pounds for wheel controllers is required when such configuration changes are made in flight under conditions representative of operational procedure. Generally, the conditions listed in table XIV will suffice for determination of compliance with this requirement. (For airplanes with variable-sweep wings, additional requirements will be imposed consistent with operational employment of the vehicle.) With the airplane trimmed for each specified initial condition, the peak force required to maintain the specified parameter constant following the specified configuration change shall not exceed the stated value for a time interval of at least 5 seconds following the completion of the pilot action initiating the configuration change. The magnitude and rate of trim change subsequent to this time period shall be such that the forces are easily trimmable by use of the normal trimming devices. These requirements define Level 1. For Levels 2 and 3, the allowable forces are increased by 50 percent.

Comparison

Sufficient flight test data are not available to completely validate this requirement. Reference 8 contains F-5 compliance flight test data applicable to MIL-F-8785 (2) dated 17 October 1955. Pitch trim change conditions of table XIV of MIL-F-8785B are somewhat different from those tested. Table 1 (3.6.3.1) presents data from Reference 8 which are partially applicable for comparison with the requirements of this paragraph. Disagreement in the form of noncompliance is exhibited for some conditions representative of those in table XIV. This is shown as control forces greater than the maximum 10 pounds allowed.

Resolution

Although noncompliance is shown for some conditions, pitch trim changes on the F-5 have been found to be quite acceptable by USAF and Northrop test pilots. Automatic trim devices are used to reduce pitch trim changes induced by flaps and speed brakes. Trim scheduling of these mechanical devices (interconnects between flaps, speed brakes and tail) was optimized during the development of the F-5 airplane. The F-5 pitch trim changes reach a maximum control force of 18 pounds, Table 1 (3.6.3.1). Nevertheless, this experience is not considered sufficiently substantiative to recommend 18 pounds for a new requirement.

Requirement

Paragraph 3.6.3 Transients and trim changes. The transients and steady-state trim changes for normal operation of secondary control devices (such as throttle, flaps, slats, speed brakes, deceleration devices, dive recovery devices, wing sweep, and landing gear) shall not impose excessive control forces to maintain the desired heading, altitude, attitude, rate of climb, speed or load factor without use of the trimmer control. This requirement applies to all in-flight configuration changes and combinations of changes made under service conditions, including the effects of asymmetric operations such as unequal operation of landing gear, speed brakes, slats, or flaps. In no case shall there be any objectionable buffeting or oscillation of such devices. More specific requirements on secondary control devices are contained in 3.6.3.1, 3.6.4, and 3.6.5 and in MIL-F-9490 and MIL-F-18372.

Comparison

The F-5 exhibits agreement. Transients and trim changes from throttles and landing gear are negligible. Transients and trim changes from flaps and speed brakes are automatically compensated for by mechanical inputs to the horizontal tail controls as a series input independent of pilot input.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.2 Speed and flight-path control devices. The effectiveness and response times of the fore-and-aft force controls, in combination with the other longitudinal controls, shall be sufficient to provide adequate control of flight path and airspeed at any flight condition within the Operational Flight Envelope. This requirement may be met by use of devices such as throttles, thrust reversers, auxiliary drag devices, and flaps.

Comparison

The F-5 utilizes speed brakes to control fore-and-aft forces. The speed brakes when used in combination with other longitudinal controls are in agreement with the requirements of this paragraph.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.1.4 Trim system irreversibility. All trimming devices shall maintain a given setting indefinitely, unless changed by the pilot, by a special automatic interconnect such as to the landing flaps, or by the operation of an augmentation device. If an automatic interconnect or augmentation device is used in conjunction with a trim device, provision shall be made to ensure the accurate return of the device to its initial trim position on completion of each interconnect or augmentation operation.

Comparison

The F-5 exhibits agreement. Aileron and pitch trim actuators are irreversible. Flap and speed brake interconnects to the horizontal tail controls are positive acting push-pull cables driven in each direction by two-way actuating cams. The yaw augmenter actuator used for rudder trim is irreversible considering closed-loop feedback. A positive centering spring returns the actuator to neutral trim position in case of augmenter failure or shut-off.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.1.3 Stalling of trim systems. Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. Specifically, the longitudinal trim system shall be capable of operating during the dive recoveries of 3.2.3.6 at any attainable permissible n, at any possible position of the trimming device.

