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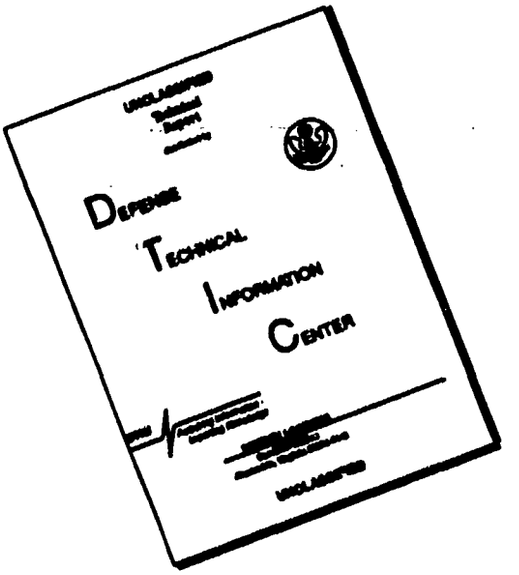

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PHASE II
FLIGHT EVALUATION

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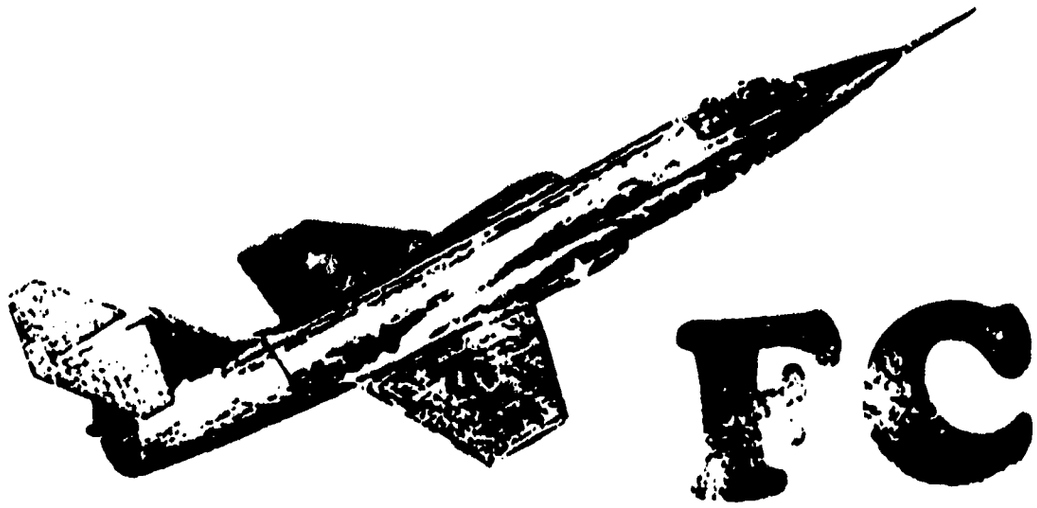

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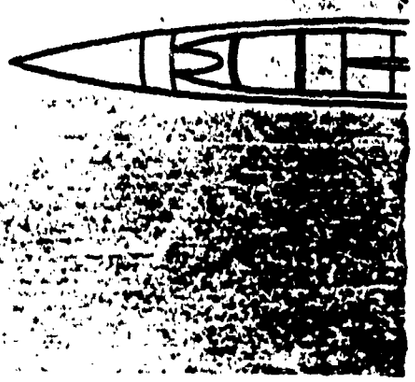
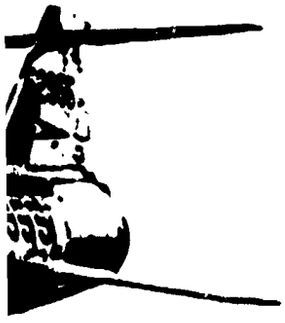


F-104A PHASE II EVALUATION



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AFFTC-TR-56-31 ■ December 1956

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– FMS STUDIOS, PASADENA, CALIFORNIA, 21 DECEMBER 1956

56AA 63938

ABSTRACT

The performance of the F-104A is superior to the performance of other Century Series Fighters currently in production.

Low internal fuel capacity severely limits the combat radius of action. This factor, together with the large variations of performance with temperature change, the high speed and high altitude capabilities, all combine to complicate optimum utilization of the aircraft. Successful integration of the F-104A into the SAGE system is essential if its maximum potential as an interceptor is to be realized.

The fine handling characteristics in the normal flight region are offset by an uncontrollable pitch-up at high angles of attack which increases in severity during accelerated maneuvers. Spin recovery has not been demonstrated. Characteristics preceding pitch-up are similar to those experienced in the F-101, but result in more violent maneuvers. It is expected that little or no natural warning will occur at supersonic speeds, and that the high load factors encountered in a supersonic pitch-up may lead to loss of both aircraft and pilot.

***This report has been reviewed and approved. /7* DECEMBER 1956**

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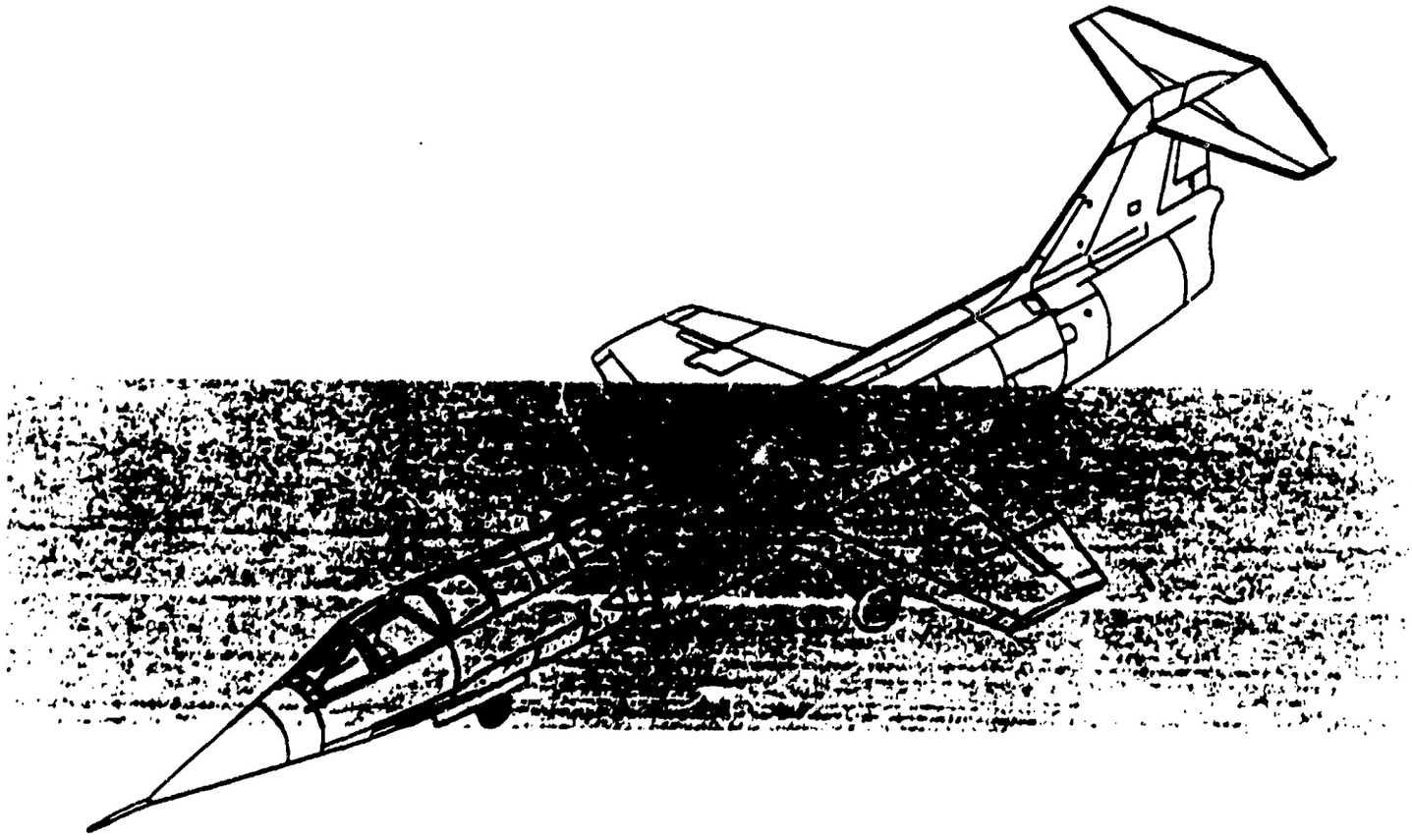
*NOTE: Due to the limited requirement for the information contained in this Appendix, it has been published in a separate volume. Copies of this Appendix can be obtained by writing to the Armed Services Technical Information Agency (see inside cover).

INTRODUCTION

This report presents the results of the Phase II flight test of the F-104A, S/N 55-2955. A human factors evaluation is presented in Appendix III. The program was conducted at Edwards Air Force Base, California, from 27 July 1956 to 23 August 1956. Flying time amounting to 18 hours was obtained during 29 flights.



The F-104A is a single place, light weight, air superiority fighter powered by one General Electric J79-GE-3 axial flow turbojet engine with afterburner. The outstanding external features of the aircraft are the extremely short wing span, the long, needle nosed fuselage and a high horizontal stabilizer. In addition to the normal wing flaps mounted in the trailing edge of the wing, a three position flap has been incorporated in the leading edge of the wing. A blowing type boundary layer control system is incorporated which begins operation after the trailing edge flaps have been deflected past the take-off position. Speed brakes are mounted on either side of the aft fuselage. Air intakes are "D" shaped with very sharp leading edges, and are cheek-mounted on the fuselage. The cockpit has an unconventional side opening canopy. Emergency escape is provided by downward seat ejection through a hatch just forward of the nose gear. A drag chute is housed in the bottom of the fuselage near the aft end of the aircraft.

The engine was designed to have a high thrust/weight ratio and to deliver 15,600 pounds thrust augmented and 10,000 pounds thrust unaugmented at standard sea level static conditions. The compressor is a 17 stage, 12 to 1 pressure ratio, single rotor, axial flow type with the first six stator stages variable. The turbine rotor has three stages and is designed for a turbine inlet temperature of 1700 degrees Fahrenheit.

The test aircraft was equipped with a "Phase O" engine. This engine will be supplanted by a "Phase I" engine in the Phase IV airplane. It is anticipated that a preliminary report will be distributed during January 1957 comparing performance of the F-104A with the two engines.

Control is provided by conventional ailerons, an all-moving stabilizer and flap type rudder. Pitch, roll, and yaw dampers are provided. The horizontal tail is actuated through a fully irreversible system. Hydraulic pressure is supplied by the engine to two hydraulic cylinders. An emergency wind driven turbine is installed to provide hydraulic pressure in the event of engine failure. Artificial feel is produced by a preloaded spring and a bobweight. The directional control system has a plain flap-type rudder which is actuated by a cable from the rudder pedals. Rudder boost is not provided. A preloaded centering spring provides artificial feel and a spring loaded centering lock holds the rudder at zero deflection against airloads when there is no force on the rudder pedals. The lateral control system consists of flap type outboard ailerons which are actuated by hydraulic cylinders, and is fully irreversible.

No external stores were available at the time these tests were made; consequently, all data in this report is representative of the clean airplane only.

The exhaust nozzle is a variable area, converging-diverging aerodynamic type. The nozzle assembly consists of a primary and secondary nozzle, each equipped with movable fingers. The secondary nozzle shroud moves aft as the fingers open.

TEST RESULTS: PERFORMANCE

cockpit evaluation

The cockpit arrangement is generally satisfactory. With the exception of the flip-up standby compass, all controls and switches are easily accessible to the pilot even with the shoulder harness locked. Pilots were favorably impressed with the logical arrangement of the instrument panel and the efficient presentation of the control elements, which are well forward on the console. The dual rear view mirror, similar to that in the F-101, is quite satisfactory. It provides better rear visibility than the single centered mirror in the XF-104. The following items are unsatisfactory:

1. The circuit breakers reveal a white color at the base when "popped" that is similar to the aluminum background. This makes it necessary to identify a "popped" breaker from its height compared to surrounding breakers.

2. The drag chute "T" handle is not easily accessible and is similar to the emergency landing gear "T" handle, which is adjacent but rotated 90 degrees.

3. The master caution light is red but should be amber. Also, it is not bright enough to attract attention in direct sunlight, or even when shaded. Other lights on the master warning panel are not satisfactorily bright in direct sunlight.

4. The 2-inch accelerometer is unsatisfactory for reasons discussed under "Intercept Mission".

5. The M-1 airspeed-Mach indicator is unsatisfactory for the usual reasons of poor readability and congested scale.

6. It is awkward for the pilot to raise the canopy to the ventilate position when wearing a pressure suit. This condition does not exist when wearing a flying suit, and is caused only by the restrictive nature of the pressure suit.

7. The compressor inlet temperature gage is displayed on the lower section of the instrument panel in F-104A's other than the one tested. It was located next to the airspeed-Mach indicator in the test aircraft. This is a desirable location since such a critical indicator belongs in the group of flight instruments. A detailed evaluation of the cockpit is presented as Appendix III of this report.

starting, taxiing, and ground handling

Starting procedure is similar to that used in other Century Series Fighters. A start switch must be actuated to allow starting air to turn the engine, and the throttle is opened to idle at 13 percent rpm.

Taxiing characteristics of the F-104A are satisfactory, and visibility is excellent. Power must be advanced to 82 percent rpm to initiate taxi roll to avoid operation in the restricted range of 67 to 82 percent rpm. The throttle is then moved to idle, which provides adequate thrust for taxi. Nose wheel steering, while not as precise as that in the F-100, is satisfactory, and secondary directional control with braking is adequate. The requirement that the cockpit air conditioning system be set to "RAM AIR" during ground operations is unsatisfactory. It is necessary, however, to prevent hot air from the heat exchanger from being exhausted into the aft fuselage section with consequent overheating.

take-off and acceleration to climb schedule

Excessive brake pedal force is required to hold the aircraft with military power, and the brakes must be "pumped" several times to increase their effectiveness before the throttle is advanced to this setting. The afterburner is lit after initiation of ground roll. Directional control is maintained with nose wheel steering until the rudder becomes effective at 60 knots IAS. The aircraft is rotated at 150 knots IAS, and take-off is made at 170 knots with either military or maximum power. The present tire appears to be critical during relatively long take-off rolls. The tread peeled off one tire during a mission when a military power take-off was made in relatively cool air.

The gear is retracted as soon as the aircraft becomes airborne. Retraction time is fast, and the rapid acceleration after take-off does not impair gear retraction.

The take-off flaps are not retracted before 250 knots IAS to prevent inducing stall warning buffet. The nose up trim change which occurs during flap retraction is not objectionable.

Take-off performance is shown graphically in Figure 1, and data at optimum speeds is tabulated below:

TAKE-OFF PERFORMANCE

	True Speed at Take-off kts.	IAS at Take-off kts.	True Speed at 50 Feet kts.	Ground Roll ft.	Total Distance Over 50 Foot Obstacle--ft
MAX. PWR.	170	170	200	2200	3350
MIL. PWR.	170	170	200	4600	7150

■ climb

Acceleration after take-off with maximum power is very rapid, and care must be taken not to overshoot the recommended climb schedule of .925 Mach number. Satisfactory transition to a climbing attitude was made by holding acceleration at 1.5g above Mach .75 until the desired flight path was intercepted. Normal acceleration is then reduced to .8g to remain on climb schedule. Optimum climb technique at maximum power calls for a subsonic climb to 35,000 feet, acceleration to Mach 2.0 at this altitude and supersonic climb to the desired altitude. Although higher rates of climb may be obtained at 1.6 to 1.7 Mach number, the supersonic climb should be made at 2.0 Mach number since the higher total energy developed at that speed increases maneuvering capabilities and permits higher altitudes to be reached during zoom climbs.

Zoom capability is outstanding. (Reference Figure 16.) An altitude of 70,000 feet may be reached during a zoom from near absolute ceiling even though the afterburner "blows out" at about 65,000 feet.

Climb performance with military power is quite modest and results in a combat ceiling of 40,000 feet. While military power may be used to extend range, the afterburner should be lit before reaching 29,000 feet, since consistent lights can not be obtained above that altitude. Service ceiling with maximum power is approximately 55,000 feet at 1.9 Mach number. Climb performance is shown in Figures 2 and 3 and is summarized in the following table for a gross weight of 18,500 pounds at engine start:

MILITARY CLIMB PERFORMANCE

ALTITUDE FT.	RATE OF CLIMB FT MIN.	TIME TO CLIMB MIN.	FUEL FLOW LB HR.	MACH NUMBER
SEA LEVEL	12,100	0	10,400	.820
10,000	10,500	.9	7,900	.850
20,000	7,900	1.95	5,900	.880
30,000	4,400	3.6	4,200	.915
40,000	500	—	2,800	.925

MAXIMUM CLIMB PERFORMANCE

ALTITUDE FT.	RATE OF CLIMB FT MIN.	TIME TO CLIMB MIN.	FUEL FLOW LB HR.	MACH NUMBER
SEA LEVEL	41,300	0	41,600	.925
10,000	35,800	.30	31,000	.925
20,000	28,700	.60	22,600	.925
30,000	20,200	1.05	16,200	.925
40,000	15,700	4.70*	22,000	1.9
50,000	7,500	7.55*	14,500	1.9
55,000	700	8.75*	11,200	1.9

*Includes time to accelerate to supersonic climb speed at 35,000 feet.

■ level flight

Maximum range is obtained near .9 Mach at about 35,000 feet. Power required data at 35,000 feet indicates a recommended cruise speed of .89 Mach number for a weight of 16,400 pounds. A speed of .98 Mach number was obtained with military power at that altitude and weight. Initial acceleration from cruise speed is comparatively low and reaches a minimum at 1.1 Mach number. At speeds above 1.1 Mach number the acceleration increases and a maximum rate is obtained at 1.6 to 1.7 Mach number. At higher speeds acceleration becomes lower although it is still quite rapid at the limiting speed of Mach 2.0. (Limit imposed by compressor inlet temperature of 250 degrees Fahrenheit.)

Accelerations made at 35,000, 40,000 and 45,000 feet are presented in Figures 10, 11 and 12. A summary of these plots is tabulated for an acceleration start weight of 16,700 pounds.

ACCELERATION PERFORMANCE

	MACH NUMBER	TIME MIN.	FUEL USED LB.	DISTANCE N.M.
35,000 Feet				
	1.05	0	0	0
	1.2	.55	150	6
	1.4	1:15	330	13
	1.6	1.65	520	20.5
	1.8	2.20	740	29
	2.0	2.95	1120	43
40,000 Feet				
	1.05	0	0	0
	1.2	.95	220	10.5
	1.4	1.95	460	22.5
	1.6	2.6	650	32
	1.8	3.2	860	42
	2.0	4.05	1180	57
45,000 Feet				
	1.05	0	0	0
	1.2	1.7	290	18
	1.4	3.5	650	40
	1.6	4.6	910	56
	1.8	5.5	1170	71.5
	2.0	6.6	1500	91

It should be pointed out that operation at low supersonic speeds (below 1.3 Mach) in afterburner becomes very costly in terms of fuel consumed, as does acceleration at less than maximum power.

It becomes virtually impossible to fly the airplane at constant speed and altitude at supersonic speeds since the drag and thrust required curves are nearly parallel. However, supersonic cruise may be maintained by adjusting altitude to keep Mach number constant. Supersonic cruise data was obtained at 40,000 and 45,000 feet from accelerations and decelerations, and is presented in Figure 8. It is interesting to note that specific range at Mach 2.0 at 40,000 feet is only one-fourth that obtained at the recommended cruise speed at 35,000 feet.

Speed brake effectiveness is satisfactory. Deceleration with speed brake extension is higher than that obtained with the F-100. The speed brake switch must be left in the "up" position during normal flight to prevent the brakes from being "sucked" open. This condition is unsatisfactory.

turning performance

Although the turning capability of the F-104A at subsonic speeds is poor, turning capability at supersonic speeds is outstanding. Data from steady turns is presented in Figure 13 and from decelerating turns in Figure 14. A brief summary of both types of turn is shown for a weight of 16,000 pounds:

MAXIMUM LOAD FACTORS IN STEADY FLIGHT

ALTITUDE — FT.	MACH NUMBER	LOAD FACTOR — G
35,000	1.5	2.7
35,000	1.7	2.85
35,000	1.9	2.7
40,000	1.5	2.1
40,000	1.7	2.25
40,000	1.9	2.15
45,000	1.5	1.7
45,000	1.7	1.8
45,000	1.9	1.7

CHANGE IN HEADING FOR LOSS IN MACH NUMBER OF .1 DURING DECELERATING TURNS

ALTITUDE — FT.	LOAD FACTOR — G	MACH NUMBER	°HEADING/ .1 MACH NUMBER
40,000	3	1.5	50
40,000	3	1.7	60
40,000	3	1.9	61
40,000	4	1.5	28
40,000	4	1.7	30
40,000	4	1.9	29
50,000	2	1.5	39
50,000	2	1.7	44
50,000	2	1.9	41
50,000	3	1.5	20
50,000	3	1.7	21
50,000	3	1.9	21
55,000	2	1.5	24
55,000	2	1.7	25
55,000	2	1.9	25

descents

No particular descent technique was determined during Phase II tests. Satisfactory let-down from maximum speed at maximum altitude or from zooms can be accomplished by reducing power to military. The aircraft will decelerate with decreasing altitude, reaching subsonic speed at about 30,000 feet. Constant airspeed descents provide more uniform pitch angles than do constant Mach number descents, and are considered to be more practical. Maximum range will probably be achieved by reducing power to idle when subsonic flight is reached, and maintaining a constant glide speed of about 290 knots IAS. If a higher rate of descent is desired, the speed brake can be extended or let-down speed increased.

landing

Initial approach should be made near 300 knots* with flaps extended to the take-off position immediately before "break". Two features of landing flap operation are undesirable. First, an objectionable lateral trim change occurs with extension of the landing flaps. Second, positive selection of take-off flaps is not provided. The flaps may be raised inadvertently to the full up position when making a go around. A wide pattern with power on is made. The gear is lowered on the down wind leg opposite the end of the runway at 230 to 250 knots. Airspeed is held at 220 knots on the base leg until flaps are extended to the landing position. The speed is then allowed to decrease and the final turn completed at not less than 190 knots. A speed on final approach of 160 knots for a single airplane and 170 knots for formation flight is recommended. Partial power reduction and speed brake deployment during flare decelerate the aircraft to a normal touchdown speed of 145 knots or to 140 knots for a minimum distance landing. The nose wheel is lowered immediately after touchdown and the drag chute is deployed. Nose wheel steering is engaged as the drag chute is deployed to augment the rudder and brakes for directional control. Heavy braking should not be attempted until the chute has deployed since the sudden deceleration caused by chute deployment tends to make the pilot increase brake pedal force more than desired.

Landing performance was obtained with flaps in the landing position and with the boundary layer control operating. A graphical presentation is made in Figure 17 for a weight of 14,500 pounds and is summarized in the following table:

** All speeds recommended are indicated airspeeds unless otherwise noted.*

LANDING PERFORMANCE

	True Speed at Touchdown kts.	IAS at Touchdown kts.	True Speed Over 50 Foot Obstacle—kts.	Ground Roll ft.	Total Distance Over 50 Foot Obstacle—ft.
W/Drag Chute	142	140	155	2820	5550
Wo/Drag Chute	142	140	155	3080	—

These distances could be reduced by improving brake effectiveness, and by reducing the time required for the chute to deploy in order that heavy braking may be applied sooner.

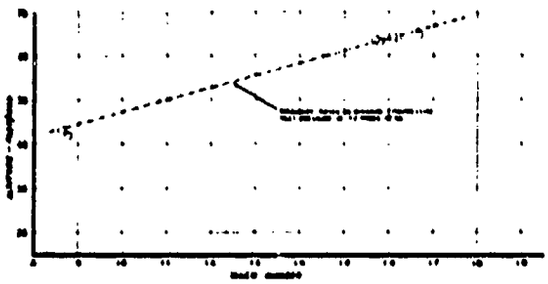
Reliability of the drag chute was unsatisfactory. Three failures were experienced during the program.

No dead stick landings or simulated dead stick landings were made with the F-104A. An evaluation of simulated dead stick landings was made in the XF-104 prior to the Phase II program. A high key of 18,000 feet was required with gear down and take-off flaps to complete a 360 degree overhead approach. A rate of descent of approximately 10,000 feet per minute can be expected with a dead engine. A speed of 240 knots should be maintained in the landing pattern to allow sufficient maneuvering for flare and touchdown at 190 knots. The pilot has little margin for error in completing a landing pattern. Even if a successful demonstration of a dead stick landing is made on Rogers Dry Lake in an F-104A, the feasibility of landing on operational airstrips is questionable. Bail-out could be the other alternative in tactical use.

engine performance

It was known prior to the initiation of the Phase II tests that the engine in the F-104A did not meet the guarantees in thrust and that specific fuel consumptions were higher than design values. In addition, other engine deficiencies limit the operational use of the F-104A.

- *Afterburner Ignition and Operation:* Afterburner lights could not be obtained consistently above 29,000 feet at subsonic speed, and afterburner operation was limited in altitude to 40,000 feet at subsonic speeds and to 65,000 feet at 1.7 Mach number during zoom climbs. Above these altitudes compressor inlet pressure was below the minimum required for afterburner operation. The afterburner blew out on five occasions during supersonic zooms, but did not cause compressor stalls. Two subsonic blowouts at approximately 40,000 feet induced compressor stalls and one engine flameout.



- *RPM Restriction:* The "Phase O" engine cannot be operated continuously between idle and 82 percent rpm due to a critical harmonic frequency in the compressor blades. This restriction is particularly objectionable during landing.

- *Engine Cooling:* Cooling of the engine during ground operation is unsatisfactory if the cockpit air conditioner is in operation. This required that all taxiing be done with the temperature control in the "RAM AIR" position.

- *Throttle Operation:* The force required to move the throttle is abnormally high, and no friction control is provided. Friction is so great that metal flakes are ground off the throttle arm during formation flights when the throttle is used extensively. Graphite lubrication temporarily corrected the situation and reduced the force required to move the throttle, but after 30 minutes of formation flying the lubrication had worn away.

  A dead band in the throttle position makes adjustments in power uncertain. This dead band allows afterburner operation to continue after the throttle has been moved from the afterburner detent to the military power position, and causes excessive throttle movement in formation flying.

  Throttling in afterburner does not provide a uniform variation in thrust. Three discontinuities are evident when moving the throttle from minimum afterburner to maximum power, the most noticeable being when the change is made from sector to uniform burning.

  The throttle movement in afterburner is too short. The restricted throttle movement together with the dead band and discontinuities in thrust make adjustment of intermediate power settings virtually impossible, and preclude formation flying at supersonic speeds.

  The control would be more useful for supersonic formation flying if engine speed could be varied while in minimum afterburner to provide airplane speed control in the extremely wide range between military power and minimum afterburner.

  Secondary airflow was controlled by means of a test installation. By-pass area was selected manually by operating three switches controlling three segments of flaps. Areas recommended by the contractor, as listed below were used throughout the test program.

	BY-PASS AREA IN.	FLAP POSITION A	B	C
BELOW 1.5 MACH NUMBER	44	Open	Closed	Closed
ABOVE 1.5 MACH NUMBER	82	Open	Closed	Open

- *Intercept Missions:* Simulated intercept missions, representative of those planned by the Air Defense Command, were made. Only maximum power, short range intercept missions were flown, and the following profile was followed in all cases:

  1. Maximum power take-off and climb to 35,000 feet.
  2. Acceleration at 35,000 feet to 1.9 Mach number.
  3. Climbing turn at 1.9 Mach number holding 1.5g through 180 degrees.
  4. Climb and acceleration to Mach 2.0.
  5. Zoom climb and interception at 63,000 feet. (Note: All missions were made without missiles and with the standard internal fuel quantity of 763 gallons, which resulted in an engine start weight of 18,500 pounds.)

  The above conditions result in interception at only 10 miles from take-off, after an interval of 10 minutes. A considerable improvement in this radius of intercept may be made, provided more time is available: First, if military power is used during take-off and climb to 35,000 feet, 30 more miles are covered while using 300 pounds less fuel; Second, the mission outlined above results in a fuel quantity remaining at intercept of 1500 pounds. With 200 to 300 pounds used during descent, the fuel reserve for landing is considered more than adequate. Best cruise speed could be maintained following a military power climb for a distance of about 70 miles and the same profile followed through intercept. Sufficient fuel for descent and cruise back would be available to permit landing with a fuel reserve of approximately 1000 pounds. This procedure would result in an intercept radius of 110 miles. A further

increase in radius would be obtained during intercepts made at lower altitudes. An additional increase would also result from accelerations at somewhat higher altitudes, but at the expense of time to intercept.

The above discussion is predicated on standard day temperatures. Acceleration performance in particular varies widely with ambient temperatures. The variation in acceleration performance from a Mach number of 1.05 to 2.0 is shown below:

VARIATION OF ACCELERATION PERFORMANCE AT 35,000 FEET WITH TEMPERATURE

	TEMPERATURE		
	10 DEGREES BELOW STANDARD	STANDARD (— 55°C)	10 DEGREES ABOVE STANDARD
TIME — MINUTES	2.15	2.95	4.35
DISTANCE TRAVELED — NAUTICAL MILES	30	43	65
FUEL USED — POUNDS	830	1120	1580

In order to obtain maximum radius of action, acceleration must be made at the optimum altitude.

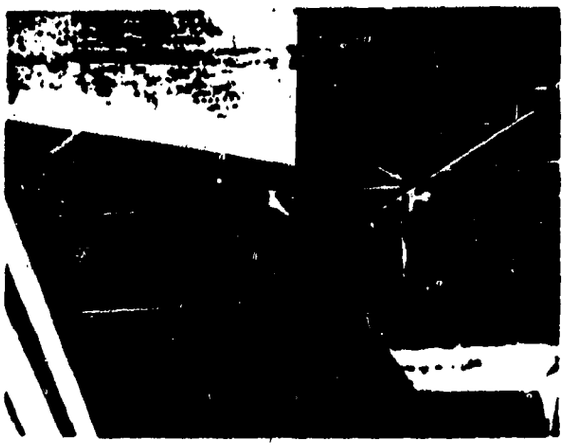
Considering the high speeds and altitudes planned for the F-104A at intercept, its limited endurance, and variation in performance with ambient conditions, an interception must, of necessity, be a very precise maneuver. It becomes evident that successful integration of the F-104A into the SAGE system is essential to make the F-104A of value as an interceptor.

A sensitive accelerometer becomes essential during an intercept mission. First, it aids the pilot in making a transition from acceleration after takeoff to the desired climb schedule. Second, it is necessary during a controlled turn to avoid undue errors in position. Finally, it should be used during zoom climbs so they may be made with a minimum loss of energy.

TEST RESULTS: STABILITY AND CONTROL

control system

The excellent lateral and longitudinal control systems in the XF-104 have been fairly well duplicated in the F-104A. A reduction of force in the lateral control system is desirable to make it match the feel system in the longitudinal control. The longitudinal trim response is excessively slow. Trim rate is satisfactory once a constant rate has been attained.



Unsatisfactory damper operation existed throughout most of the program. There were two damper failures in flight, and during approximately six flights, improper damper operation produced snaking or residual directional oscillations in varying degrees of amplitude that were not correctable by varying the gain settings. Two other test aircraft flown by the Air Force revealed the same deficiency. The dampers required considerable maintenance during the test program. The three axes damper system needs to be greatly refined before reliable operation can be expected.

Quantitative stability data was gathered with all dampers operating, and a qualitative evaluation was made with the dampers inoperative.

Control friction was determined in a closed hangar. Plots of surface deflection versus force are presented in Figures 61 through 63, and breakout forces are listed in the following table:

	STATIC FRICTION FORCE — LB.	MAXIMUM ALLOWABLE FORCE FROM MIL-F-8785(ASG) — LB.
STABILIZER	4	3
AILERON	5	2
RUDDER	40	7

Stabilizer force is over the specification limit but is not considered objectionable. In-flight aileron breakout forces become higher than those obtained on the ground because of cable binding in the fuselage and are unsatisfactory. Rudder breakout force is satisfactory although it is made high to keep the rudder from being moved from its neutral position as discussed under "Directional Control".

level flight handling characteristics

Longitudinal Stability: Dynamic longitudinal stability with the pitch damper operating properly is satisfactory at all subsonic speeds, but becomes unsatisfactory at supersonic speeds. (Reference Figures 23 through 34.) Poor damping at supersonic speeds may have resulted from insufficient testing by the contractor to establish an optimum pitch damper gain setting.

Static longitudinal stability is satisfactory in spite of the wide cg travel with [illegible] consumed. Very minor trim changes, hardly [illegible] able to the pilot, occur at low supersonic speeds.

Maneuvering capabilities are poor at subsonic speeds. (Reference Figures 38 and 39.) Initial buffet occurs at 1.4g in the power-approach configuration at 200 knots IAS, dictating an unusually wide landing pattern. Lack of maneuverability makes combat at high subsonic speeds inadvisable. Maneuvering flight capabilities at supersonic speeds are outstanding. Maneuvering flight is characterized by large stick displacement from trim at high load factors. The light stick forces are satisfactory and permit this movement with no great exertion. There are areas in the supersonic flight regime between 35,000 and 45,000 feet where the maximum load factor is dictated by limit stabilizer deflection rather than

limit load factor, pitch-up, or deterioration of directional staiblity. This is an unsatisfactory condition and does not meet the requirements of MIL-F-8785 (ASG). Lack of stabilizer effectiveness during take-off roll limits the nose wheel lift-off speed to a minimum of 140 knots IAS with maximum stabilizer deflection. This condition is not considered serious with the existing cg at take-off. The recommended addition of internal fuel and resulting forward cg travel, however, would probably provide unsatisfactory stabilizer effectiveness.

Longitudinal trim changes are generally small, as indicated in the following table. None are considered large enough to be objectionable.

INITIAL TRIM CONDITIONS							
ALTITUDE — FT.	IAS — KTS.	GEAR	FLAPS	POWER	CONFIGURATION CHANGE	PARAMETER HELD CONSTANT	STICK FORCE AFTER CONFIGURATION CHANGE*
10,000	250	Down	Up	PLF	Flaps to TO	Altitude	8
10,000	250	Down	TO	PLF	Idle power	Speed	—11
10,000	170	Down	TO	PLF	Landing flaps	Speed	— 4
15,000	230	Down	TO	TO	Gear up	R/C	— 6
40,000	285	Up	Up	Mil	Idle power	Altitude	0
40,000	285	Up	Up	Mil	Extend speed brake	Altitude	— 5

* Negative sign denotes push force. Maximum forces within 5 seconds are listed.

A nose up trim change results from speed brake deployment at speeds less than 1.75 Mach number. A transition to a nose down trim change occurs at 1.85 Mach number with no noticeable trim change between 1.75 and 1.85 Mach number.

■ *Pitch-Up Characteristics:* In general, the handling characteristics of the aircraft are excellent, but are offset by pitch-up characteristics which are as serious or worse than those exhibited by the F-101. Although no pitch-ups were made during these tests, characteristics preceding a pitch-up were explored for several flight conditions at subsonic speeds. As pitch-up is approached at .9 Mach number and below, buffet is encountered followed by a noticeable lateral instability. This characteristic is the same during both unaccelerated and accelerated flight. Although this buffet and instability would appear to constitute adequate warning it is felt that pilots will maneuver in buffet because of the wide buffet region and low load factors available at subsonic speeds, and in effect lose natural warning of pitch-up. The speeds listed below were determined from unaccelerated flight at a we[illegible] of 15,000 pounds with a center of gravity of 1[illegible] MAC.

GEAR	FLAPS	IAS AT INI[illegible] BUFFET [illegible]S.	IAS AT INITIAL LATERAL INSTABILITY
Up	Up	255	190
Down	TO	155	*
Down	Landing	**	148

* Not determined.
** No apparent buffet.

No pitch-ups have been made on an F-104A, but those made with the XF-104 were uncontrollable, becoming much more violent during accelerated flight. No pitch-ups have been encountered at supersonic speed, but it is predicted that very little or no warning will exist, and that the resulting load factors at the higher indicated airspeeds may be high enough to destroy the airplane. An automatic pitch control has been installed in all F-104's which is designed to push forward on the stick in the event of an impending pitch-up; however, the device has not been developed to the point where consistently satisfactory operation may be expected, and insuffi-

cient flight data has been gathered to set an optimum boundary to provide a minimum loss of maneuverability.

- *Directional Stability:* Tests for directional stability indicated satisfactory results to a Mach number of 1.90; however, all tests at high Mach number were restricted to 1g flight. Preliminary investigation by the contractor indicates a deterioration in directional stability at high Mach numbers, becoming more pronounced as angle of attack is increased. With the yaw damper inoperative the aircraft is safe for flight but is tactically unusable.

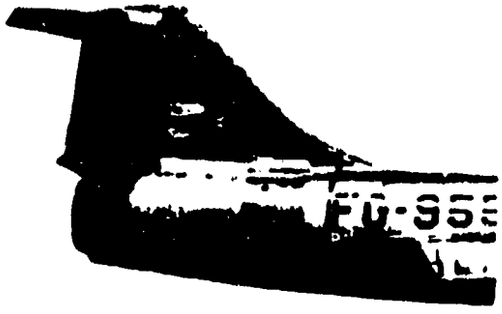
  The unboosted rudder control does not allow the pilot to move the rudder surface by an appreciable amount except at low speed. At supersonic speeds the F-104A is essentially a two-control airplane. Sideslip characteristic at low speed are satisfactory and no undesirable characteristics were noted with full rudder pedal deflection throughout the speed range. It was found best to fly with feet off the rudder pedals at cruise speeds and above to prevent inadvertently moving the rudder out of lock and allowing it to counteract consequent yaw damper action by floating in the opposite direction. The rudder lock develops play after a few flights. Periodic adjustment was necessary throughout the test program to prevent rudder float with consequent yaw damper action. This condition is unsatisfactory.

- *Lateral Stability:* A limit of one-half maximum aileron deflection in the clean configuration was imposed during these tests pending a complete inertial coupling investigation. Lateral control in the clean configuration does not meet the MIL-F-8785 (ASG) requirement for changing bank angle 100 degrees in one second after application of lateral control force. (A bank angle of about 40 degrees was obtained after one second at 40,000 feet at Mach number of both .9 and 1.9.) Full aileron deflection was permitted in the landing configuration and satisfactory roll rates were obtained. Very little adverse yaw resulted from rolls made in the clean configuration through 360 degrees. It should be noted that all rolls were made with 1g entry; adverse yaw is expected to increase with less than 1g entry at cruise speeds and above.

contractor development program

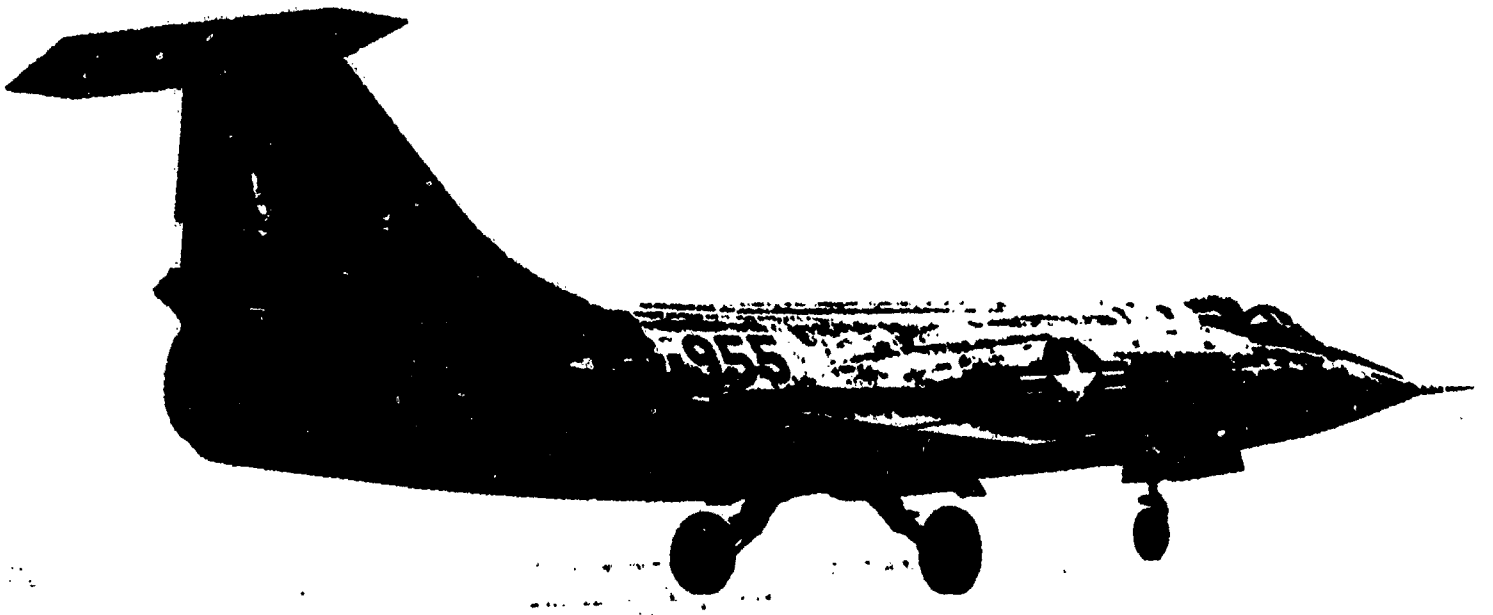
At this time the test aircraft inventory (comprised of the first 35 airplanes) are in various stages of acceptance flying, instrumentation and manufacture. The following items were installed on the Phase II test aircraft, but availability on other USAF test aircraft is questionable:

1. Engine by-pass air flap selection.
2. Boundary layer control.
3. Lateral control with reduced friction.
4. Symmetrical speed brake operation.
5. Firewall kit and three-bottle oxygen supply.



Certain deficiencies, discussed below, were known to exist when Phase II tests were initiated. Emphasis has been placed on correcting these deficiencies on production aircraft allocated to the Air Defense Command (article 36 and subsequent), but modification of USAF test aircraft has not been planned by the contractor. Delivery date of the first production aircraft is scheduled for March 1957.

- *Ventral Fin:* This fix is proposed to improve directional stability of the aircraft under critical flight conditions. The production plastic version, housing antenna equipment, is proposed for the first tactical aircraft. Until tested, the worth of this item remains questionable.
- *Auto Pitch Control:* The optimum setting has yet to be determined. Present contractor estimate of 1 December 1956 for an acceptable fix is felt to be unduly optimistic.
- *Aileron Stops:* Aileron throw will be restricted by aileron stops to approximately one-half maximum travel in the clean configuration. Inertial coupling tests will eventually determine the allowable aileron deflection.
- *Missile Launchers:* The acquirement of missile launchers for the test aircraft is expected to be late because of the change in mission for the F-101A in April 1956. It is unknown when testing will start on missile firing and what organization will conduct the test program.
- *Tip Tanks:* No tip tanks have been flown on the F-101A to date. It is expected that tip tanks will be available prior to the scheduled delivery date of the first production aircraft.
- *Pylon Tanks:* Since it was learned that pylon tanks will be required in conjunction with project Red Dog, there have been repeated delays in the negotiations between WSPO and the contractor. The contractor has just now received approval of design and is searching for a vendor. It is estimated that these tanks will be available no earlier than January 1958.
- *Spin Tests:* Contractor spin tests have been delayed. There is a requirement for spin information from contractor flight tests before the delivery of the aircraft to the Air Force. The contractor is further delaying his spin tests with WSPO approval by changing the schedule of flight test. After several lg pitch-ups have been made and it is determined that the aircraft is not prone to spin from unaccelerated flight, the No. 3 aircraft will be diverted to an "auto pitch control" program. This program is inadequate in that it will not provide early information on spin characteristics and characteristics of pitch-up in high-speed accelerated flight. For this reason it is desirable that the No. 3 aircraft be used for spin tests immediately and the other aircraft which are available be used for development of the auto pitch control device.



CONCLUSIONS

Performance of the F-104A is outstanding. Time to climb from brake release to 35,000 feet is approximately 2.5 minutes. Acceleration is rapid above 1.3 Mach number and excess thrust is still available at the limit speed of 2.0 Mach number. A service ceiling of about 55,000 feet is attained with maximum power at 1.9 Mach number and zoom climb may be made to much higher altitudes. A maximum of 70,000 feet may be reached, even with the afterburner blowing out at 65,000 feet. Turning capability at supersonic speed is excellent, but at subsonic speed the turning capability becomes poor and is suitable only for non-combat operations. Maneuverability is restricted at supersonic speeds between 35,000 and 45,000 feet by the lack of stabilizer effectiveness. A further loss in maneuverability will result from the proposed addition of internal fuel and resulting forward cg travel. Also, stabilizer effectiveness may become unsatisfactory at take-

off with additional internal fuel at the existing limit stabilizer deflection.