Comparison

The F-5 exhibits agreement. Trim system operating forces are independent of aerodynamic loads since full-powered controls are used. The electromechanical trim actuators for aileron and pitch control systems, and the electro-hydraulic yaw augmenter actuator used for rudder trim are designed to operate against maximum system friction, feel spring, and servo valve forces as applicable.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.1.2 Rate of trim operation. Trim devices shall operate rapidly enough to enable the pilot to maintain low control forces under changing conditions normally encountered in service, yet not so rapidly as to cause over-sensitivity or trim precision difficulties under any conditions. Specifically, it shall be possible to trim the elevator control forces to less than ±10 pounds for center-stick airplanes and ±20 pounds for wheel-control airplanes throughout (a) dives and ground attack maneuvers required in normal service operation and (b) level-flight accelerations at maximum augmented thrust from 250 knots or $V_{R/C}$, whichever is less, to V_{max} at any altitude when the airplane is trimmed for level flight prior to initiation of the maneuver.

Comparison

The F-5 exhibits agreement. Available trim rates are:

Horizontal tail	0.25 degree/second at 0 degree to 2 degrees/second at 9 degrees; the trim range is 0 degree to 9 degrees
Aileron	0.75 degree/second
Rudder	50 degrees/second

Time history of a typical dive is depicted in Figure 1 (3.2.3.5). Time histories of several level-flight maximum accelerations are depicted in Figure 1 (3.2.1.1) to Figure 5 (3.2.1.1).

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6.1.1 Trim for asymmetric thrust. For all multiengine airplanes, it shall be possible to trim the elevator, rudder, and aileron control forces to zero in straight flight with up to two engines inoperative following asymmetric loss of thrust from the most critical factors (3.3.9). This requirement defines Level 1 in level-flight cruise at speeds from the maximum-range speed for the engine(s)-out configuration to the speed obtainable with normal rated thrust on the functioning engine(s). Systems completely dependent on the failed engines shall also be considered failed.

Comparison

The F-5 exhibits agreement. The engine thrust vector is close enough to the plane of symmetry that trim required for asymmetric thrust is negligible compared to other factors which determine the required trim range.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.6 Characteristics of secondary control systems.

Paragraph 3.6.1 Trim system. In straight flight, throughout the Operational Flight Envelope the trimming devices shall be capable of reducing the elevator, rudder, and aileron control forces to zero for Levels 1 and 2. For Level 3, the untrimmed cockpit control forces shall not exceed 10 pounds elevator, 5 pounds aileron, and 20 pounds rudder. The failures to be considered in applying the Level 2 and 3 requirements shall include trim sticking and runaway in either direction. It is permissible to meet the Level 2 and 3 requirements by providing the pilot with alternate trim mechanisms or override capability. Additional requirements on trim rate and authority are contained in MIL-F-9490 and MIL-F-18372.

Comparison

Trim system failure, particularly runaway trim, would degrade F-5 handling qualities to Level 3. No backup or override system is provided. Extreme forces required to hold controls at full trim position in one direction with runaway trim in the other direction are: aileron 8 pounds, rudder 19 pounds, and horizontal tail 48 pounds. The probability of F-5 trim failure (sticking or runaway) is less than 10^{-5} per flight per control.

Resolution

As shown in the comparison, there is partial disagreement between the requirements of this paragraph and the F-5 characteristics. The F-5 does not meet the Level 3 maximum force requirement for aileron and horizontal tail systems but meets the rudder force requirement. However, the high reliability demonstrated for the F-5 trim system, together with the low probability of encountering the need for full opposite trim, based on F-5 service life, substantiate that the F-5 trim system is acceptable. Nevertheless, the requirement of this paragraph is considered to be reasonable since it allows override capability. Special exception to full compliance should be on the basis of very low probability of failure as exhibited on the F-5.

Recommendation

None

Requirement

Paragraph 3.5.6.2 Trim changes. The control forces required to maintain attitude and zero sideslip for the situations described in 3.5.6 shall not exceed the following limits for at least 5 seconds following the transfer:

Elevator-------20 pounds
Aileron -------10 pounds
Rudder --------50 pounds

These requirements apply only for Airplane Normal States.

Comparison

The F-5 aircraft exhibits agreement with the requirement specified. The maximum control forces which are shown in the comparison portion of paragraph 3.5.5.2 are directly applicable to this paragraph.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.6.1 Transients. With controls free, the transients resulting from the situations described in 3.5.6 shall not exceed the following limits for at least 2 seconds following the transfer:

Within the Operational flight envelope	±0.05g normal or lateral acceleration at the pilot's station and ±1 degree per second roll.
Within the Service Flight Envelope	±0.5g at the pilot's station, ±5 degrees per second roll, and the lesser of ±5 degrees sideslip or the structural limit

These requirements apply only for Airplane Normal States.

Comparison

The F-5 aircraft installation and alignment tolerances for the augmentation system do not agree with the ±0.05g normal acceleration requirement.

Resolution

It was considered economically impractical, to align the pitch augmentation system to within ± 0.0125-degree of tail deflection which is required to meet the ±0.05g requirement. This is based on maximum horizontal tail surface effectiveness of 4g/degree. The F-5 pitch augmentation system is acceptably aligned to within ±0.06-degree of tail deflection.

Recommendation

It is recommended that the "±0.05g normal acceleration" be changed to "±0.25g normal acceleration."

Requirement

Paragraph 3.5.6 Transfer to alternate control modes. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be small and gradual enough that dangerous flying qualities never result.

Comparison

The F-5 aircraft shows agreement with the requirements except for one condition. That condition exists when a hardover type failure is present and an attempt is made to engage the system. In this case, unacceptable flying qualities may result.

Resolution

The requirement is lacking because it does not have specific requirements in a failed condition or after a failure. The F-5 exhibits unacceptable flying qualities if engagement of the augmentation system is attempted after a hardover failure.

Recommendation

Add the following to the end of the paragraph:

"Engagement of a failed system must not be possible if dangerous flying qualities will result."

Requirement

Paragraph 3.5.5.2 Trim changes due to failures. The control forces required to maintain attitude and zero sideslip for the failures described in 3.5.5 shall not exceed the following limits for at least 5 seconds following the failure:

Elevator-------20 pounds
Aileron -------10 pounds
Rudder --------50 pounds

Comparison

The F-5 aircraft shows agreement with the requirements specified. A maximum of 19 pounds is required to compensate for 1.5 degrees hardover limit of the horizontal tail with full forward trim. Maximum rudder force of 32 pounds is required for 10 degrees rudder correction, assuming 4 degrees maximum rudder trim to one side and hardover failure of 6 degrees to the opposite side. The aileron system is not affected.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.5.1 Failure transients. With controls free, the airplane motions due to failures described in 3.5.5 shall not exceed the following limits for at least 2 seconds following the failure, as a function of the Level of flying qualities after the failure transient has subsided:

Level 1 (after failure)	±0.05g normal or lateral acceleration at the pilot's station and 1 degree per second in roll
Level 2 (after failure)	±0.5g at the pilot's station, ±5 degrees per second roll, and the lesser of ±5 degrees sideslip or the structural limits
Level 3 (after failure)	No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

Comparison

The F-5 need not comply with Level 1 (after failure) because the probability of failure is less than 10^{-2} per flight.

Agreement exists with Level 2 (after failure) requirements except for conditions resulting from hardover failures of the servo actuators. Agreement with Level 3 (after failure) requirements exists for these conditions. Compliance in a hardover failure is compared to Level 3 (after failure) requirements based on the low probability of encountering this failure which is 1.52×10^{-5} per flight.

The F-5 can be safely flown without the benefit of augmentation. Consequently, whenever a failure occurs the augmentation system can be easily disengaged by the pilot. Nevertheless, the requirement of this paragraph is considered adequate.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.5 Failures. If the flying qualities with any or all of the augmentation devices inoperative are dangerous or intolerable, special provisions shall be incorporated to preclude a critical single failure. Failure-induced transient motions and trim changes resulting either immediately after failure or upon subsequent transfer to alternate control modes shall be small and gradual enough that dangerous flying qualities never result.