One of the most serious deficiencies in the aircraft is its limited combat radius. There is a critical need for an increase in thrust and a decrease in specific fuel consumption which will reduce the fuel required to accelerate, and for additional internal fuel. A considerable increase in thrust will be required to retain satisfactory performance in the supersonic region when external stores are added to the aircraft.

Interception with the F-104A demands a very precise maneuver. Successful integration of the F-104A into the SAGE system is essential to make it of value as an interceptor.

The fine handling characteristics in the normal flight region are offset by an uncontrollable pitch-up at high angles of attack which increases in severity during accelerated maneuvers. Characteristics preceding and during pitch-up are similar to those experienced in the F-101, but result in more violent maneuvers. It is expected that little or no warning will occur at supersonic speeds and that the high load factors encountered may lead to destruction of the aircraft.

In addition, it is expected that inertial coupling characteristics at load factors of less than 1g, and directional stability near limit speed at high angles of attack will be unsatisfactory. This is concluded from studies made by the contractor in areas which were restricted during Phase II tests and where very little flight testing has been accomplished by the contractor.

Rolling performance is unacceptable at all speeds in the clean configuration with the existing limit of one-half of maximum aileron deflection.

Damper operation must be made more reliable to insure tactical usefulness.

RECOMMENDATIONS

It is recommended that:

1. The following items be completed before the airplane is released for tactical use:

a. Pitch-up boundary investigation and consequent development of the automatic pitch control.

b. Spin tests.

c. Dead stick landings.

d. Determination of directional stability characteristics at high speed.

e. Thorough investigation of inertial coupling characteristics.

2. The combat radius of action be increased by:

a. Increasing engine thrust and decreasing specific fuel consumption to meet original guarantees.

b. Increasing the internal fuel supply to a maximum.

3. Rolling performance be increased to provide a minimum response of 100 degrees change in bank angle in one second in the clean configuration at cruise speeds and above.

4. The following engine refinements be made:

a. Remove restriction on engine rpm at low speeds.

b. Remove restriction on ground operation with cabin air conditioning operating.

c. Make the engine air by-pass flap operation automatic.

d. Install compressor inlet temperature gage next to the airspeed indicator in all aircraft. Improve the accuracy of the temperature pick-up and program the warning light to flash 10 degrees ahead of the critical temperature.

5. Afterburner operation be improved to:

a. Permit a more positive ignition procedure and increase subsonic ignition reliability up to 49,000 feet.

b. Extend afterburner blowout boundary to include subsonic speeds up to 50,000 feet and supersonic speeds up to 75,000 feet.

6. Throttle control of the engine be improved by:

a. Eliminating the dead band in the throttle control.

b. Providing a smoother and more precise selection of afterburner power.

c. Providing main engine power control in the minimum afterburner position.

d. Reducing throttle friction.

7. The three axes damper system be improved by:

a. Optimizing the gain settings, and increasing pitch damper effectiveness if longitudinal damping remains unsatisfactory.

b. Improving system reliability.

c. Establishing more efficient trouble shooting procedures.

8. Improve stabilizer effectiveness so that:

a. Maneuvering flight capabilities will not be limited by stabilizer deflection.

b. Satisfactory nose wheel lift off speeds are maintained with the addition of internal fuel and external stores.

9. The lateral control breakout force and force gradient be decreased to the extent that it is compatible with the longitudinal control.

10. Wheel brake effectiveness be increased.

11. Tire strength be increased.

12. Drag chute reliability be improved and time for deployment reduced.

13. A sensitive accelerometer 3 inches in diameter be installed in place of the second heading indication on the left side of the instrument panel.

14. The tendency for the rudder lock to develop play in a relatively short number of flights be eliminated.

15. The longitudinal trim response be increased by 100 percent.

16. The lateral trim change that occurs with extension of the flaps from take-off to landing position be eliminated.

17. The flap selector be modified to require outboard movement when going from the take-off position to the landing position.

18. The brightness of lights on the master warning panel, be increased and the color of the master caution light be changed from red to amber.

19. The size of the drag chute handle be increased and the handle extended an additional inch from the instrument panel.

20. The circuit breakers be colored to reveal a red color on the stem when a breaker has "popped".

21. A new GFE airspeed indicator be developed.

22. The need for leaving the speed brake switch in the "up" position during normal flight be eliminated.

23. Items discussed under "Contractor Development Program" be installed in test aircraft in cases where these items become essential in making an adequate evaluation.

The following recommendations are based on information presented in Appendix III.

1. Emphasize accelerated development of the Model "D" seat which on the basis of scale testing indicates a potential capability of providing for successful emergency escape over a greater portion of the performance range of this aircraft.

2. Modify standardized warning streamer requirements for seat initiator safety pins to allow the item to be placed over the stick.

3. Provide an improved parachute support to be utilized with the MC-1 aircraft cushion which will effectively remove parachute weight from the shoulders of the pilot.

4. Remove the adjustment mechanism for the lap belt tie down strap from its present position below the seat pan and install a simple buckle adjustment similar to that utilized on the shoulder harness. This arrangement will make the adjustment readily available to the pilot above the level of the seat pan. Strap ends may then be firmly attached to the seat proper.

5. Although unusual structural and canopy opening features are present in this aircraft the addition of external canopy release mechanism to the left side of the fuselage should be given consideration. External canopy release instructions should be emphasized with more attention-diverting markings and colors.

6. The emergency escape data provided to the pilot in the emergency procedure section of the proposed handbook should be expanded to include more detail data on potential safe ejection areas, critical times, etc.

7. Improved integrated oxygen equipment and the seat cushion survival kit assembly should be incorporated as soon as possible.

8. Location of oxygen hose should be such that minimum clutter occurs on the body of the pilot.

9. The landing gear down lock override should be located or re-designed so as to preclude interference from panel structure.

10. The manually activated controls on the forward instrument panel which include manual gear release, turbine extension, pylon tank jettison, and tip tank jettison should be identified with colors symbolizing emergency controls. This is also true of the manual hatch release.

11. Determine the feasibility of interchanging the Radio Magnetic Indicator with a V-8 or similar type settable dial moving pointer indicator.

12. Provide a snap up card holder to hold index charts of frequencies assigned to the UHF channels. This should be located so as to be legible and accessible to the pilot.

13. Identify switch guards on alternators, fuel tank jettison controls, and the stability augmentation control panel with word marking on the switch covers proper.

14. Provide storage area for maps or reference data readily available to the pilot.

15. Color-code prime radar control switches on the armament panel as an aid for distinguishing functions involved.

16. Evaluate the requirement for an air source as required for the ventilating garment component of the anti-exposure suit assembly.

This appendix includes a description of the methods employed in reducing the test data to standard conditions. The following references have been used and will be mentioned during the discussion:

1. Air Force Flight Test Center Technical Note R-12, "Standardization of Take-Off Performance Measurements for Airplanes".

2. AF Technical Report Number 6273, "Flight Test Engineering Manual".

3. General Electric Report Number R55AGT400, "Estimated Minimum Performance of the General Electric J79 Turbojet Engine.

APPENDIX I

Data Analysis Methods

■ take-offs

Corrections to ground rolls and air distances for wind velocity were made using equations in reference 1. Remaining corrections, also based on reference 1 were made as follows:

$$\frac{S_{g_s}}{S_{g_t}} = \left(\frac{W_s}{W_t}\right)^{2.3} \left(\frac{\sigma_s}{\sigma_t}\right) \left(\frac{\bar{F}_s}{\bar{F}_t}\right)^{-1.3}$$

$$\frac{S_{a_s}}{S_{a_t}} = \left(\frac{W_s}{W_t}\right)^{2.3} \left(\frac{\sigma_s}{\sigma_t}\right)^{-0.7} \left(\frac{\bar{F}_s}{\bar{F}_t}\right)^{-1.0}$$

where:

W = airplane gross weight, pounds

σ = air density ratio

$\bar{F}$ = thrust at mean speed, pounds

subscripts $_s$ and $_t$ refer to standard and test conditions.

■ level flight

Subsonic power required data was obtained in stabilized level flight using the constant weight-pressure ratio technique. Since nozzle area remains constant from an rpm of 85 percent to normal rated power at 100 percent, conventional methods of correcting data to standard conditions were employed.

At military and afterburning power settings the engine no longer behaved as a simple jet and it became necessary to make corrections for variations in temperature based on the engine manufacturer's estimated data. Accelerations with intermediate afterburner settings were made to limit speed, and decelerations with minimum afterburning were made from limit speed until a stabilized speed was approached. Fuel flows were corrected to standard temperature using plots of $\frac{\Delta(w_f/\delta_{t_2})}{\Delta T_a}$ versus T_a derived from reference 3.

Values of excess thrust were computed and the fuel flows required for "stabilized" level flight determined from plots of $\frac{\Delta w_f}{\Delta T_{ex}}$ versus (T_a) also derived from reference 3.

where:

w_f = fuel flow, pounds/hour
T_{ex} = excess thrust, pounds
δ_{t_2} = compressor inlet pressure/ambient sea level pressure
T_a = ambient temperature, °K

Fuel flows were then corrected to a standard weight by interpolating between test values of weight-pressure ratio on a plot of fuel flow versus Mach number.

climbs

It is convenient to work with the following equation to find rate of change of specific energy at test weight and thrust:

$$\frac{1}{W}\frac{dE}{dt} = \sqrt{\frac{T_{at}}{T_{as}}} \cdot \frac{1}{\Delta t}\left[\left(\frac{V_{t_{a_1}} + V_{t_{a_2}}}{2g}\right)\left(V_{t_{a_2}} - V_{t_{a_1}} + w_2 - w_1\right) + \left(H_{c_2} - H_{c_1}\right)\right]$$

where:

W = airplane gross weight, pounds
$\frac{1}{W}\frac{dE}{dt}$ = rate of change of specific energy, feet/minute
T_{at} = test ambient temperature, °K
T_{as} = standard ambient temperature, °K
Δt = difference in time, minutes
V_{t_a} = true speed, feet/second
g = acceleration of gravity, 32.16 feet/second²
w = wind speed, feet/second, (tailwind +)
H_c = calibrated altitude

subscripts 1 and 2 refer to data at times of 1 and 2.

Temperature corrections were made by using plots showing variation in corrected net thrust with temperature and converting the thrust correction to a rate of climb correction through the following equation:

$$\Delta R/C = \frac{\Delta F_n \cdot V_t \cdot 60}{W}$$

where:

$\Delta R/C$ = rate of climb correction for deviation in temperature, feet/minute
ΔF_n = net thrust correction for deviation in temperature, pounds
V_t = true speed, feet/second
W = airplane gross weight, pounds

Weight corrections were determined from equations found in reference 2. Summing the rates of climb corrections and test rate of change of specific energy produces a standard rate of climb at zero acceleration. Where climbs were made at other than constant true speed the following equation was used:

$$R/C_s = \frac{\frac{1}{W_s}\frac{dE}{dt}}{1 + \frac{V_t}{g}\frac{dV}{dh}}$$

where:

R/C_s = standard rate of climb, feet/minute
$\frac{1}{W_s}\frac{dE}{dt}$ = standard rate of change of specific energy, feet/minute
$\frac{dV}{dh}$ = rate of change of speed with altitude, feet/second per foot

Specific energy was computed in order to find time to climb:

$$\frac{E}{W} = \frac{V_t^2}{2g} + H_c$$

E/W was then plotted against the reciprocal of the rate of change of specific energy, $W\frac{dt}{dE}$. Times to climb were found by integrating under the curve.

It was found necessary to make corrections to altimeter readings for lag in the static pressure. Lag corrections were determined from the equation:

$$\Delta H = \lambda_{ML} \cdot \frac{\mu_a}{\mu_{ML}} \cdot \frac{1}{\delta_a} \cdot \frac{dh}{dt}$$

where:

ΔH = altimeter lag, feet

λ_{SL} = lag constant at sea level, seconds

$\frac{M_\mu}{\mu_{SL}}$ = viscosity coefficient ratio

δ_a = ratio of pressure at altiude to that at sea level

$\frac{dh}{dt}$ = rate of climb, feet/minute

A sea level lag constant, λ_{SL}, was determined from a comparison of altitude recorded in the airplane to that recorded by Askania cameras during maximum power climb.

accelerations

Rate of change of specific energy was found for standard conditions as described under Climbs. This was converted to excess thrust by the equation:

$$T_{ex} = \frac{\frac{1}{W}\frac{dE}{dt} \cdot W}{V_t \cdot 60}$$

where:

T_{ex} = excess thrust, pounds

Time to accelerate was also computed as for the climbs by integrating under a curve of

$$W\frac{dt}{dE} \text{ versus } E/W.$$

turns

Data was plotted in the form:

$$\frac{T_{ex}}{\delta_a} \text{ versus } \left(\frac{W}{\delta_a} \cdot \frac{n}{M}\right)^2$$

where:

n = load factor

Data for this plot was taken from level accelerations, steady turns, and decelerating turns. Lines of constant Mach number were then faired through the points. Data for specific conditions was expanded from this plot. Radius of turn was computed from the equation:

$$R = \frac{V_t^2}{11.3\sqrt{n^2 - 1}}$$

where:

R = turning radius, feet

Deceleration for turns using load factors higher than maximum for steady conditions was computed from:

$$\frac{1}{V_t} \cdot \frac{g}{W}\frac{dE}{dt}$$

Change in heading with loss in Mach number was calculated from turning radius and deceleration.

landings

Ground rolls were corrected with the equation:

$$\frac{S_{g_s}}{S_{g_{t_w}}} = \left(\frac{V_{t_d} + V_w}{V_{t_d}}\right)^{1.85} \frac{P_{st}}{P_{as}} \frac{T_{as}}{T_{st}}$$

where:

S_{g_s} = standard landing distance, feet

$S_{g_{t_w}}$ = test landing distance, feet

V_{t_d} = airspeed at touchdown, feet/second

V_w = component of wind velocity parallel to runway, feet/second (headwind +)

Correction to air distance for wind velocity was made as for take-offs.

pitch and yaw corrections

Errors in indicated angles of attack and sideslip resulted from differences between local airflow over the vanes, and free stream air flow. No calibrations have been made on an F-104A but those made on the XF-104 produced the following results:

ANGLE OF ATTACK

	TRUE ANGLE — DEG.	INDICATED ANGLE — DEG.	$\frac{d\alpha \text{ IND.}}{d\alpha \text{ TRUE}}$
BELOW 1.2 MACH NUMBER	0	—4	1.75
OVER 1.4 MACH NUMBER	0	—2	1.5

ANGLE OF YAW

	TRUE ANGLE — DEG.	INDICATED ANGLE — DEG.	$\frac{d\psi \text{ IND.}}{d\psi \text{ TRUE}}$
BELOW 1.2 MACH NUMBER	0	0	1.75
OVER 1.4 MACH NUMBER	0	0	1.5

performance and stability plots

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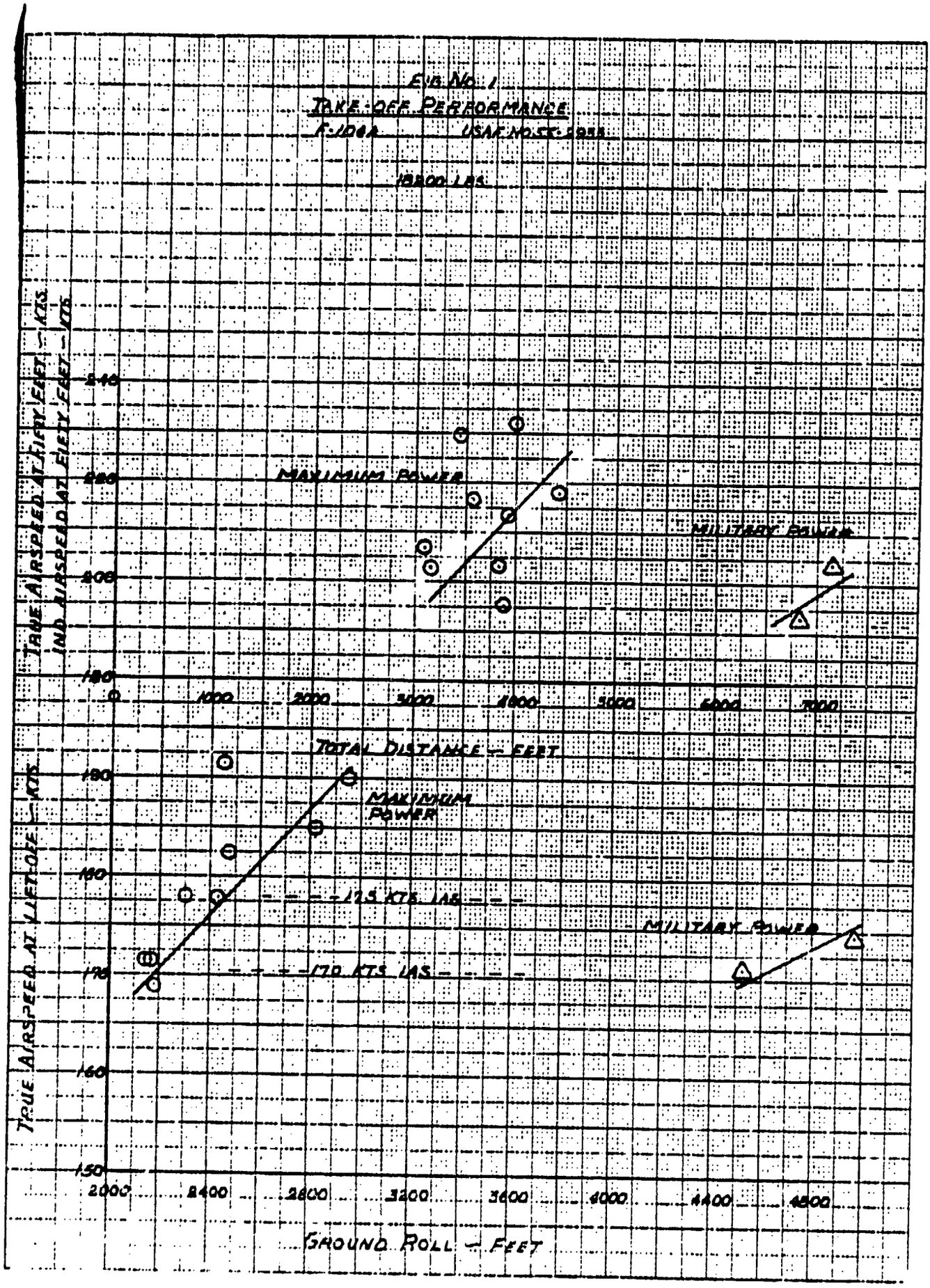


FIG. No. 1
TAKE-OFF PERFORMANCE
F-106A
USAF NO. 55-2955
18200 LBS.
TRUE AIRSPEED AT FIFTY FEET — KTS.
IND. AIRSPEED AT FIFTY FEET — KTS.
240
220
200
180
MAXIMUM POWER
MILITARY POWER
0
1000
2000
3000
4000
5000
6000
7000
TOTAL DISTANCE — FEET
TRUE AIRSPEED AT LIFT-OFF — KTS.
190
180
170
160
150
MAXIMUM POWER
175 KTS. IAS
170 KTS. IAS
MILITARY POWER
2000
2400
2800
3200
3600
4000
4400
4800
GROUND ROLL — FEET

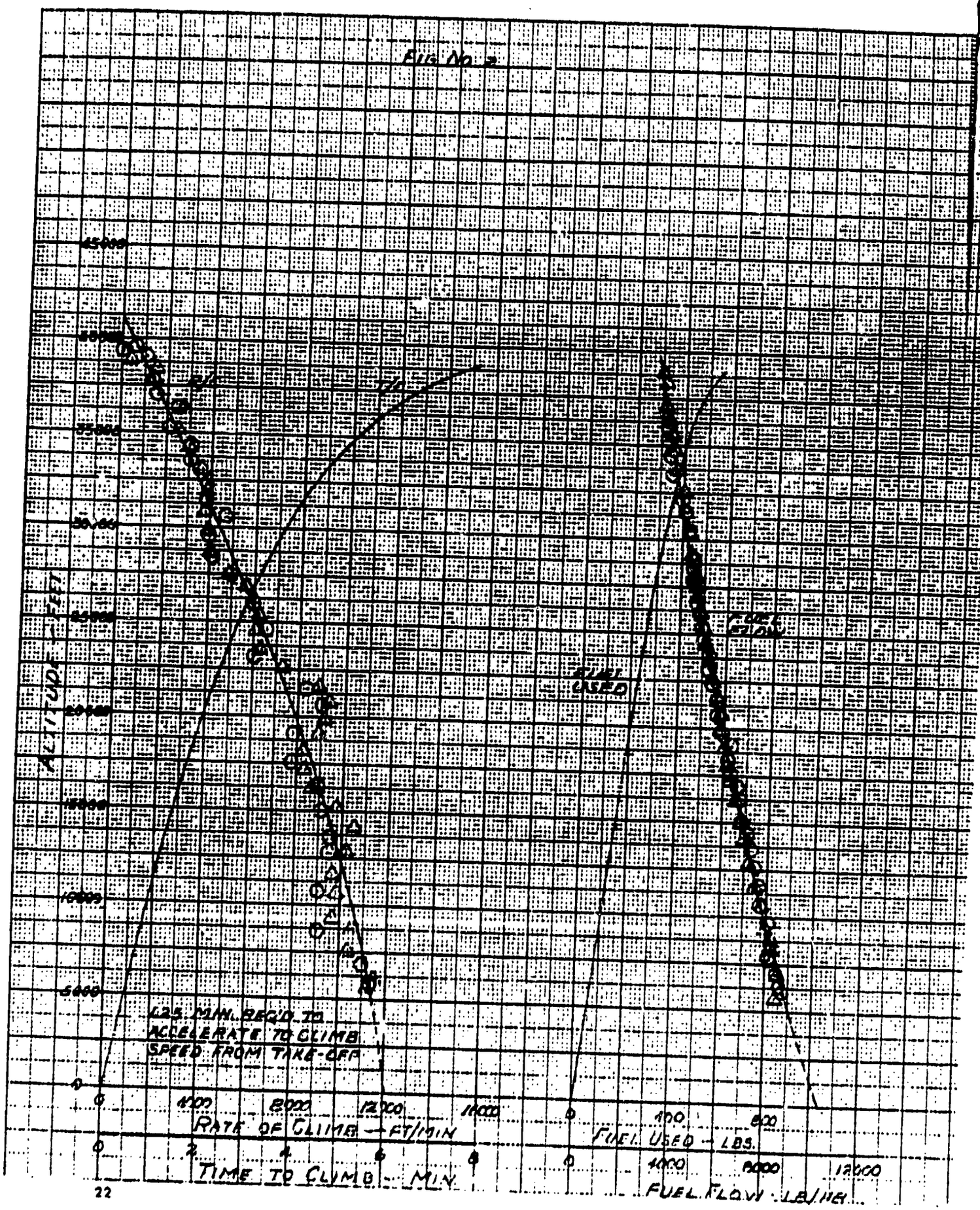


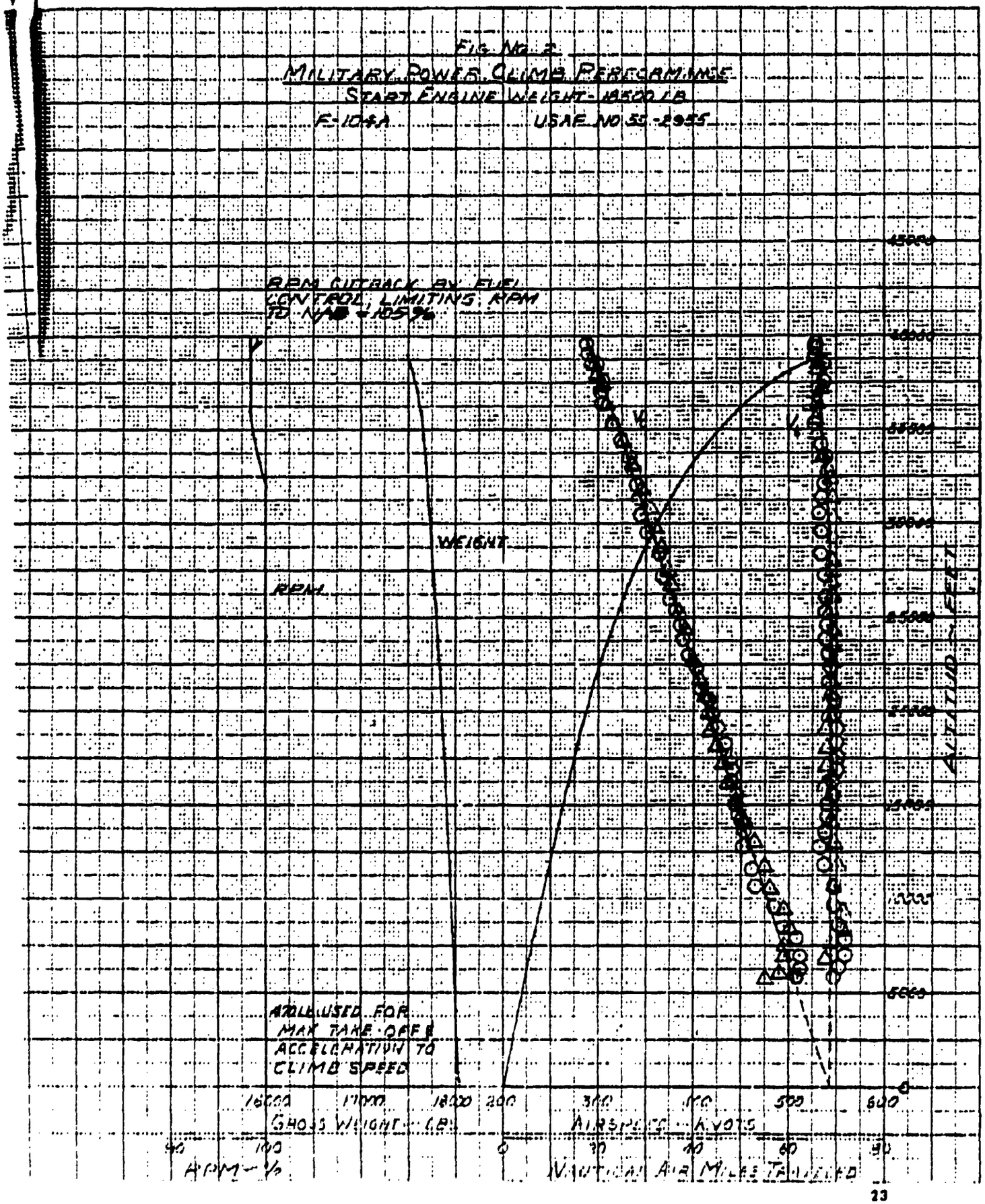
FIG No 2
ALTITUDE - FEET
45000
40000
35000
30000
25000
20000
15000
10000
5000
0
1.25 MIN. REQ'D TO
ACCELERATE TO CLIMB
SPEED FROM TAKE-OFF
FUEL USED
FUEL FLOW
0
4000
8000
12000
16000
RATE OF CLIMB - FT/MIN
0
400
800
FUEL USED - LBS.
0
2
4
6
8
TIME TO CLIMB - MIN
0
4000
8000
12000
FUEL FLOW - LB/HR

FIG. NO. 2

MILITARY POWER CLIMB PERFORMANCE

START ENGINE WEIGHT - 18500 LB

F-104A USAF NO 56-2955



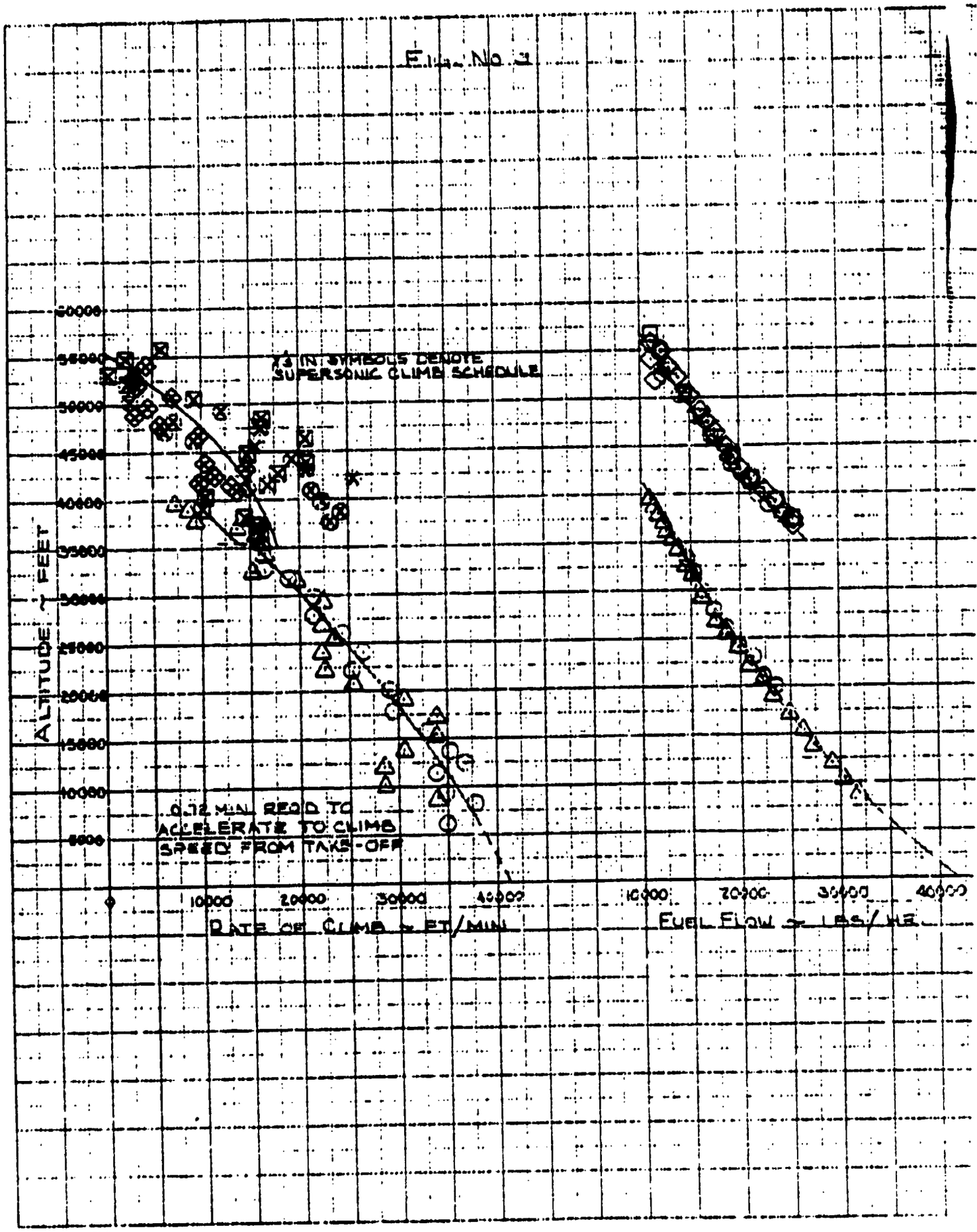


FIG. No. 3
X'S IN SYMBOLS DENOTE
SUPERSONIC CLIMB SCHEDULE
0.72 MIN. REQ'D TO
ACCELERATE TO CLIMB
SPEED FROM TAKE-OFF
ALTITUDE ~ FEET
RATE OF CLIMB ~ FT/MIN
FUEL FLOW ~ LBS/HR
10000
20000
30000
40000

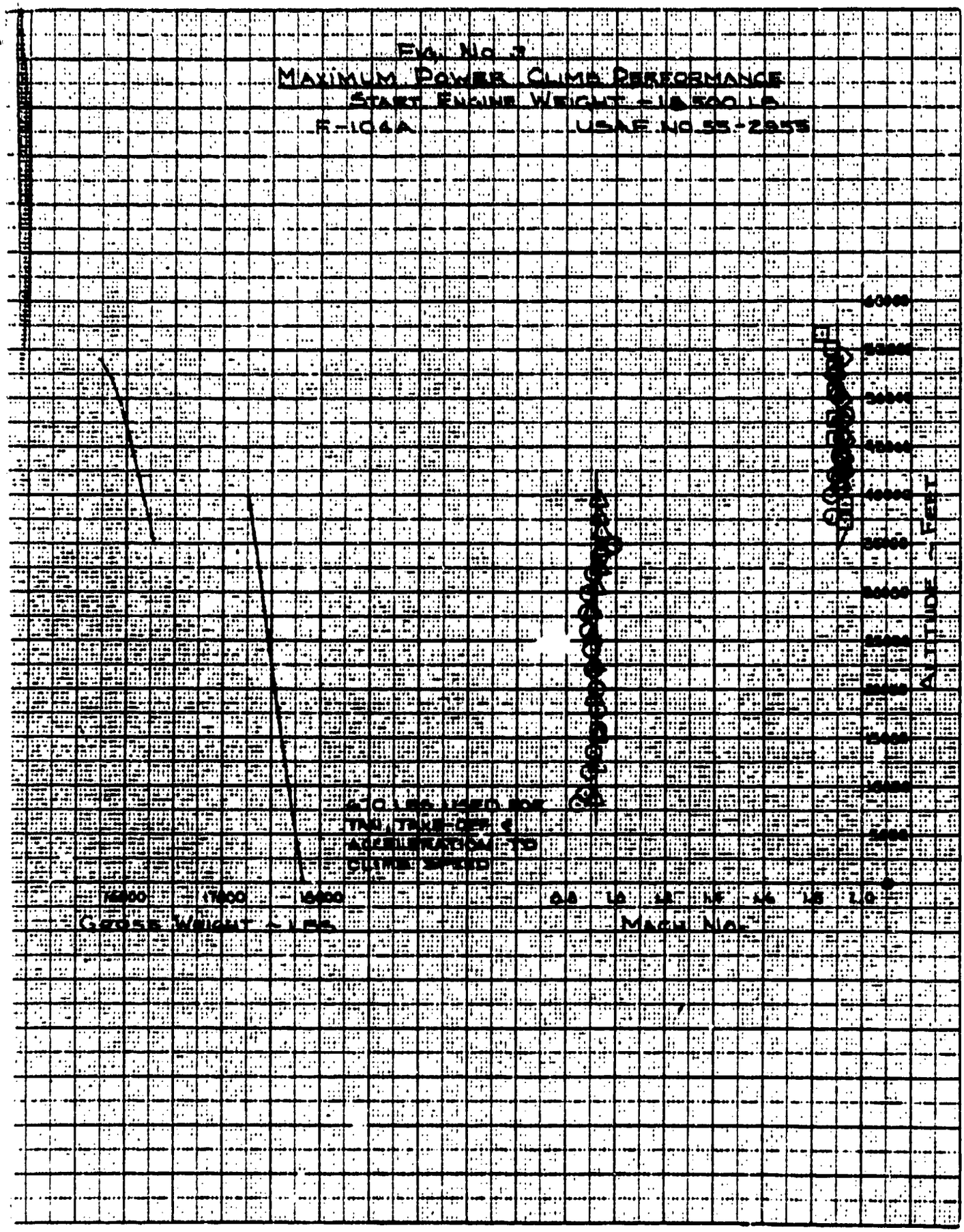


FIG. NO. 3
MAXIMUM POWER CLIMB PERFORMANCE
START ENGINE WEIGHT ~ 18,500 LB
F-104A
USAF NO 55-2955
ALTITUDE ~ FEET
GROSS WEIGHT ~ LBS
MACH NO.

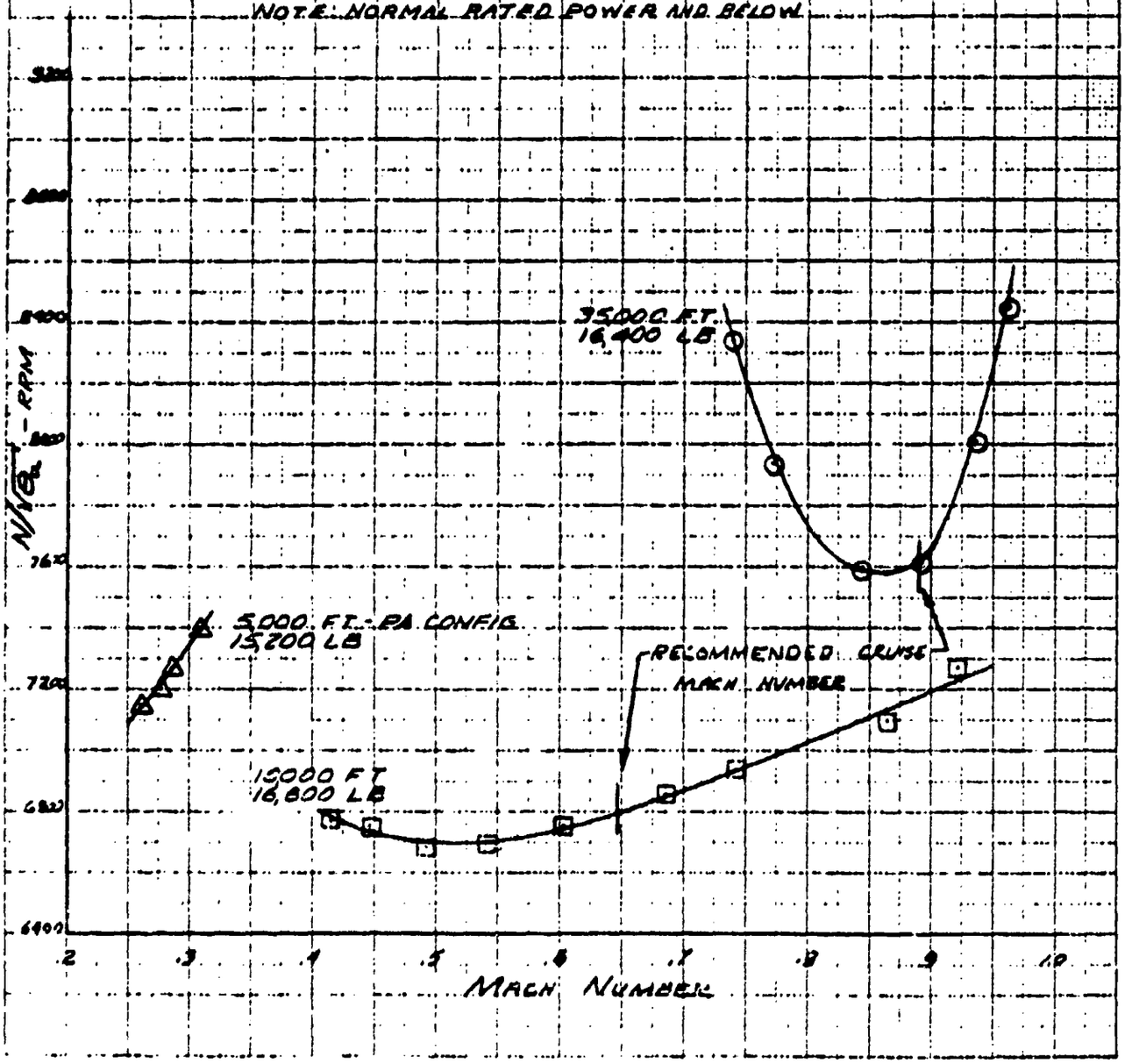
FIG. No. 41

POWER REQUIRED

F-101A S/N 55-2955

SYMBOL	ALTITUDE	GROSS WT.	CONFIG.
○	35,000	16,400	CRUISE
□	10,000	16,820	CRUISE
△	5000	15,200	PA.

NOTE: NORMAL RATED POWER AND BELOW



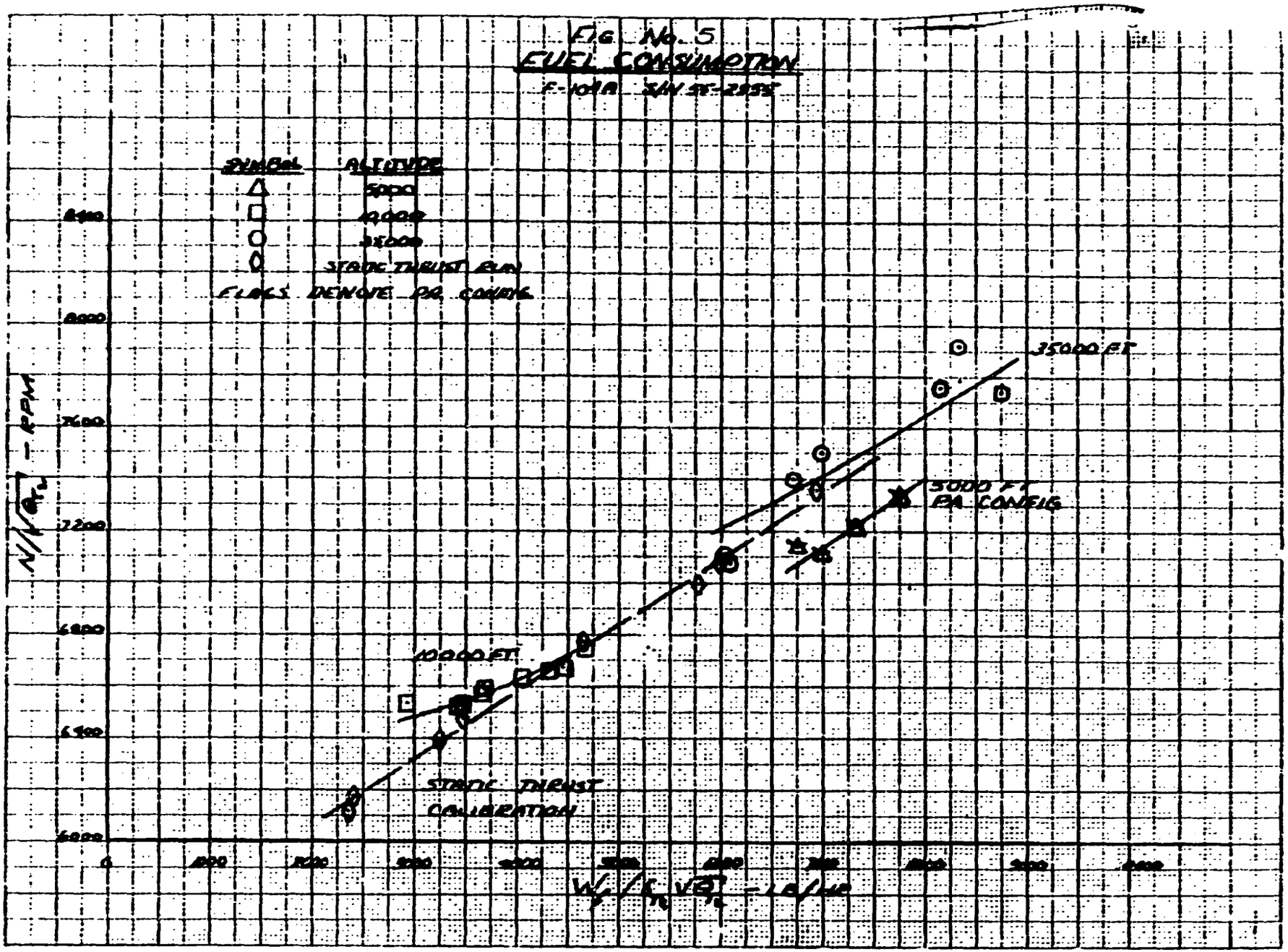


FIG. No. 5
FUEL CONSUMPTION
SYMBOL
ALTITUDE
5000
10000
35000
STATIC THRUST RUN
FLAGS DENOTE PA CONFIG
35000 FT
5000 FT
PA CONFIG
10000 FT
STATIC THRUST
CALIBRATION
8400
8000
7600
7200
6800
6400
6000
0
1000
2000
3000
4000
5000
6000
7000
8000
9000
10000
$N/\sqrt{\theta_{T_2}}$ - RPM
$W_f/\delta_{T_2}\sqrt{\theta_{T_2}}$ - LB/HR