Comparison

The F-5 aircraft exhibits agreement with the requirements specified since the aircraft can be safely flown without the use of the stability augmentation system.

Provisions have been made to protect against certain single failures which may cause undesirable transients or trim changes. The feedback transducer and servo valve, which form a part of the servo actuator assembly, are both protected against an open circuit. Short circuit protection has been provided for the feedback transducer. The pilot can also disengage either axis independently by manually overriding the magnetically held engage switches. An emergency disconnect is also provided on the stick for disabling the pitch axis augmentation.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.4.2 Saturation of augmentation systems. Limits on the authority of augmentation systems or saturation of equipment shall not result in objectionable flying qualities. In particular, this requirement shall be met during rapid large-amplitude maneuvers, during operation near V_S, and during flight in the atmospheric disturbances of 3.7.3 and 3.7.4.

Comparison

The F-5 aircraft exhibits agreement with the requirements specified although some momentary degradation of damping effectiveness may be encountered. This results from the use of limited authority servo actuators.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.4.1 Performance of augmentation systems. Performance degradation of augmentation systems caused by the atmospheric disturbances of 3.7.3 and 3.7.4 and by structural vibrations shall be considered, when such systems are used.

Comparison

There is no perceptible indication of performance degradation of the stability augmentation system, thus agreeing with the requirements specified. However, there may be certain combinations of yaw trim and augmenter commands at low speed or in extremely high turbulence which will require rudder deflections in excess of the augmentation authority. No performance degradation is attributable to structural vibrations.

Resolution

Due to limitation of augmentation control authority, some reduction in longitudinal short period damping ratios can be expected in atmospheric disturbances. This reduction in damping is not considered as being objectionable because of the inherently good short period damping characteristics of the aircraft with the augmentation inoperative. No objectionable pilot comments were reported. Consequently, the requirement is considered valid.

Recommendation

None

Requirement

Paragraph 3.5.4 Augmentation systems. Normal operation of stability augmentation and control augmentation systems and devices shall not introduce any objectionable flight or ground handling characteristics.

Comparison

The stability augmentation system (SAS) shows agreement with the requirements specified.

There have been comments regarding motion being felt at the rudder pedals with the SAS operating. This condition is generally attributable to the effect of the centering spring located internal to the SAS servo-actuator. The centering spring acts to maintain the servo-actuator at neutral (double-acting-hydraulic-unit) position when the SAS is not generating any signal. At this neutral point, under certain conditions there is a hesitancy for the servo-actuator to drive smoothly. Subsequently, this is felt in the rudder pedals when the feet are firmly on the pedals.resulting in pilot in frequent objections, however, not warranting any design corrective action.

This requirement is considered reasonable.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.3.2 Damping. All control system oscillations shall be well damped, unless they are of such an amplitude, frequency, and phasing that they do not result in objectionable oscillations of the cockpit controls or the airframe during abrupt maneuvers and during flight in the atmospheric disturbances specified in 3.7.3 and 3.7.4.

Comparison

The rudder and aileron control systems are both representative of a critically damped second order system and demonstrate agreement with the requirements specified. All control systems oscillations are well damped and do not result in objectionable oscillations of the cockpit controls or airframe under the conditions specified. Although the mechanical control system for the longitudinal axis is a lightly damped ($0.08 < \zeta < 0.2$) system exhibiting second-order response characteristics, the frequency ($15 < \omega_n < 24$ rad/sec) and built-in breakout force are such that no adverse pilot comments were reported during the flight test program.

Resolution

None

Recommendation

None

Test data showed the control system response, for the rudder and aileron, to be described by a critically damped second order transfer function. The corner frequencies are at 6.4 Hertz and 6 Hertz respectively. The maximum ω_{nd} for the rudder system was at 1.74 Hertz. The phase lag at this ω_{nd} is 25 degrees. The phase lag for all other conditions was less than 25 degrees which indicates agreement with the requirements specified.

The aileron system indicated agreement for all but 2 of 18 cases evaluated. The two marginal cases showed a phase lag of 32 degrees for Category A, Level 1, flight phase. This is 2 degrees greater than the specification allows.