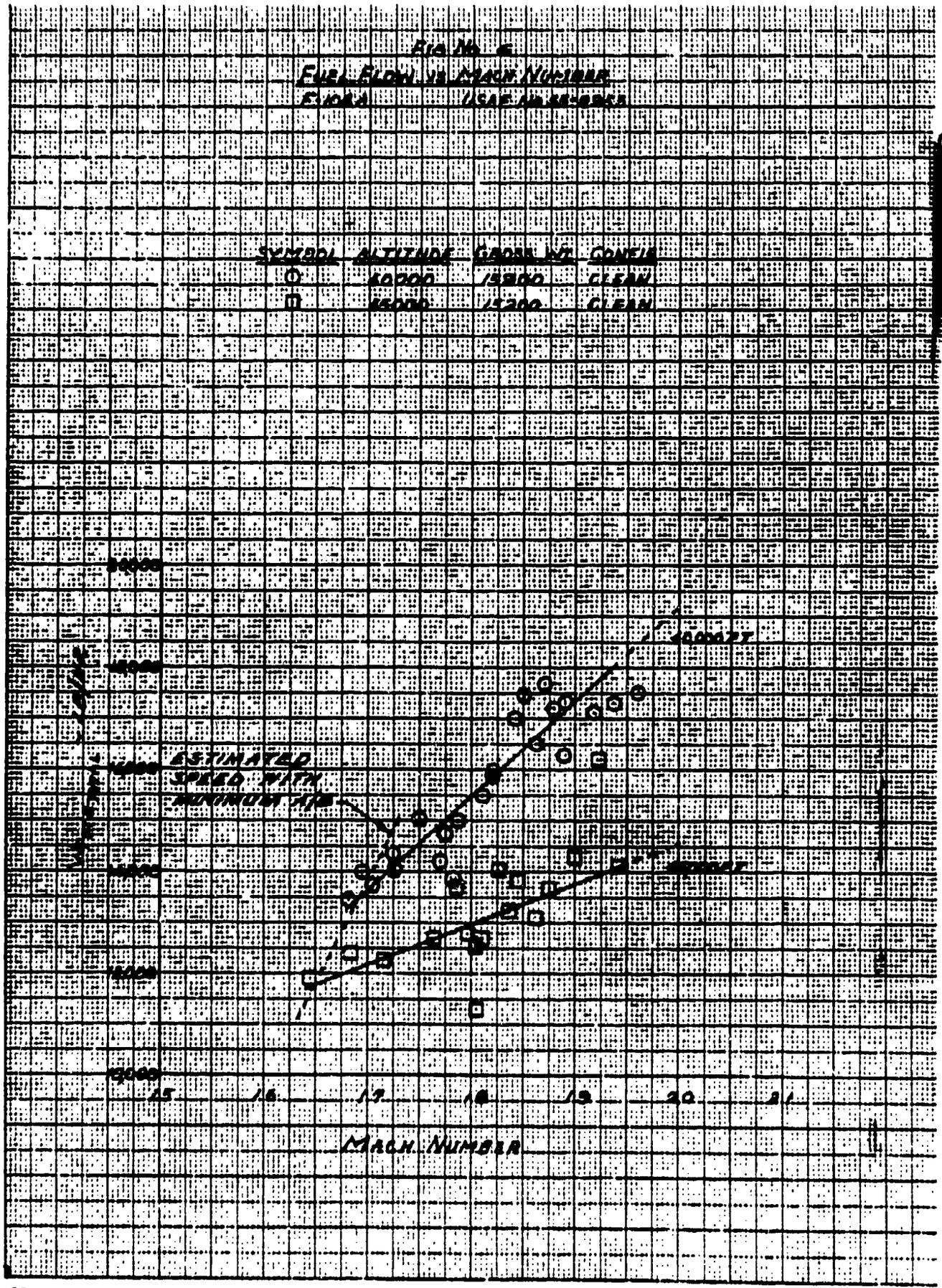


FIG. No. 6
FUEL FLOW VS MACH NUMBER
SYMBOL ALTITUDE GROSS WT. CONFIG
60000 15800 CLEAN
45000 15200 CLEAN
ESTIMATED SPEED WITH MINIMUM A/B
MACH NUMBER

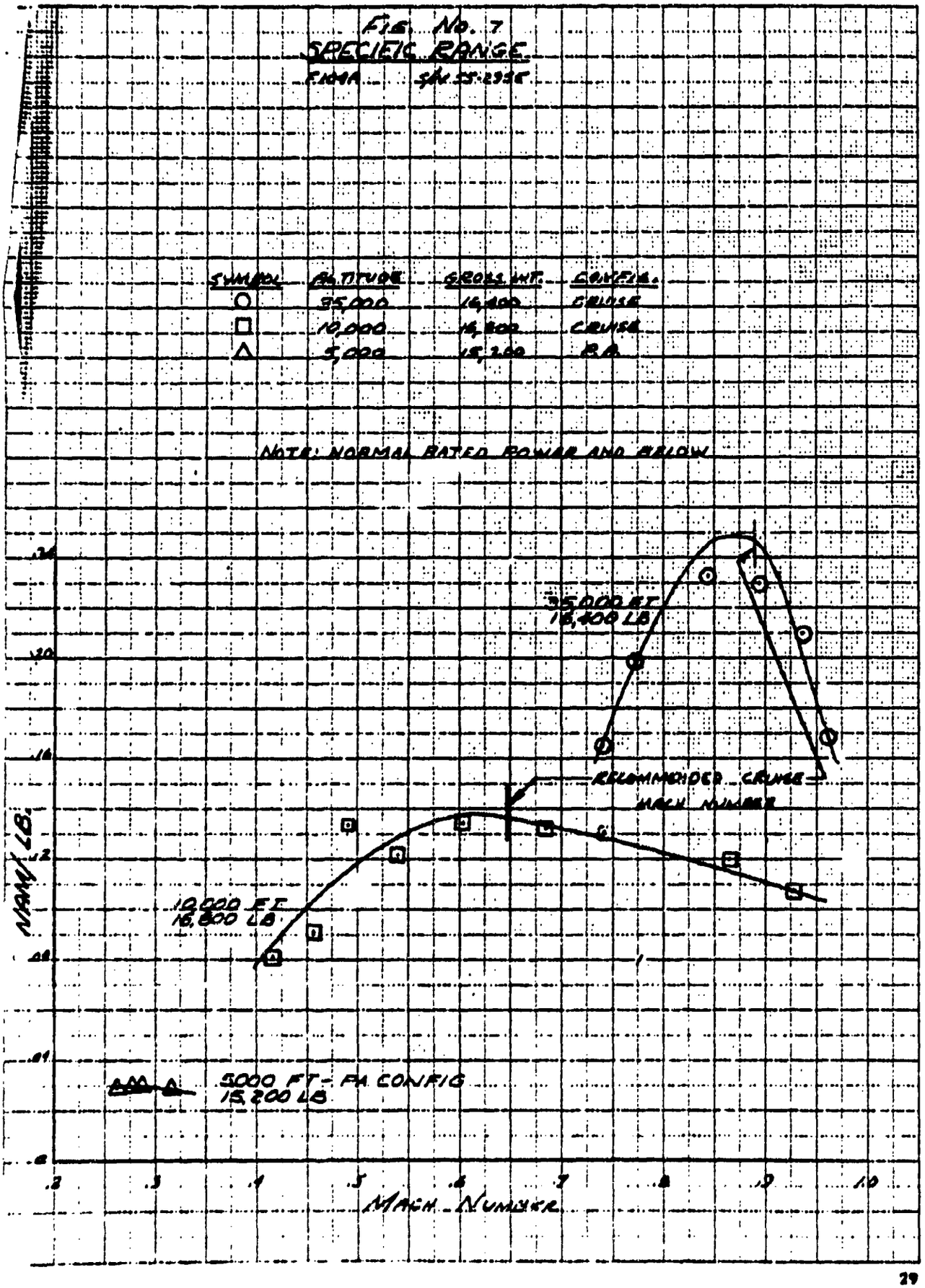


FIG. No. 7
SPECIFIC RANGE
F104A S/N 55-2956
SYMBOL ALTITUDE GROSS WT. CONFIG.
35,000 16,400 CRUISE
10,000 16,800 CRUISE
5,000 15,200 R.A.
NOTE: NORMAL RATED POWER AND BELOW
35,000 FT
16,400 LB
RECOMMENDED CRUISE MACH NUMBER
10,000 FT
16,800 LB
5000 FT - FA CONFIG
15,200 LB
NAM/LB.
.24
.20
.16
.12
.08
.04
.0
.2
.3
.4
.5
.6
.7
.8
.9
1.0
MACH NUMBER

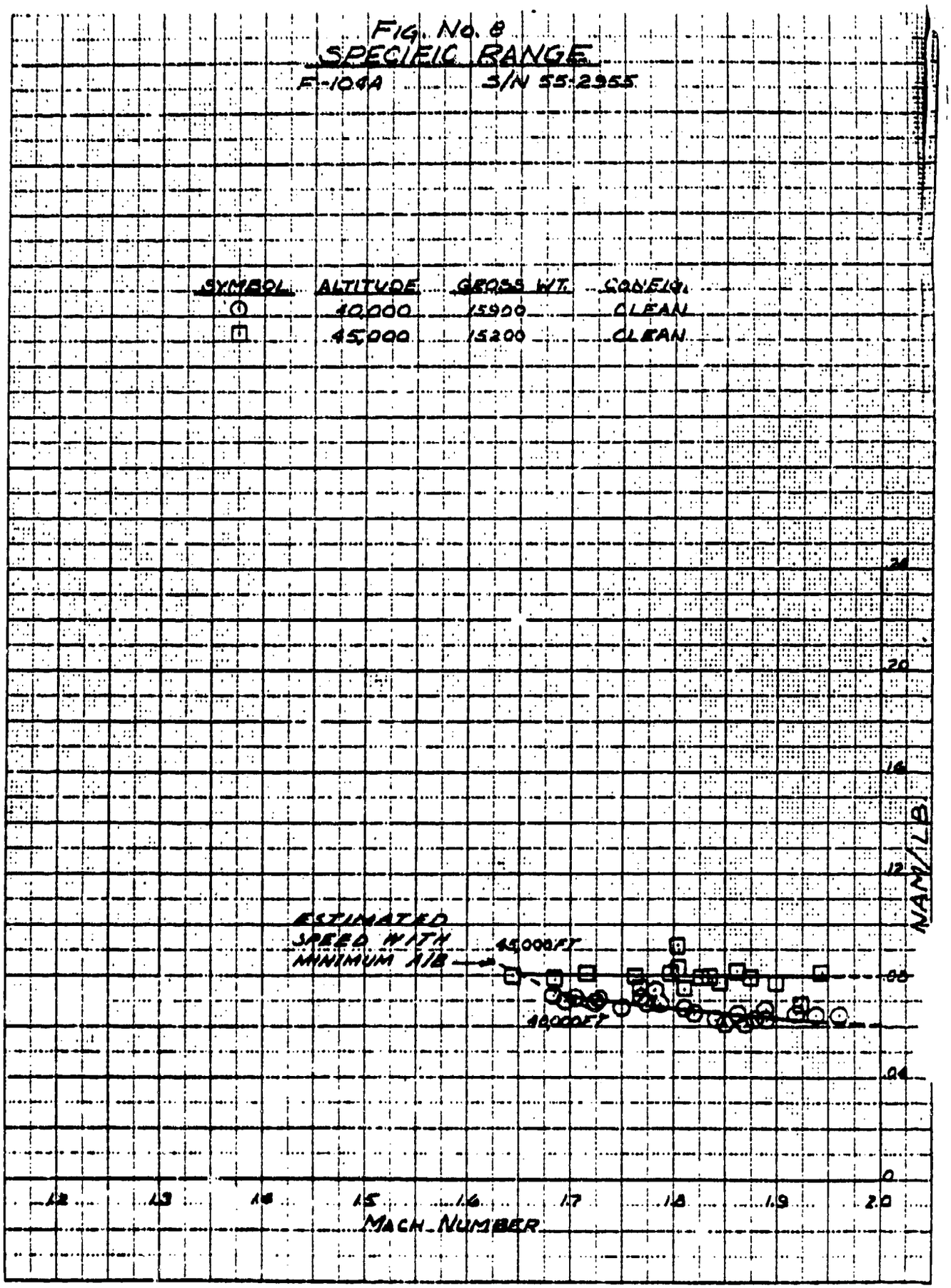


FIG. NO. 8
SPECIFIC RANGE
F-104A S/N 55-2955
SYMBOL ALTITUDE GROSS WT. CONFIG.
40,000 15900 CLEAN
45,000 15200 CLEAN
ESTIMATED SPEED WITH MINIMUM A/B
45,000 FT
40,000 FT
NAM/LB
.24
.20
.16
.12
.08
.04
0
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2.0
MACH NUMBER

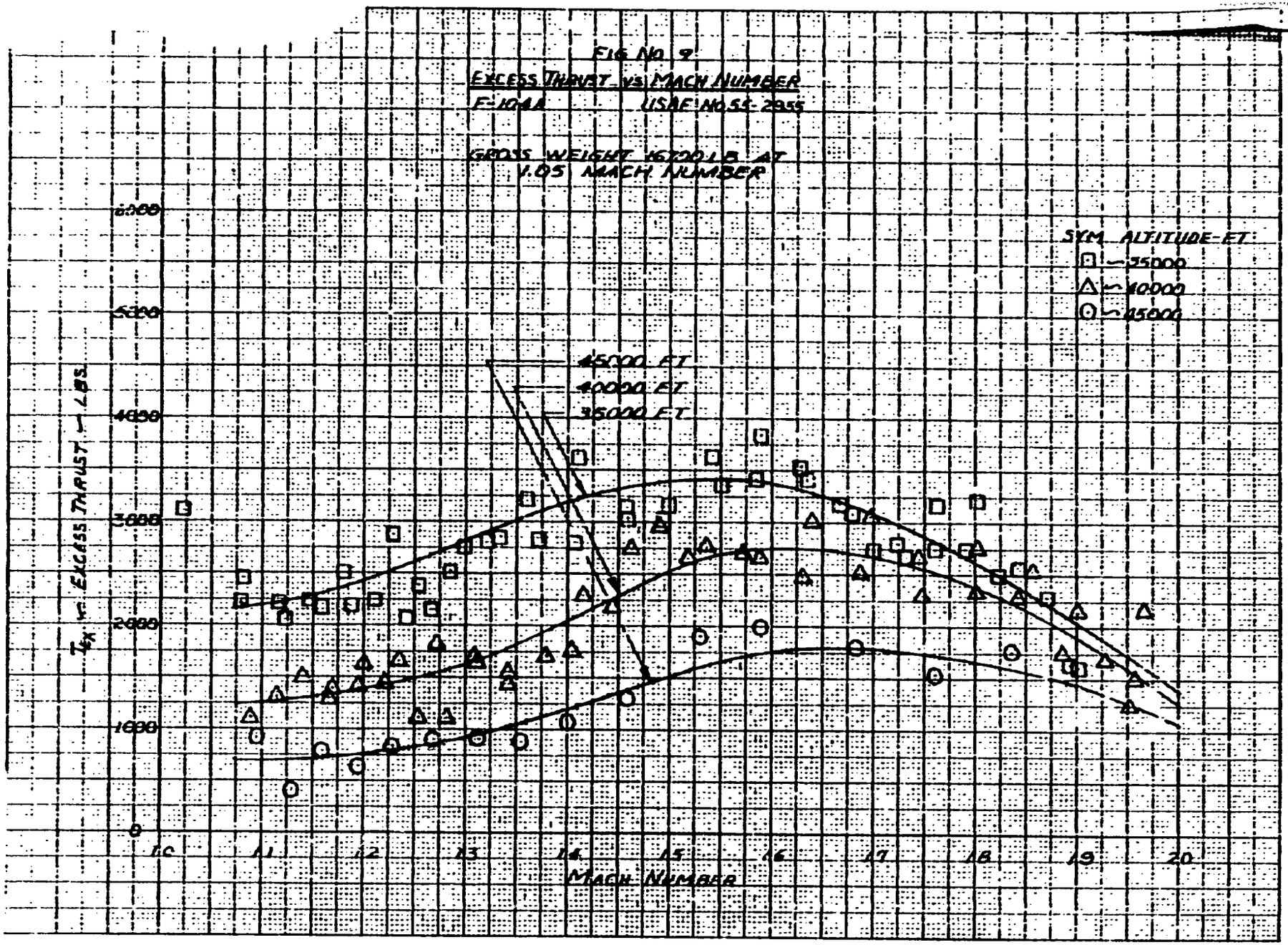


FIG NO 9
EXCESS THRUST vs MACH NUMBER
F-104A USAF NO 55-2955
GROSS WEIGHT 16700 LB AT
1.05 MACH NUMBER
SYM ALTITUDE-FT
□ ~ 35000
△ ~ 40000
○ ~ 45000
45000 FT
40000 FT
35000 FT
T_{EX} = EXCESS THRUST — LBS
6000
5000
4000
3000
2000
1000
0
1.0
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9
2.0
MACH NUMBER

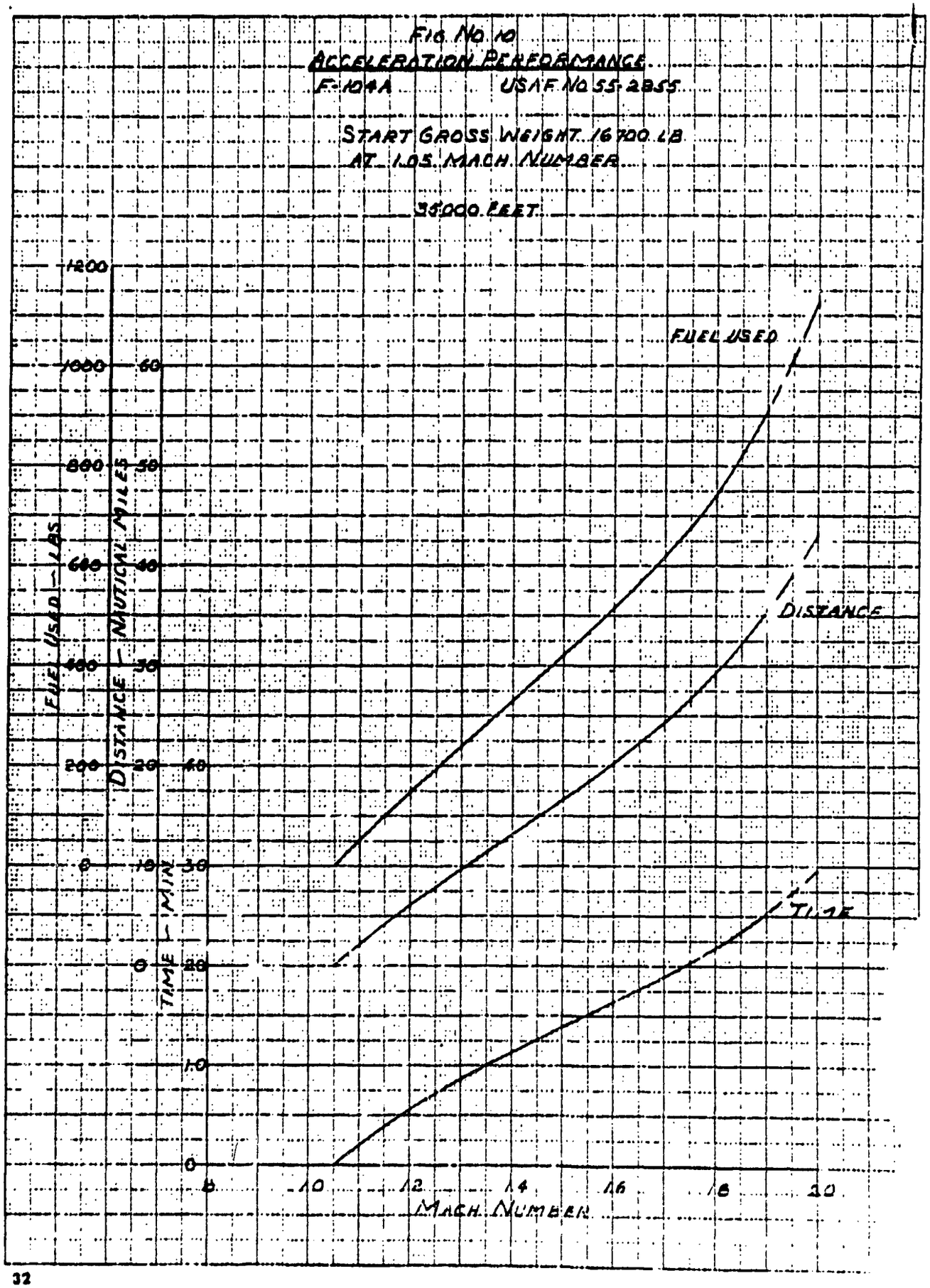


FIG NO 10
ACCELERATION PERFORMANCE
F-104A
USAF NO 55-2855
START GROSS WEIGHT 16700 LB
AT 1.05 MACH NUMBER
35000 FEET
FUEL USED
DISTANCE
TIME
FUEL USED — LBS
DISTANCE — NAUTICAL MILES
TIME — MIN
1200
1000
800
600
400
200
0
60
50
40
30
20
10
0
60
30
20
10
0
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

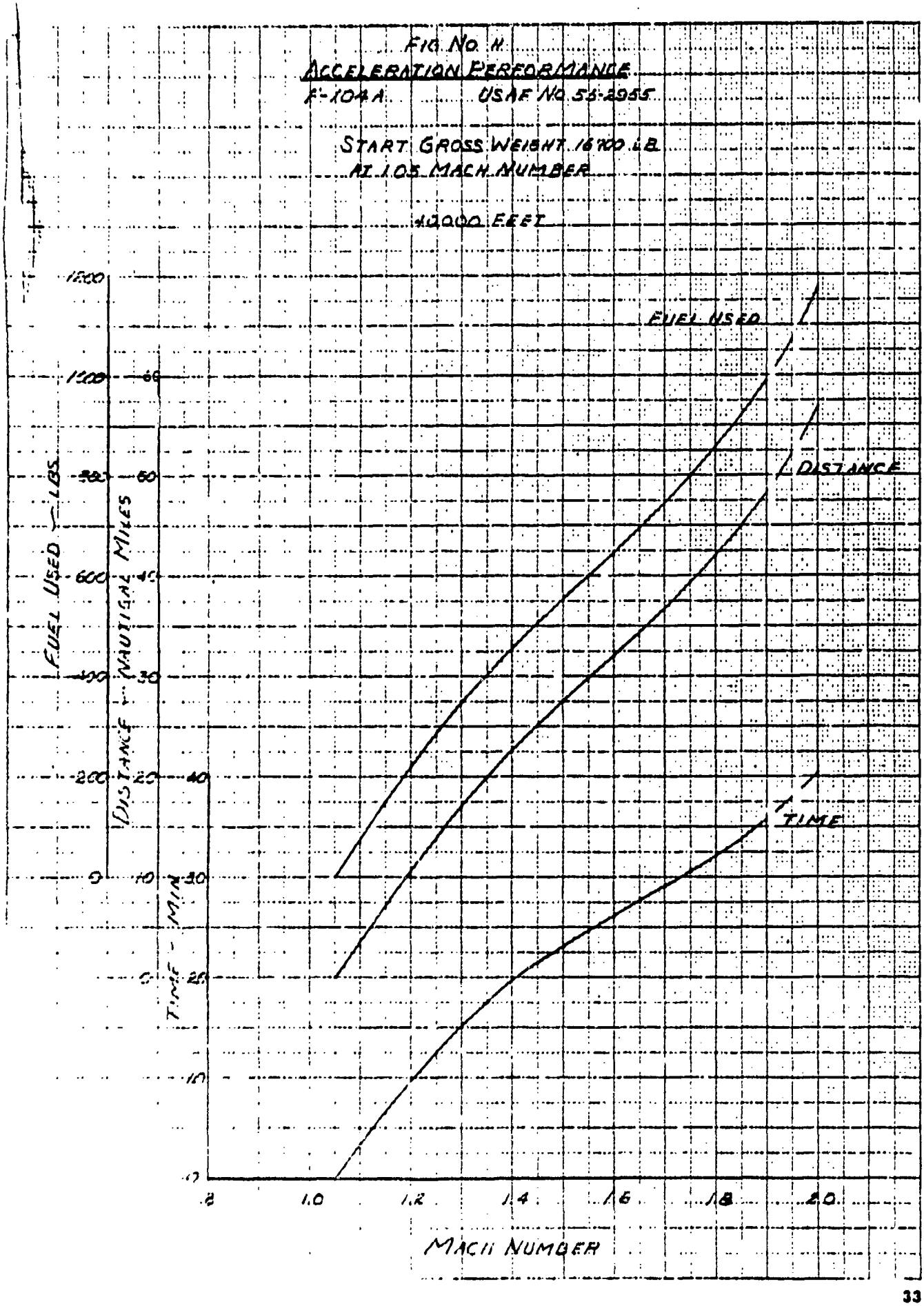


FIG. NO. 11
ACCELERATION PERFORMANCE
F-104A USAF NO. 56-2955
START GROSS WEIGHT 16700 LB.
AT 1.05 MACH NUMBER
40000 FEET
FUEL USED
DISTANCE
TIME
FUEL USED — LBS.
DISTANCE — NAUTICAL MILES
TIME — MIN.
1200
1000
800
600
400
200
0
60
50
40
30
20
10
0
40
30
20
10
0
.8
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER

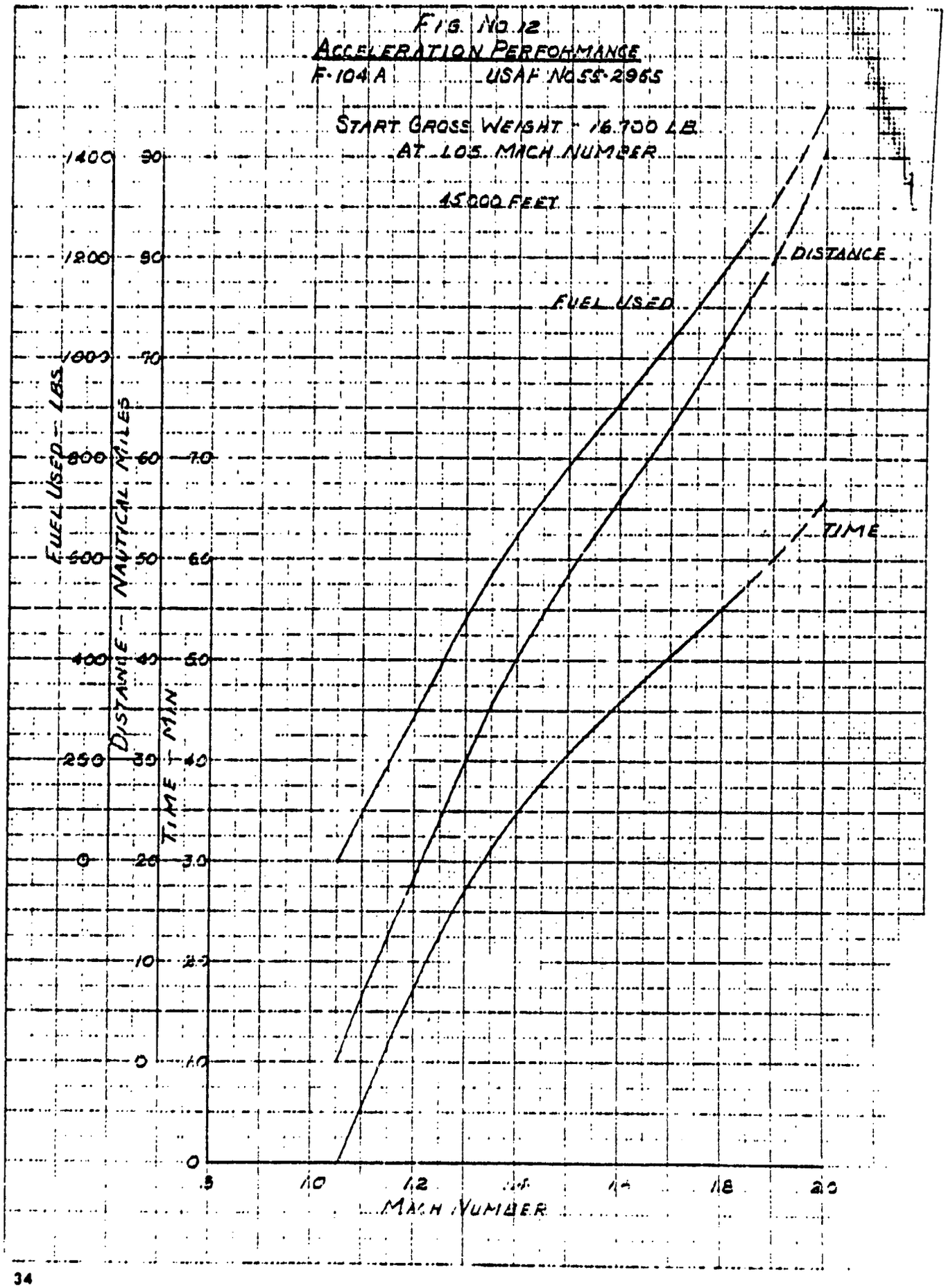


FIG. NO. 12
ACCELERATION PERFORMANCE
F-104A USAF NO. 55-2965
START GROSS WEIGHT - 16,700 LB.
AT LOS MACH NUMBER
45,000 FEET
DISTANCE
FUEL USED
TIME
FUEL USED - LBS.
DISTANCE - NAUTICAL MILES
TIME - MIN.
MACH NUMBER
1400
1200
1000
800
600
400
250
0
90
80
70
60
50
40
30
20
10
0
7.0
6.0
5.0
4.0
3.0
2.0
1.0
0
.8
1.0
1.2
1.4
1.6
1.8
2.0

FIG. No. 13

MAXIMUM STEADY TURN CAPABILITY

F-104A USAF 55-2955

16,000 LB

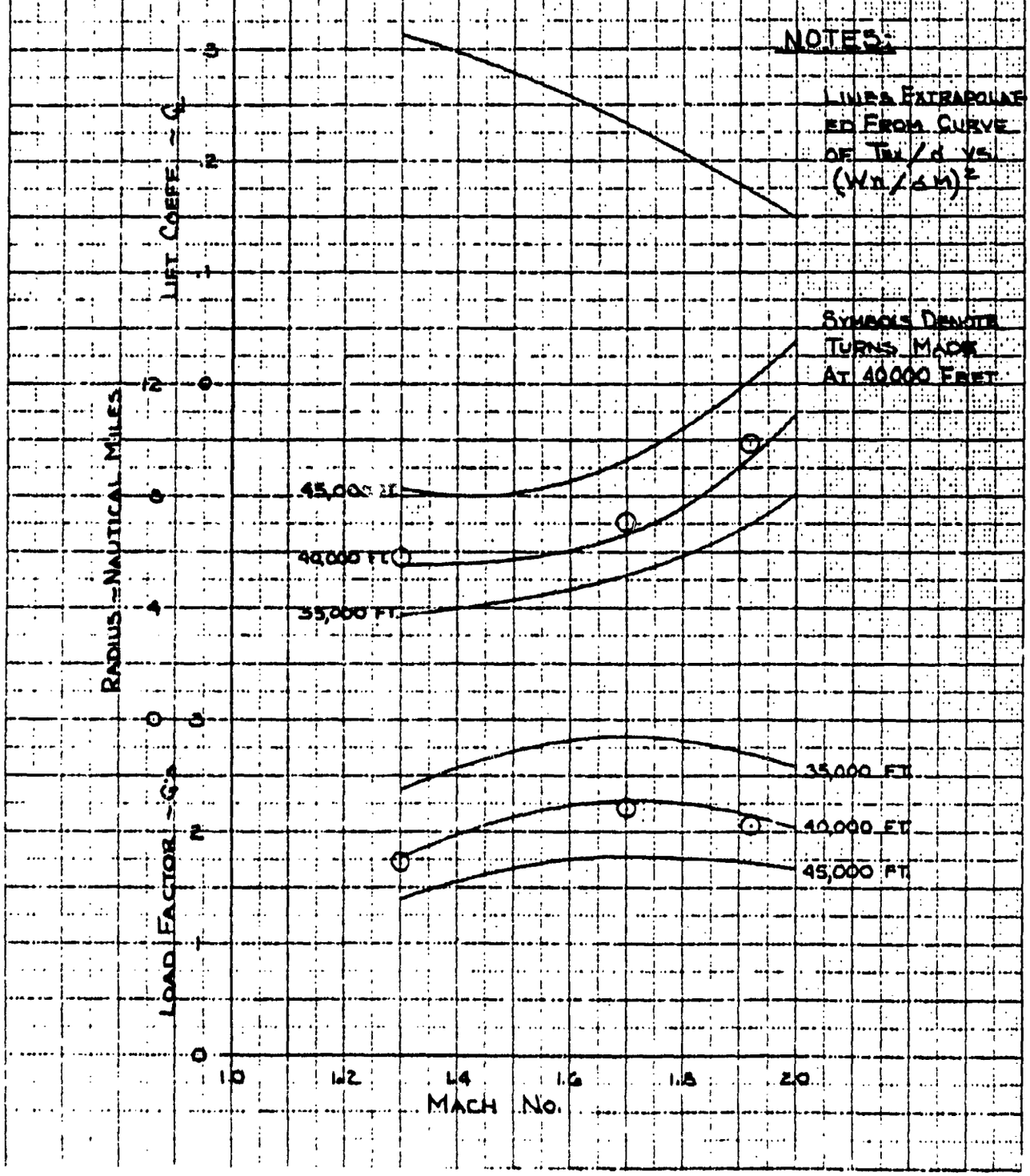


FIG NO 1.4

DECELERATING TURN CAPABILITY

F-104A USAF NO. 55-2955

NOTE:

DATA EXTRAPOLATED FROM CURVE OF $\frac{T_{ES}}{\delta}$ VS $\left(\frac{WG}{\delta M}\right)^2$

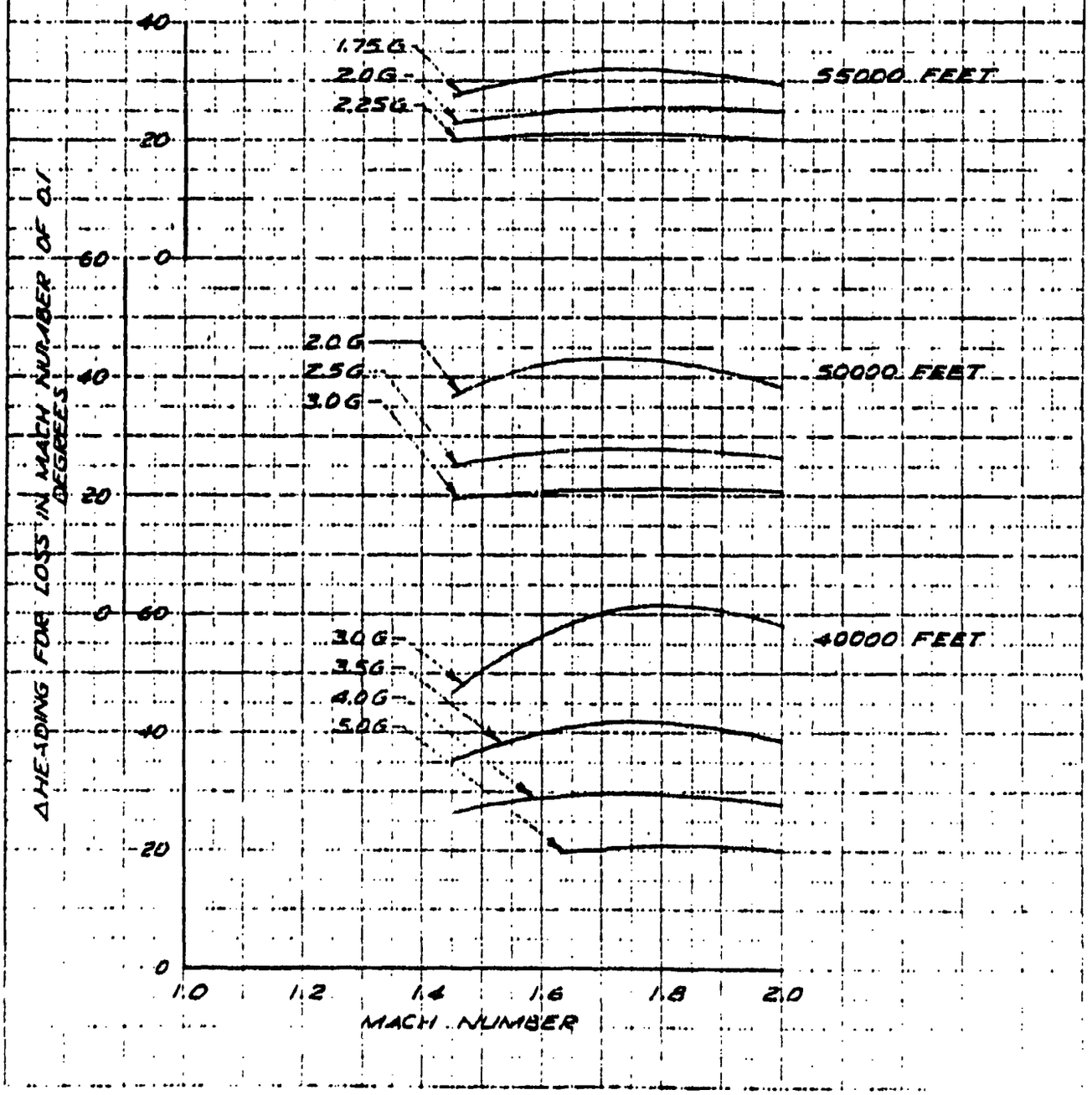
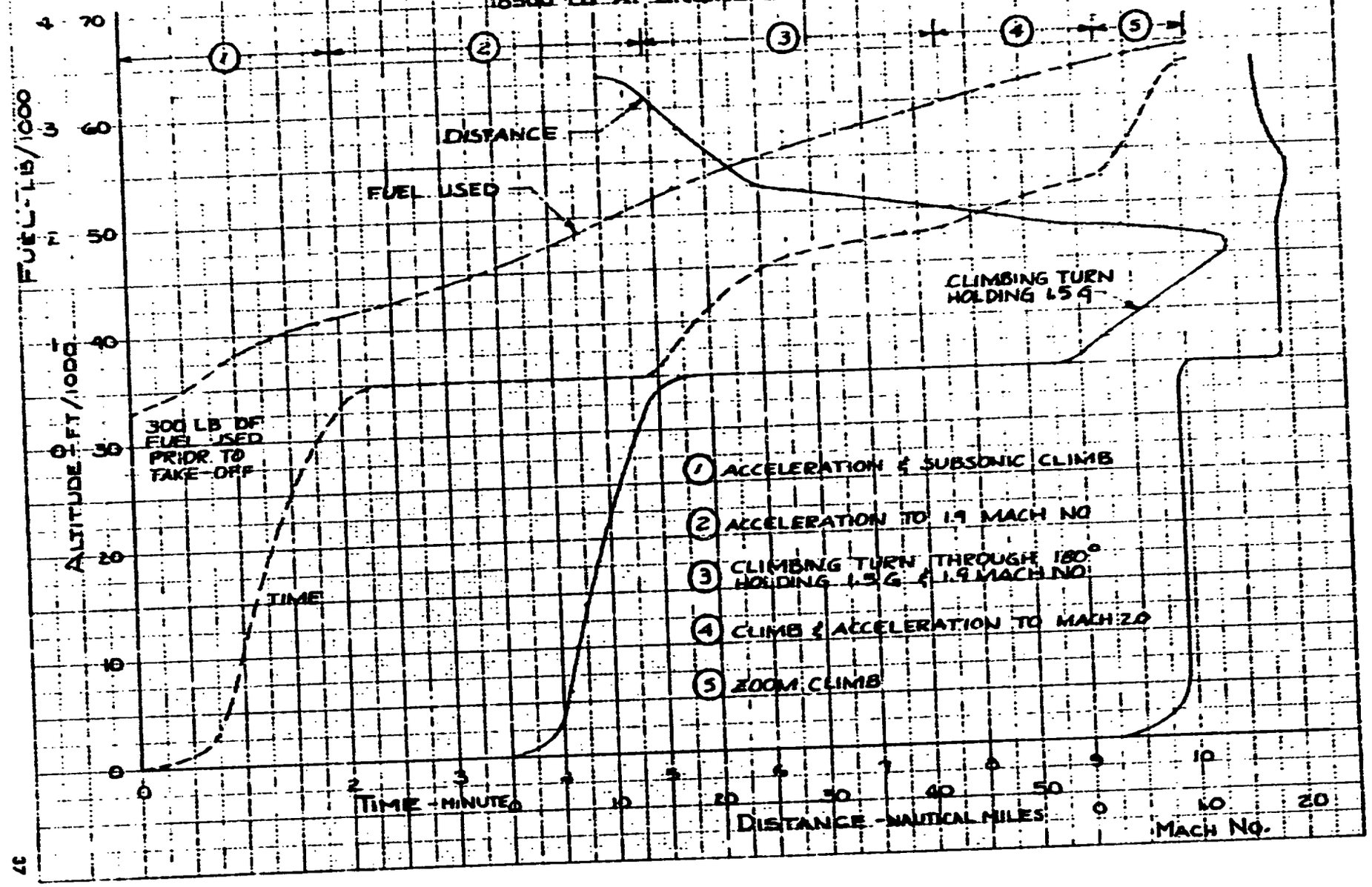


FIG No. 15

SIMULATED INTERCEPT MISSION

F-104A USAF NO 55-2955

18500 LB AT ENGINE START



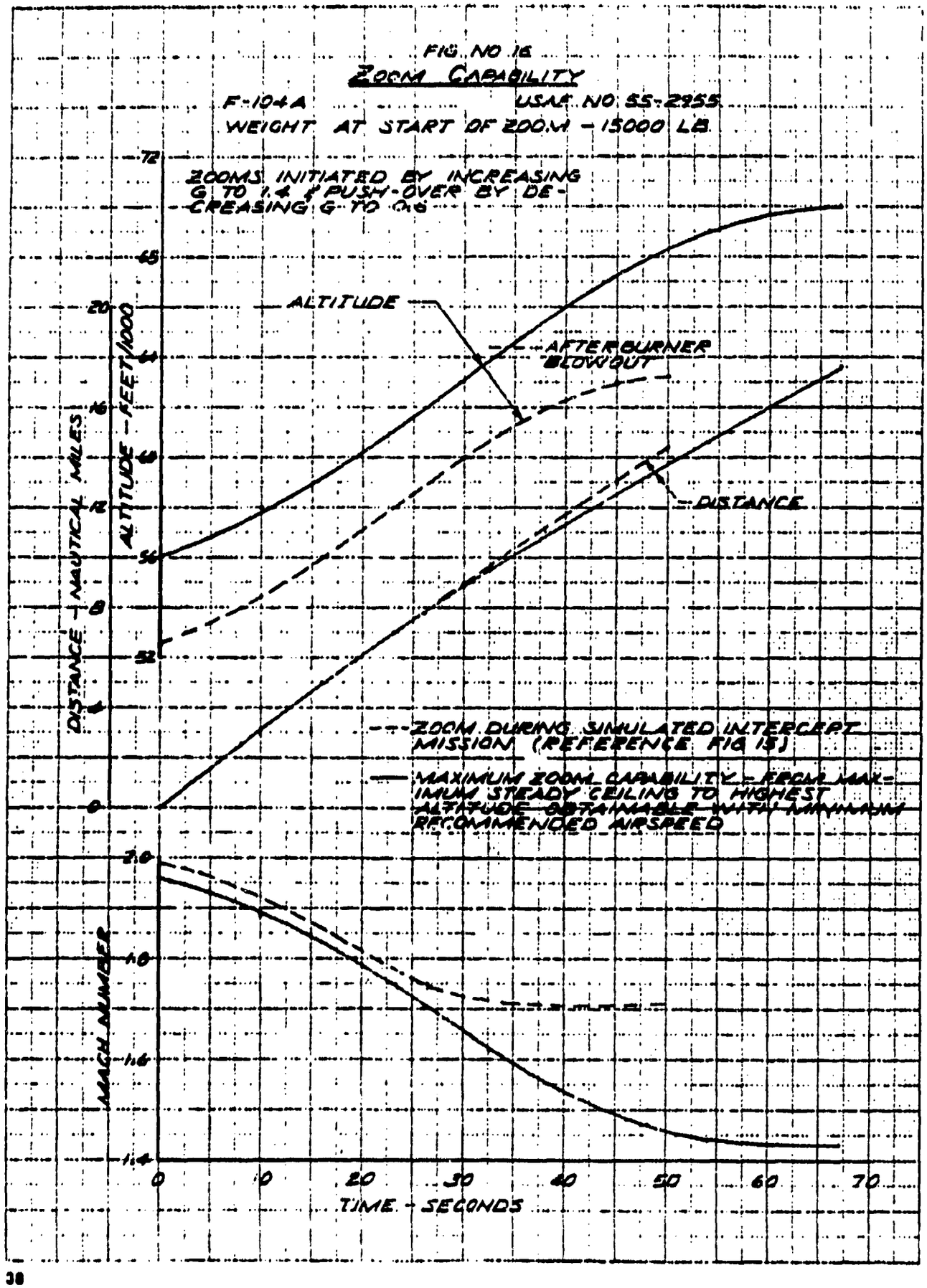


FIG. NO 16
ZOOM CAPABILITY
F-104A
USAF NO. 55-2955
WEIGHT AT START OF ZOOM - 15000 LB
ZOOMS INITIATED BY INCREASING G TO 1.4 & PUSH-OVER BY DECREASING G TO 0.6
ALTITUDE
AFTERBURNER BLOWOUT
DISTANCE
ZOOM DURING SIMULATED INTERCEPT MISSION (REFERENCE FIG 15)
MAXIMUM ZOOM CAPABILITY - FROM MAXIMUM STEADY CEILING TO HIGHEST ALTITUDE OBTAINABLE WITH MINIMUM RECOMMENDED AIRSPEED
DISTANCE - NAUTICAL MILES
ALTITUDE - FEET/1000
MACH NUMBER
TIME - SECONDS
72
65
20
16
12
8
4
0
56
52
2.0
1.8
1.6
1.4
0
10
20
30
40
50
60
70