Resolution

Partial disagreement between the F-5 characteristics and this paragraph was exhibited as shown by the 32 degrees phase lag instead of 30 degrees maximum allowable. The difference is small and infrequent. Consequently, the requirement is considered valid.

Recommendation

None

Requirement

Paragraph 3.5.3.1 Control feel. In flight, the cockpit-control deflection shall not lead the cockpit-control force for any frequency or force amplitude. This requirement applies to the elevator, aileron, and rudder controls. In flight, the cockpit-control deflection shall not lag the cockpit-control force by more than the angles listed in 3.5.3, for frequencies equal to or less than those listed in 3.5.3, for reasonably large force inputs. The lags for very small control force amplitudes shall not interfere with the pilot's ability to perform precision tasks required in normal operation.

Comparison

The F-5 aircraft shows agreement with the requirements specified. Pilot evaluation and acceptance confirms the absence of interference with the pilot in performing precision tasks required in normal operation. The control feel is similar to the feel of a hydraulic actuator which is described by a first order lag transfer function. This tends to preclude the introduction of a lead from the cockpit-control deflection, although the bobweight has the opposite tendency.

Pitch-axis flight tests were performed using sinusoidal control inputs. Input frequencies were varied between 0.2 and 1.4 Hertz to determine aircraft response to stabilizer inputs. The maximum frequency was found to be at 1.45 Hertz. The mechanical control system for the longitudinal axis was also subjected to frequency response tests. Data thus obtained were compared and used to determine phase lag.

Calculations were used to determine the change in the mechanical control system natural frequency due to the variations in surface trim positions.

The phase lag for the maximum short period oscillation ($\omega_{n_{sp}}$ = 1.45 Hertz) was 22 degrees, exhibiting agreement with the requirement of paragraph 3.5.3. The phase lag of 22 degrees was obtained from the summation of two Bode plots which represented the hydraulic actuator and mechanical control system. The actuator is described as a first-order lag with a corner frequency at 5.5 Hertz. The mechanical control system for this condition is described by a second-order response with a damping ratio of 0.15 and a ω_n of 3.7 Hertz. Contribution from the hydraulic actuator was 16 degrees of lag, and an additional 6 degrees of lag were introduced by the mechanical control system.

Comparisons were made at intermediate values as well as at extremes of horizontal surface trim. At the -9.1 degree trim position, damping ratio was 0.15 with a natural frequency of 2.45 Hertz. Under this condition, short period oscillations up to 1.4 Hertz would meet the 30 degrees phase lag requirement.

Requirement

Paragraph 3.5.3 Dynamic characteristics. The response of the control surfaces in flight shall not lag the cockpit control force inputs by more than the angles shown in table XIII, for frequencies equal to or less than the frequencies shown in table XIII.

TABLE XIII. Allowable Control Surface Lags

Level	Allowable Lag ~ deg		Control	Upper Frequency ~ rad/sec
	Category A and C Flight Phases	Category B Flight Phases	elevator	$\omega_{n_{sp}}$
1 and 2	30	45	rudder & aileron	ω_{n_d} or $1/\tau_R$ (whichever is larger)
3	60			

The lags referred to are the phase angles obtained from steady-state frequency responses, for reasonably large-amplitude force inputs. The lags for very small control-force amplitudes shall be small enough that they do not interfere with the pilot's ability to perform any precision tasks required in normal operation.

Comparison

The F-5 aircraft shows agreement with the requirements except where noted in the succeeding paragraphs. For test data analysis purposes, the rudder and aileron control systems were assumed to be described by a second-order transfer function. The corner frequencies of the Bode plot are at 3.5 and 6 Hertz respectively, with critical damping for both. The longitudinal control system is described by the following transfer function:

$$\frac{\delta_h}{F_S} = \frac{K \text{ deg/lb}}{(TS+1)\left(\frac{S^2}{\omega^2} + \frac{2\zeta}{\omega} S + 1\right)}$$

where K is the average force gradient about a trim point and within a ± force range, ω is the mechanical system natural frequency (ω < 24 rad/sec), ζ is the damping ratio ($0.08 < \zeta < 0.2$), δ_h is the tail position in degrees, F_S is the stick force in pounds.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.2.4 Adjustable controls. When a cockpit control is adjustable for pilot physical dimensions or comfort, the control forces defined in 6.2 refer to the mean adjustment. A force referred to any other adjustment shall not differ by more than 10 percent from the force referred to the mean adjustment.