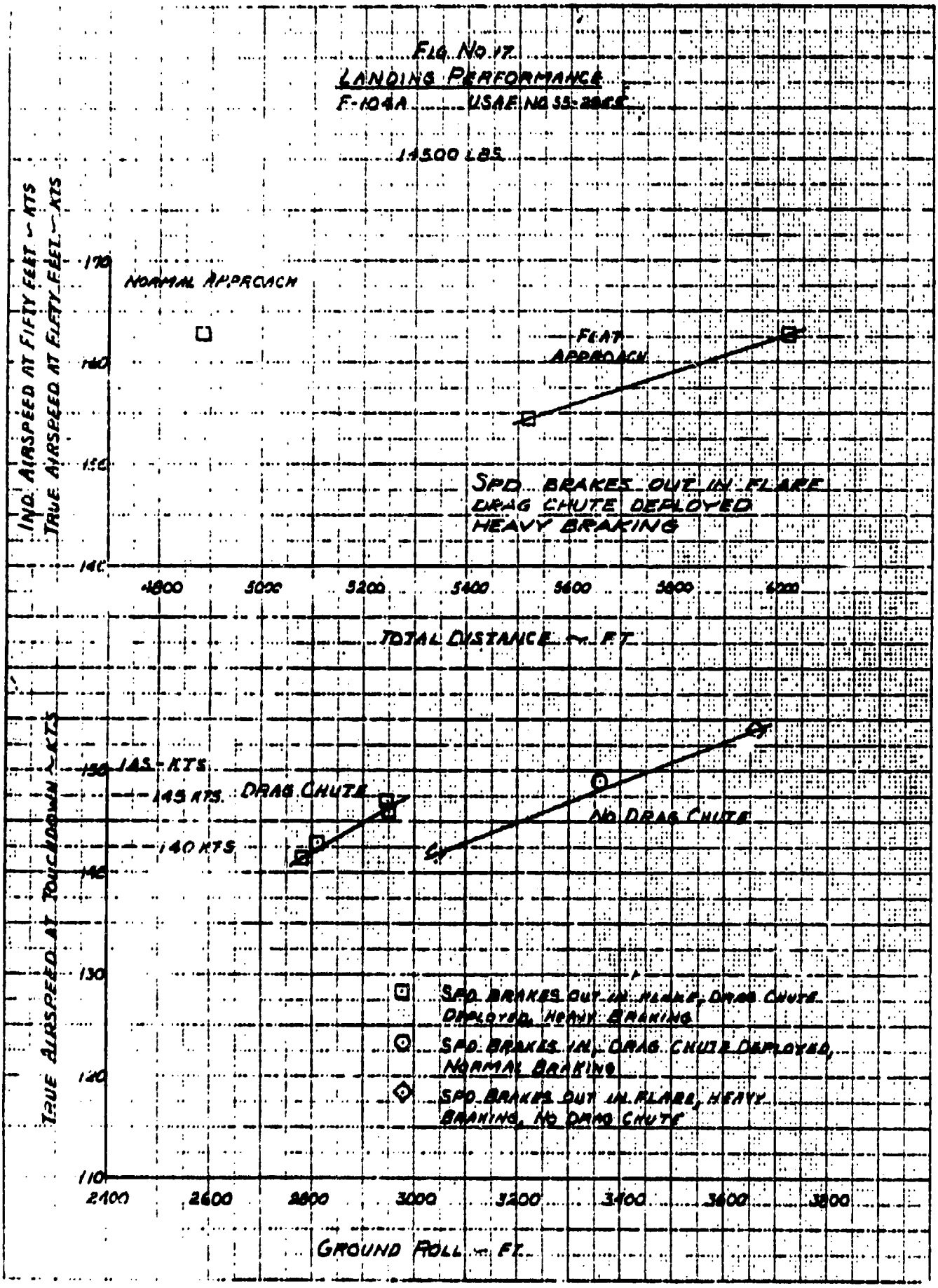


FIG. NO. 17
LANDING PERFORMANCE
F-104A USAF NO 55-2866
14500 LBS
IND. AIRSPEED AT FIFTY FEET ~ KTS
TRUE AIRSPEED AT FIFTY FEET ~ KTS
170
160
150
140
NORMAL APPROACH
FLAT APPROACH
SPD. BRAKES OUT IN FLARE
DRAG CHUTE DEPLOYED
HEAVY BRAKING
4800
5000
5200
5400
5600
5800
6000
TOTAL DISTANCE ~ FT.
TRUE AIRSPEED AT TOUCHDOWN ~ KTS
150
140
130
120
110
145 KTS
145 KTS. DRAG CHUTE
140 KTS
NO DRAG CHUTE
SPD. BRAKES OUT IN FLARE, DRAG CHUTE DEPLOYED, HEAVY BRAKING
SPD. BRAKES IN, DRAG CHUTE DEPLOYED, NORMAL BRAKING
SPD. BRAKES OUT IN FLARE, HEAVY BRAKING, NO DRAG CHUTE
2400
2600
2800
3000
3200
3400
3600
3800
GROUND ROLL ~ FT.

FIG NO 18

AFTERBURNER SLOWOUT BOUNDARY

F-104A USAF NO 55-2955

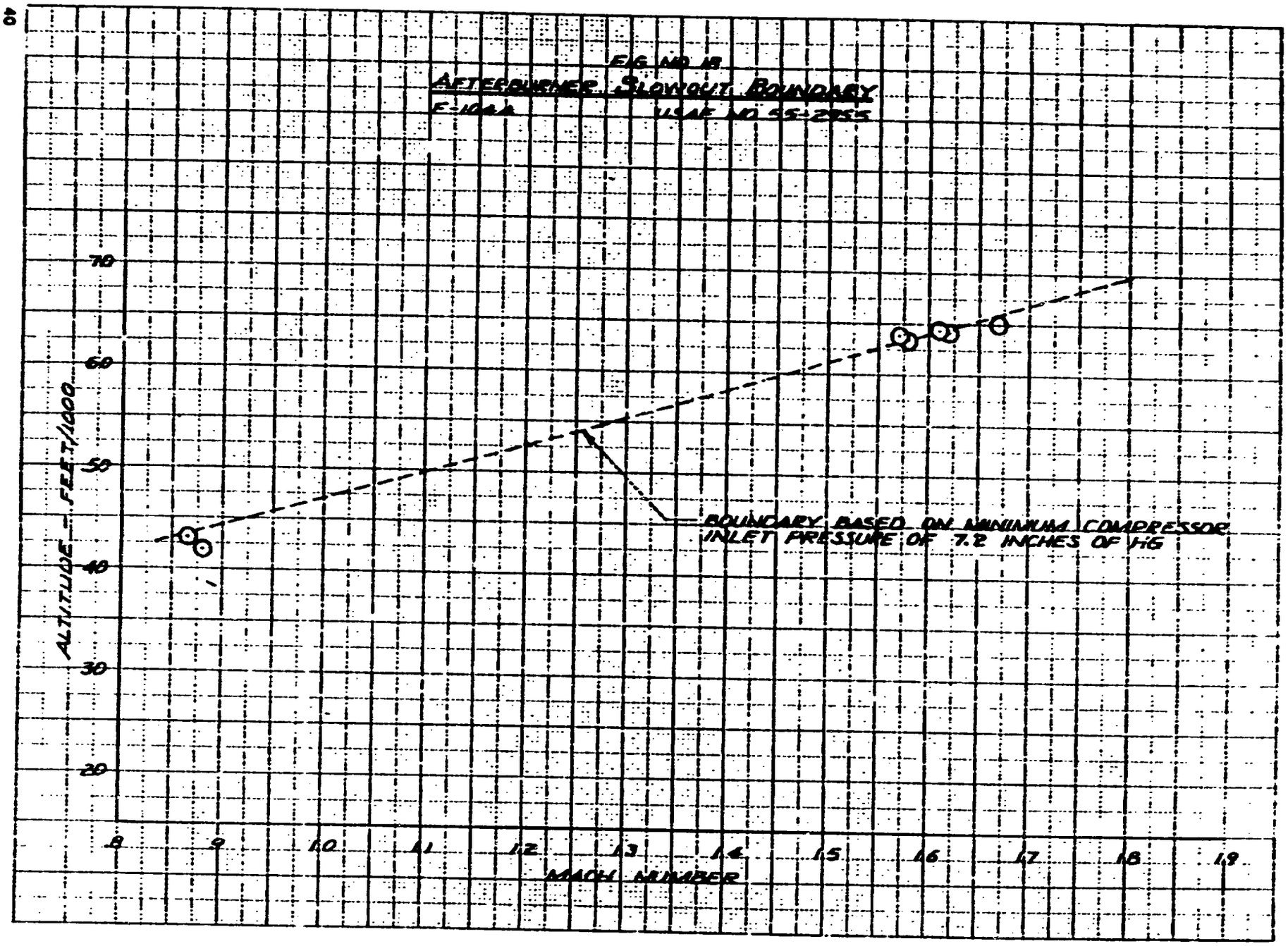


FIG. No. 19

ENGINE INDUCTION SYSTEM PERFORMANCE

F-104A USAF No. 55-2956

OVERALL TOTAL-PRESSURE RECOVERY

J79-GE-3 TURBOJET ENGINE

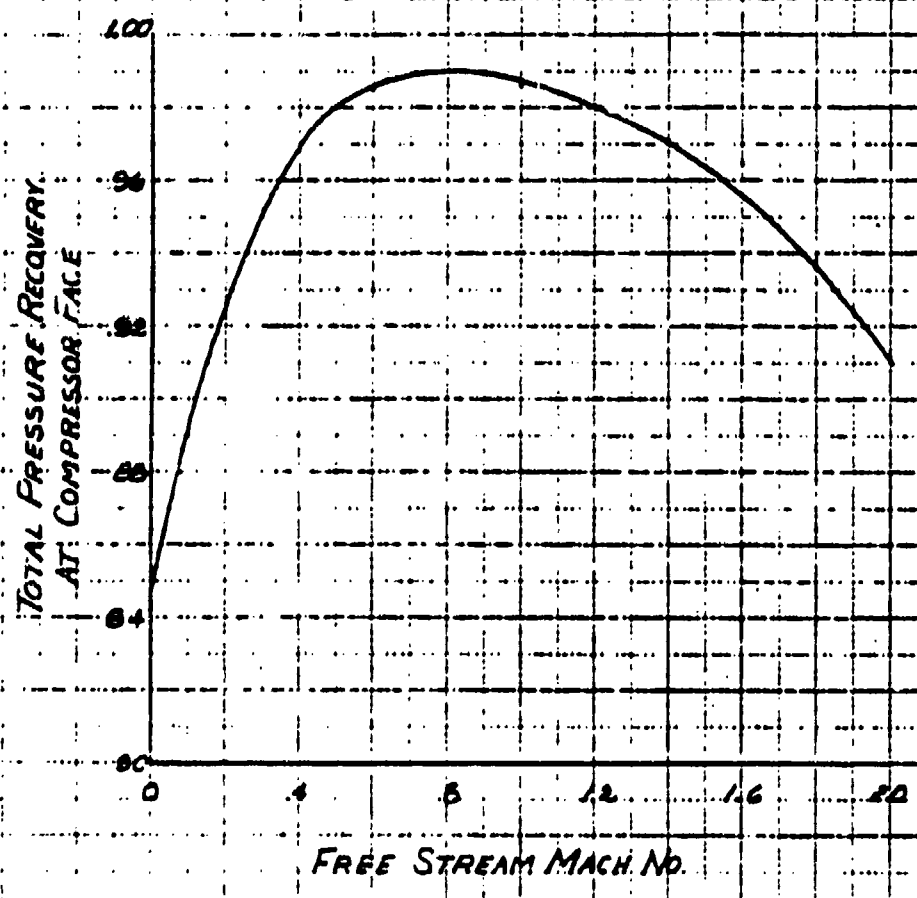
NACA STANDARD DAY

MAXIMUM POWER

SOURCE: LOCKHEED AIRCRAFT CORP

REPORT No: 10104

DATE: 8 APRIL 1955



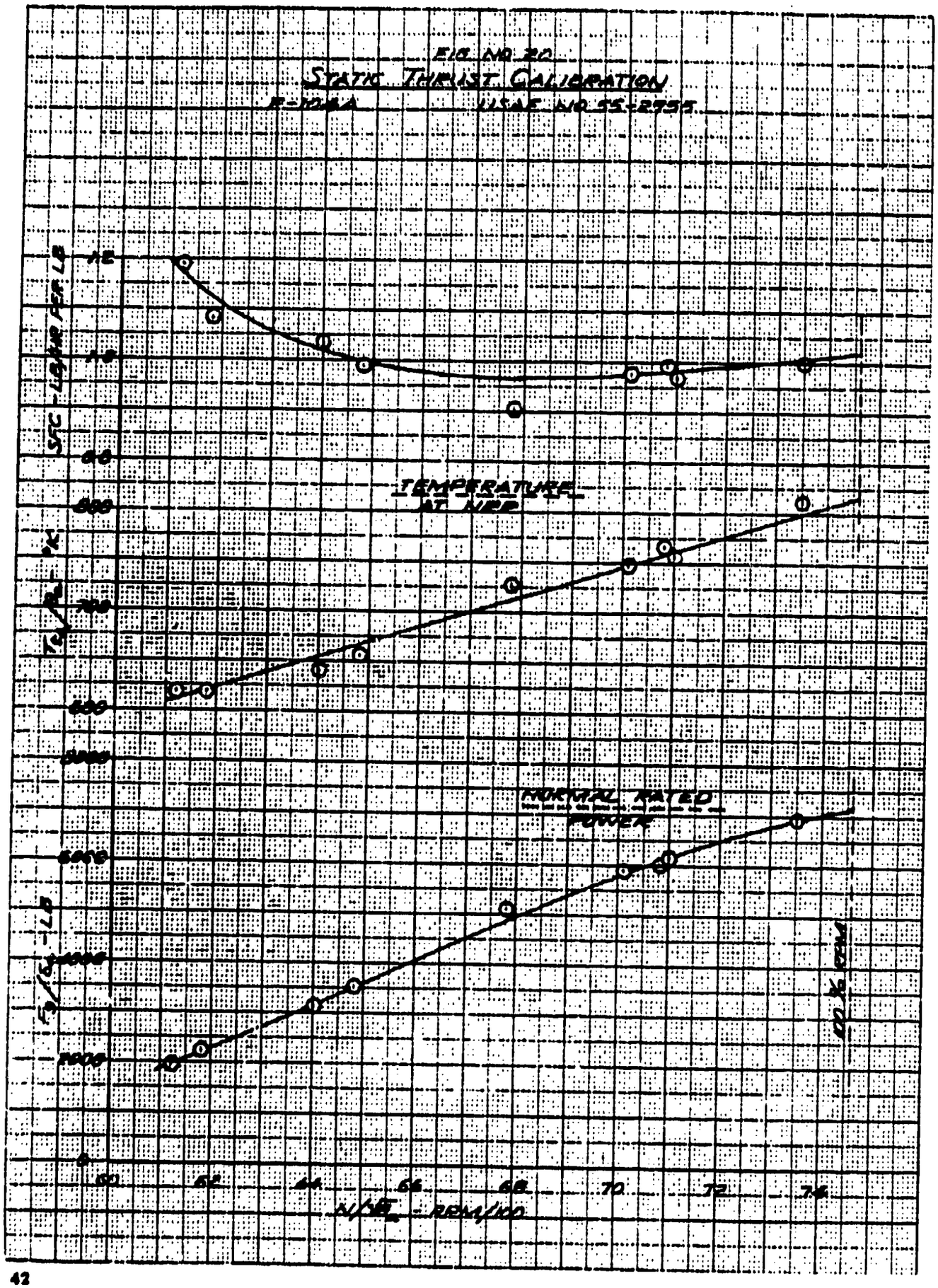


FIG NO 20
STATIC THRUST CALIBRATION
USAF NO 55-2955
SFC - LB/HR PER LB
TEMPERATURE AT NRP
NORMAL RATED POWER
100 % RPM
60
62
64
66
68
70
72
74
N/√θ - RPM/100

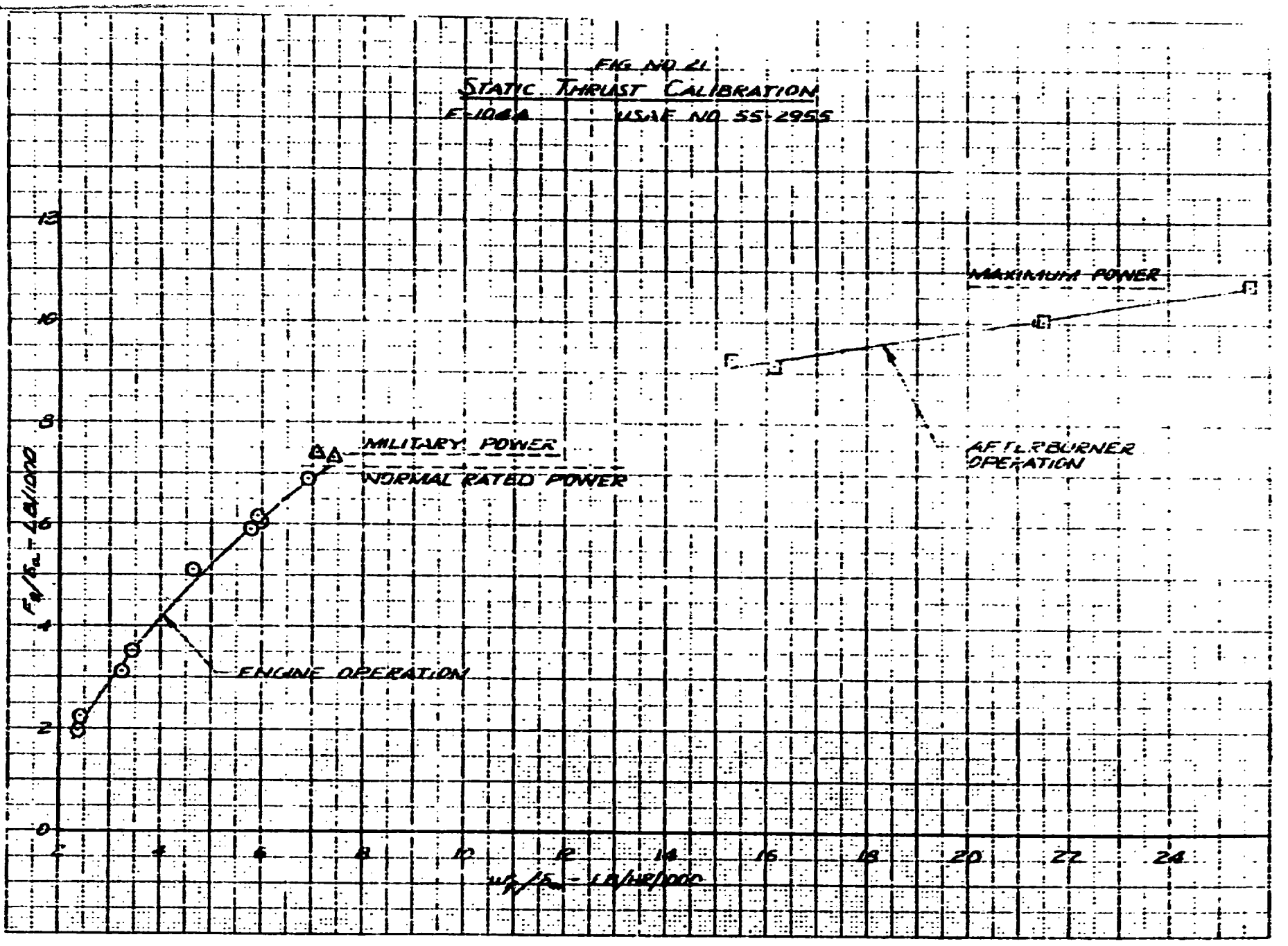
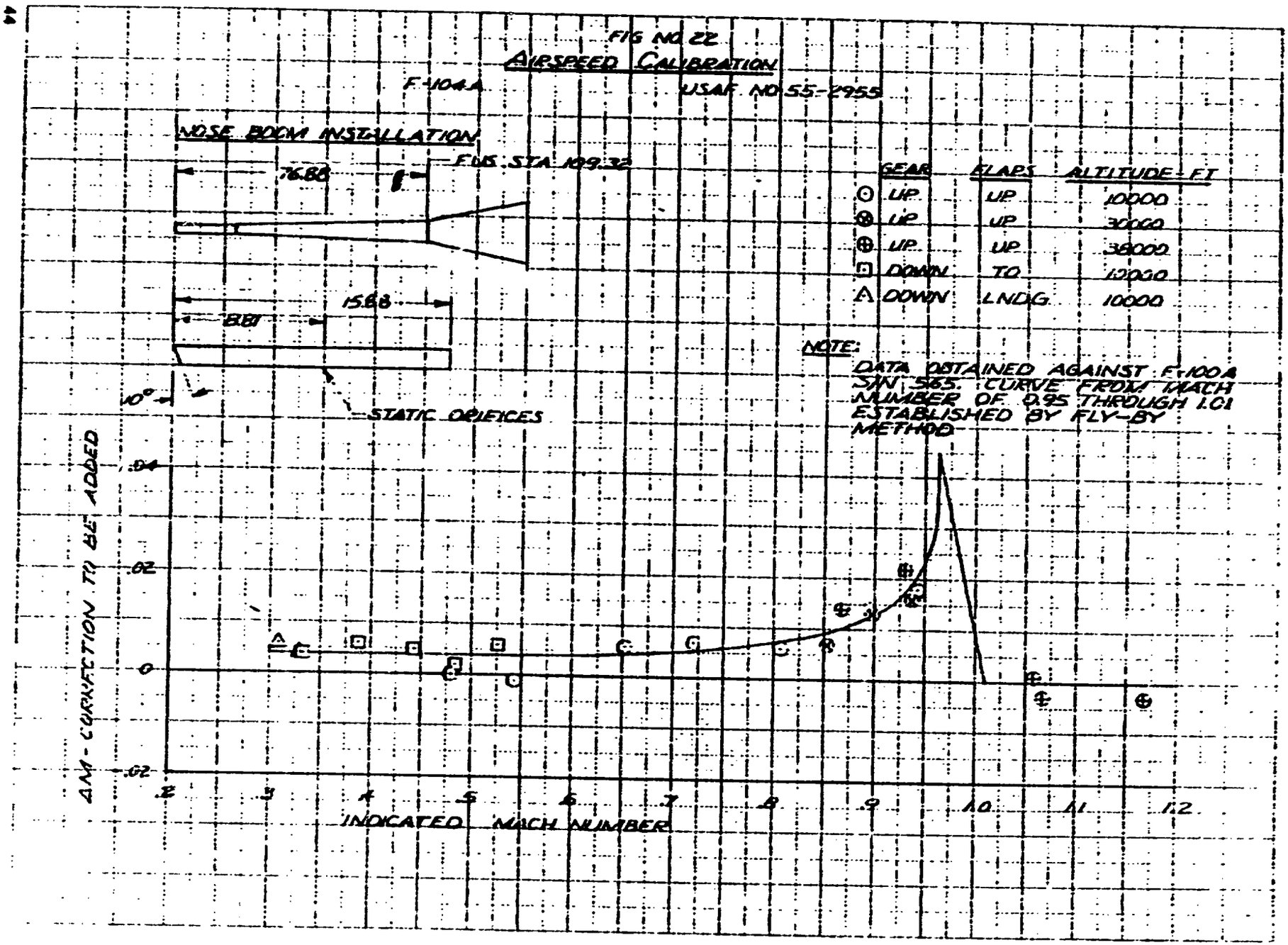