Comparison

The F-5 agrees with this requirement. The only controls with adjustable positions are the rudder pedals. Rudder and brake operating forces are essentially unaffected by pedal adjustment.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.2.3 Rate of control displacement. The ability of the airplane to perform the operational maneuvers required of it shall not be limited in the atmospheric disturbances specified in 3.7 by control surface deflection rates. For powered or boosted controls, the effect of engine speed and the duty cycle of both primary and secondary controls together with the pilot control techniques shall be included when establishing compliance with this requirement.

Comparison

The F-5 is in agreement with this paragraph. The no-load control surface deflection rates are:

aileron	120 degrees per second
rudder	50 degrees per second
horizontal tail	26 degrees per second

The no-load rates are independent of whether one or both hydraulic systems are operating. Flight tests and service operations have revealed that the hydraulic supply system is sufficient for adequate surface control at engine speeds down to windmilling speed.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.5.2.2 Cockpit control free play. The free play in each cockpit control, that is, any motion of the cockpit control which does not move the control surface in flight, shall not result in objectionable flight characteristics, particularly for small-amplitude control inputs.

Comparison

The F-5 is in agreement with the requirements of this paragraph.

Resolution

None

Recommendation

None

Yawing moment induced by ailerons provides coordinated turns without use of rudder. Rudder inputs by the pilot are only required for precision tracking or cross-wind landing. F-5 Category II tests were conducted with feel spring misrigged to 30 pounds breakout force without pilot complaint. Consequently, the 7-pound requirement could be raised considerably and be acceptable.

Recommendation

Increase requirement for rudder breakout force for Class IV aircraft to 14 pounds.

Requirements

Paragraph 3.5.2.1 Control centering and breakout forces. Longitudinal, lateral, and directional controls should exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the combined effects of centering, breakout force, stability, and force gradient shall not produce objectionable flight characteristics, such as poor precision-tracking ability, or permit large departures from trim conditions with controls free. Breakout forces, including friction, preload, etc., shall be within the limits of table XII. The values in table XII refer to the cockpit control force required to start movement of the control surface in flight for Levels 1 and 2; the upper limits are doubled for Level 3.

TABLE XII. Allowable Breakout Forces, Pounds

Control		Classes I, II-C, IV		Classes II-L, III	
		min	max	min	max
Elevator	Stick	1/2	3	1/2	5
	Wheel	1/2	4	1/2	7
Aileron	Stick	1/2	2	1/2	4
	Wheel	1/2	3	1/2	6
Rudder		1	7	1	14

Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided that qualitative agreement between ground measurement and flight observation can be established.

Comparison

The average breakout forces for the F-5 are:

horizontal tail	-	2 pounds
aileron	-	1 pound
rudder	-	13 pounds

Resolution

Rudder breakout force, although well above the required 7 pounds, has received no pilot complaints. The higher breakout force is required as a backup for the series-mounted yaw stability augmenter actuator. The breakout force consists of 2.5 pounds friction, 8.5 pounds feel spring force, and 2 pounds nose wheel steering bungee force (steering inoperative).

Requirement

Paragraph 3.5 Characteristics of the primary flight control system

Paragraph 3.5.1 General characteristics. As used in this specification, the term primary flight control system includes the elevator, aileron and rudder controls, stability augmentation systems, and all mechanisms and devices that they operate. The requirements of this section are concerned with those aspects of the primary flight control system which are directly related to flying qualities. These requirements are in addition to the requirements of the applicable control system design specification, e.g., MIL-F-9490 or MIL-C-18244.

Paragraph 3.5.2 Mechanical characteristics. Some of the important mechanical characteristics of control systems (including servo valves and actuators) are: friction and preload, lost motion, flexibility, mass imbalance and inertia, nonlinear gearing, and rate limiting. Requirements for these characteristics are contained in 3.5.2.1 through 3.5.2.4. Meeting these separate requirements, however, will not necessarily ensure that the overall system will be satisfactory; the mechanical characteristics must be compatible with the non-mechanical portions of the control system and with the airframe dynamic characteristics.

Comparison

None

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.10 Failures. No single failure of any component or system shall result in dangerous or intolerable flying qualities; Special Failure States (3.1.6.2.1) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

Comparison

All the failures cited in the comparison part of Paragraph 3.4.9 are candidates for Special Failure States due to their very infrequent occurrences. The requirements of Paragraph 3.1.6.2.1 will prevail for these cases.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.9 Transients following failures. The airplane motions following sudden airplane system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. This time delay should include an interval between the occurrence of the failure and the occurrence of a cue such as acceleration, rate, displacement, or sound that will definitely indicate to the pilot that a failure has occurred, plus an additional interval which represents the time required for the pilot to diagnose the situation and initiate corrective action.

Comparison

The only known sudden airplane system or component failure that has been experienced by F-5 and T-38 airplanes that is not due to improper maintenance is hardover failure of the stability augmenter system.

The reported occurrences have been very infrequent. The probability of occurrence is approximately 9.9×10^{-6} per flight. In such cases, pilot corrective action is possible. The pilot can disengage the pitch augmenter by either a switch on the control stick or a switch on the console. The yaw augmenter can be disengaged by a switch on the left hand console.

The F-5 and T-38 can be safely flown and landed following failure and shut-off of the augmentation systems.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.8 Effects of armament delivery and special equipment. Operation of moveable parts such as bomb bay doors, cargo doors, armament pods, refueling devices, and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any pertinent flight condition. These requirements shall be met for Levels 1 and 2.

Comparison

The F-5 carries two cannons internally in the nose and various loadings of bombs externally on fuselage and wing pylons. The firing of the cannons as well as the release of bombs will not impair the tactical effectiveness of the airplane. The discussion presented in Paragraph 3.4.7 is applicable to this paragraph.

Resolution

None

Recommendation

None

Requirements

Paragraph 3.4.7 Release of stores. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is structurally permissible.

Comparison

The F-5 is qualified to carry and release a great variety of external stores, consisting of bombs, fuel tanks, rockets and pods. To determine the flight conditions where these stores can be cleared for release, two specific tasks are conducted.

The first task deals with obtaining satisfactory and safe separation from the airplane. The second task deals with establishing that no objectionable flight characteristics will result following store release. Analytical studies are conducted to initiate these tasks. Then, wind tunnel and flight tests take place to demonstrate qualification of store releases and evaluate airplane responses and flight characteristics.

For asymmetric store releases, minimum speeds are established based on sufficient aileron control provided, first, to counter roll transient responses and second, to hold and maintain wings level. At least half of the total aileron control remains available to maneuver with when the airplane is asymmetrically configured.

The F-5 characteristics are in agreement with the requirements of this paragraph.

Resolution

None

Recommendation

None

Requirement

Paragraph 3.4.6 Buffet. Within the boundaries of the Operational Flight Envelope, there shall be no objectionable buffet which might detract from the effectiveness of the airplane in executing its intended missions.

Comparison

Within the boundaries of its Operational Flight Envelope, the F-5 exhibits no objectionable buffet. The effectiveness of the airplane in executing its intended missions is not detracted. The discussion of buffet presented in Paragraph 3.4.2.2 and its subparagraphs applies equally to this paragraph.

Resolution

None

Recommendation

None

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13. ABSTRACT

This study was conducted to validate Military Specification MIL-F-8785B(ASG), "Flying Qualities of Piloted Airplanes," dated 7 August 1969 by performing a detailed comparison of its requirements with the known characteristics of the Northrop F-5 fighter and pilot comments on them.

The comparison was based primarily on existing flight test data supplemented by analytical data as required for this evaluation process. Paragraph by paragraph, validations or discrepancies are noted, resolution attempted if necessary, and any recommendations given.

In addition, recommendations are made enumerating experimental and analytical investigations beyond the scope of this study which will provide data for further validation and updating of the requirements.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Military Flying Qualities Specification						
Flying Qualities						
Flying Qualities Requirements						
Model F-5						
Model T-38						

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