FIG NO 21
STATIC THRUST CALIBRATION
F-104A USAF NO 55-2955
MAXIMUM POWER
AFTERBURNER OPERATION
MILITARY POWER
NORMAL RATED POWER
ENGINE OPERATION
0
2
4
6
8
10
12
14
16
18
20
22
24

FIG NO 22

AIRSPEED CALIBRATION

F-104A USAF NO 55-2955



CONFIG. POWER APPROACH	FIGURE 23	DYNAMIC LONGITUDINAL STABILITY

TRIM CONDITIONS: CAS 180 KNOTS; CG 5.7 %MAC; STABILIZER 9.0 T.E. UP; ALTITUDE 10,000 FEET; WEIGHT 17,740 POUNDS; MACH NO. .328

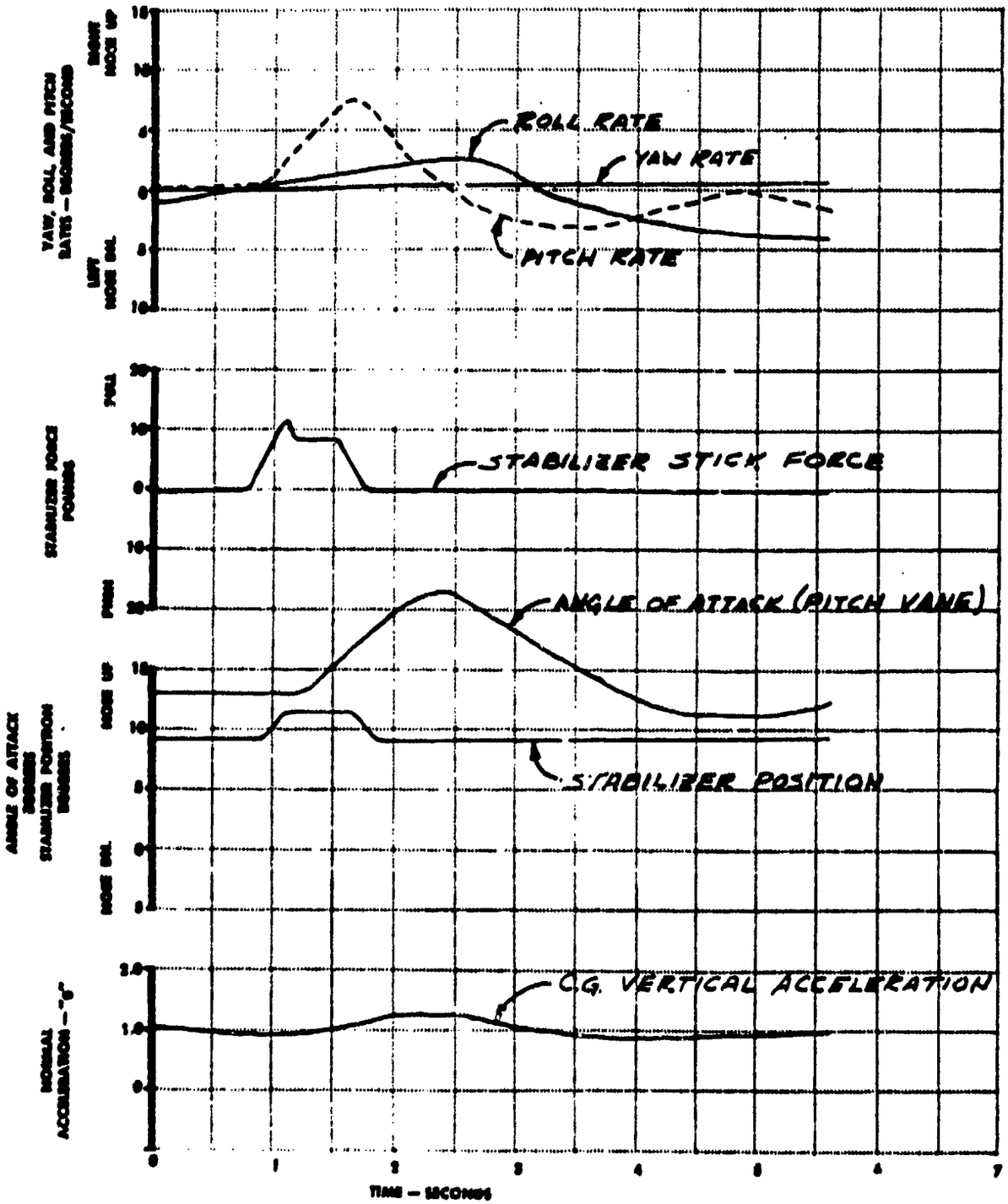
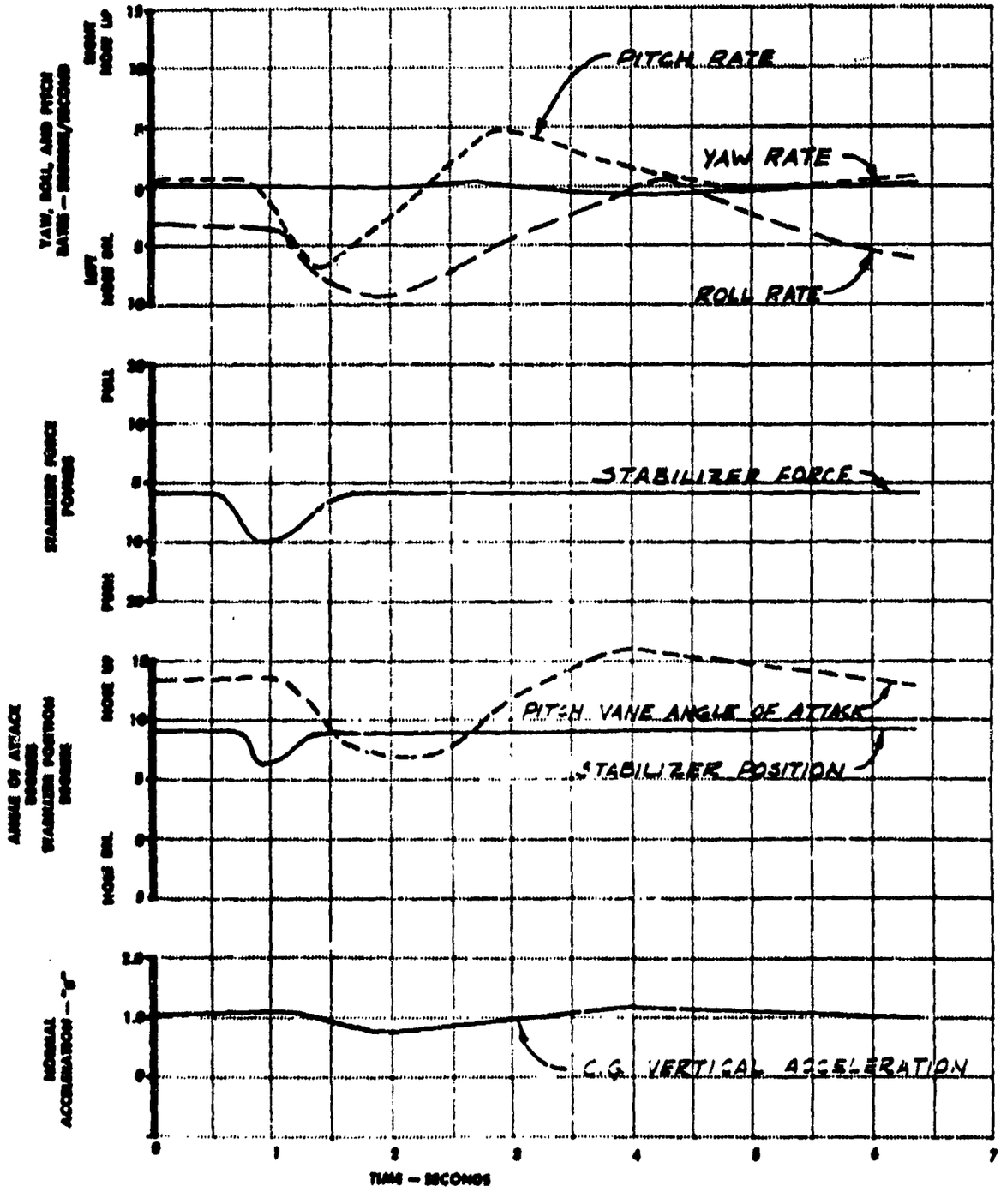


FIGURE 24	DYNAMIC LONGITUDINAL STABILITY

CONFIG. POWER APPROACH

TRIM CONDITIONS: CAS 180 KNOTS; CG 6.75 %MAC; STABILIZER 9.0 T.E. UP; ALTITUDE 8500 FEET; WEIGHT 17,240 POUNDS; MACH NO. .318



CONFIG. POWER APPROACH	FIGURE 25	DYNAMIC LONGITUDINAL STABILITY

TRIM CONDITIONS: CAS 180 KNOTS; CG 12.05 %MAC; STABILIZER 6.5 T.E. UP; ALTITUDE 10,000 FEET; WEIGHT 15,140 POUNDS; MACH NO. .328

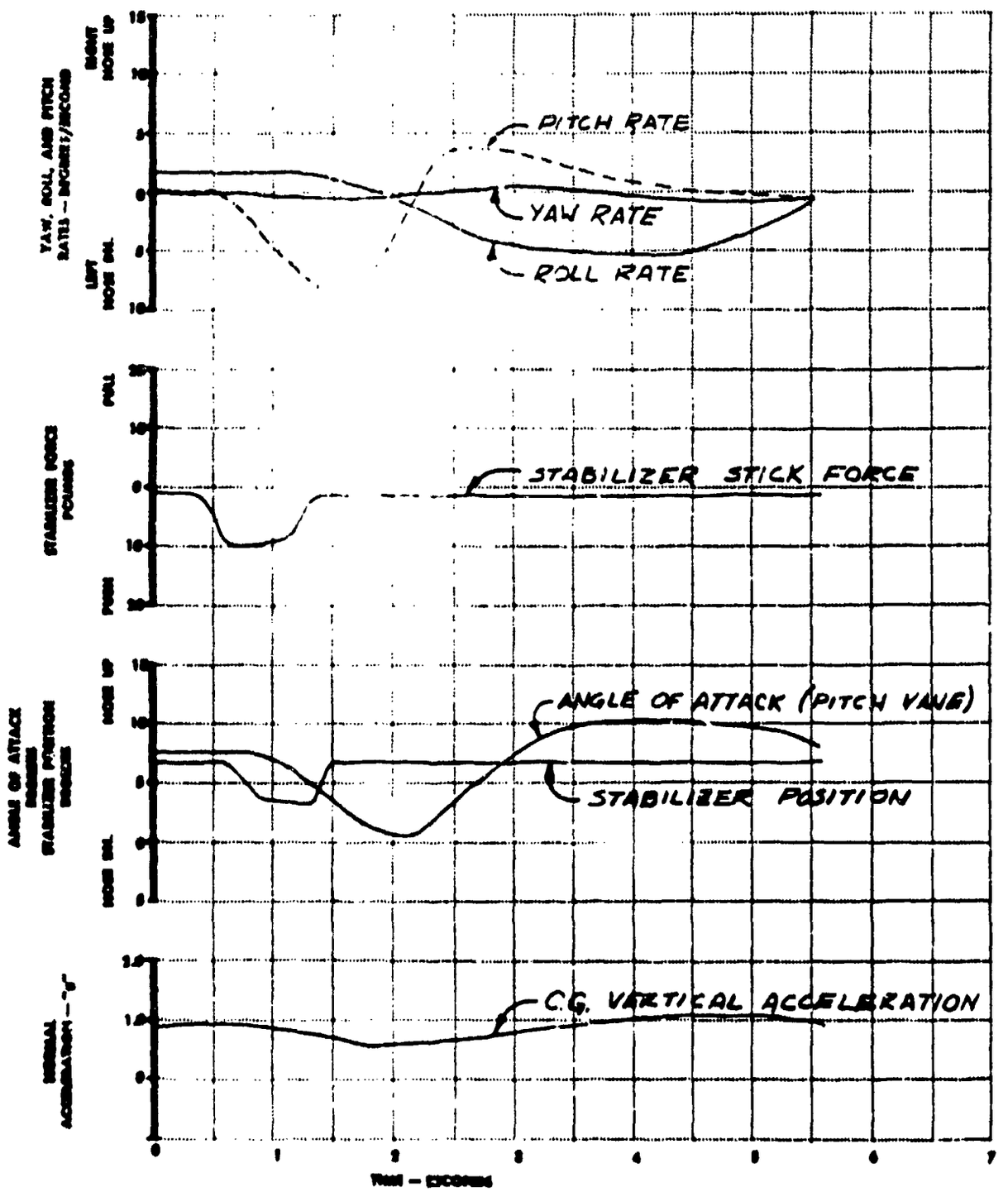
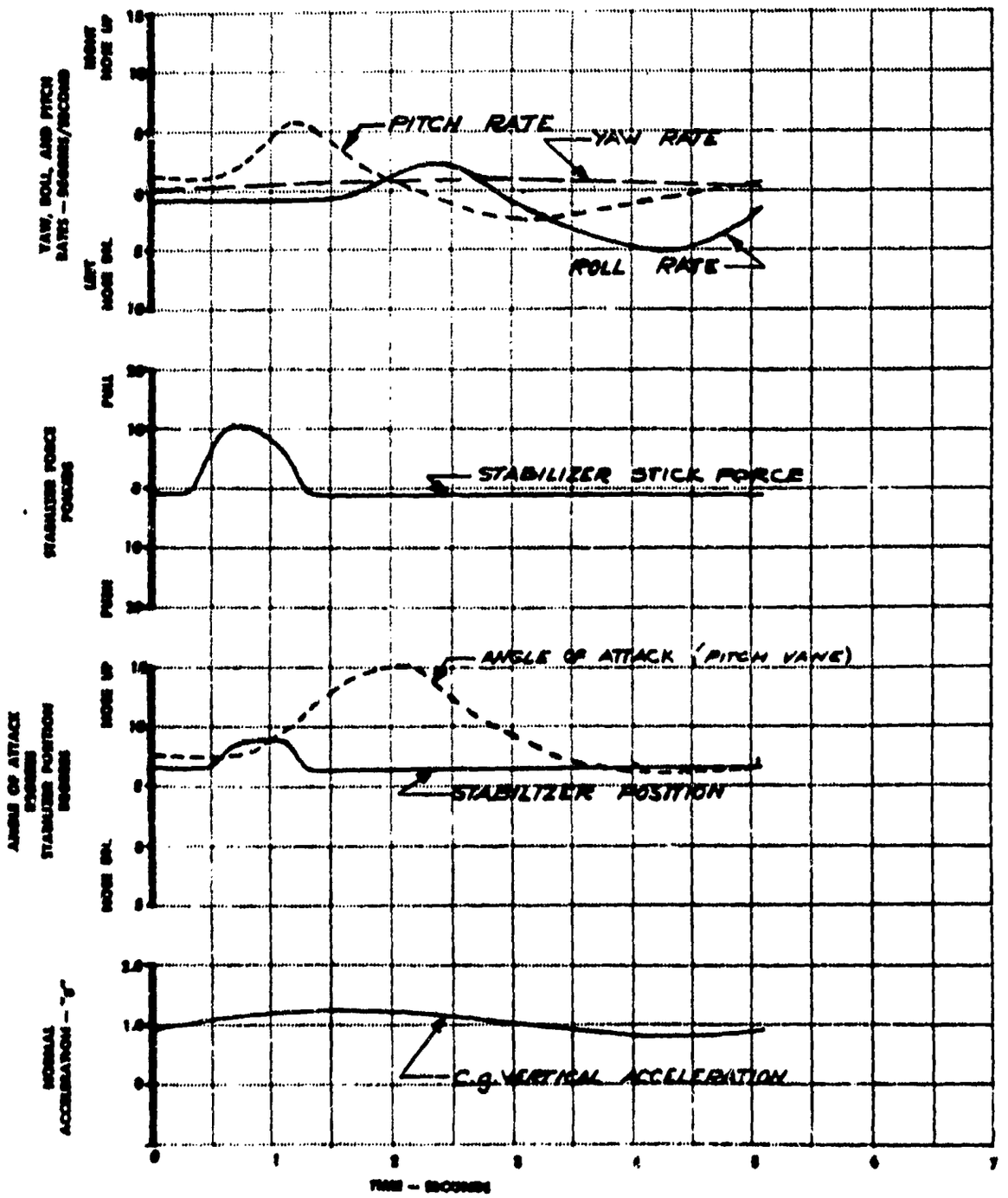


FIGURE 26	DYNAMIC LONGITUDINAL STABILITY

CONFIG. POWER APPROACH

TRIM CONDITIONS: CAS 180 KNOTS; CG 11.1 %MAC; STABILIZER 4.5 T.E.UP; ALTITUDE 14,000 FEET; WEIGHT 15,140 POUNDS; MACH NO. .334



CONFIG. CRUISE

FIGURE 27 | DYNAMIC LONGITUDINAL STABILITY

TRIM CONDITIONS: CAS 325 KNOTS | CG 5.9 %MAC | STABILIZER 1.1 TE UP

ALTITUDE 10350 FEET | WEIGHT 17900 POUNDS | MACH NO. .714

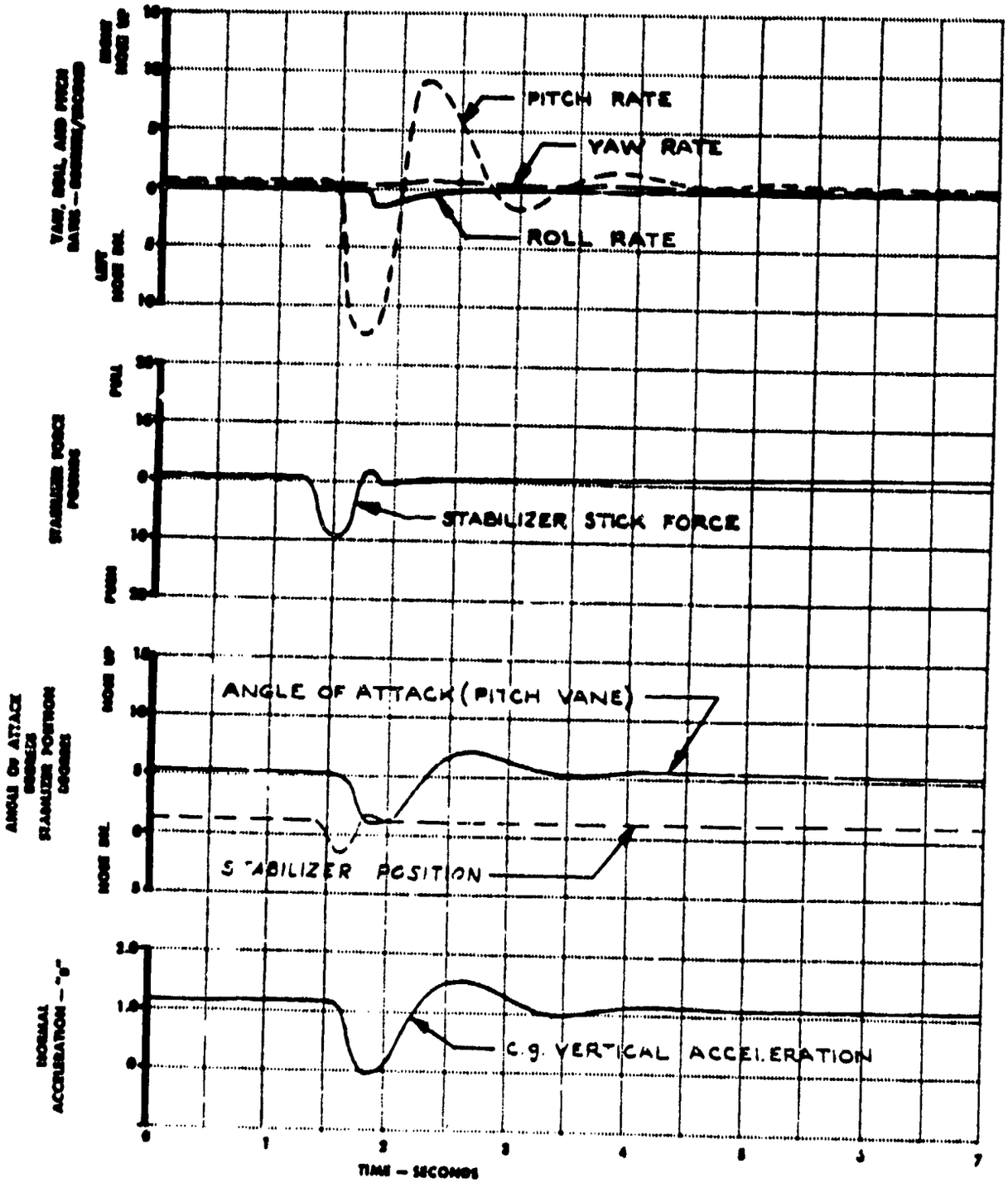


FIGURE 26 | DYNAMIC LONGITUDINAL STABILITY

CONFIG. CRUISE

TRIM CONDITIONS: CAS 395 KNOTS | CG 5.9 %MAC | STABILIZER 11 TE UP
ALTITUDE 10,250 FEET | WEIGHT 17,900 POUNDS | MACH NO. .714

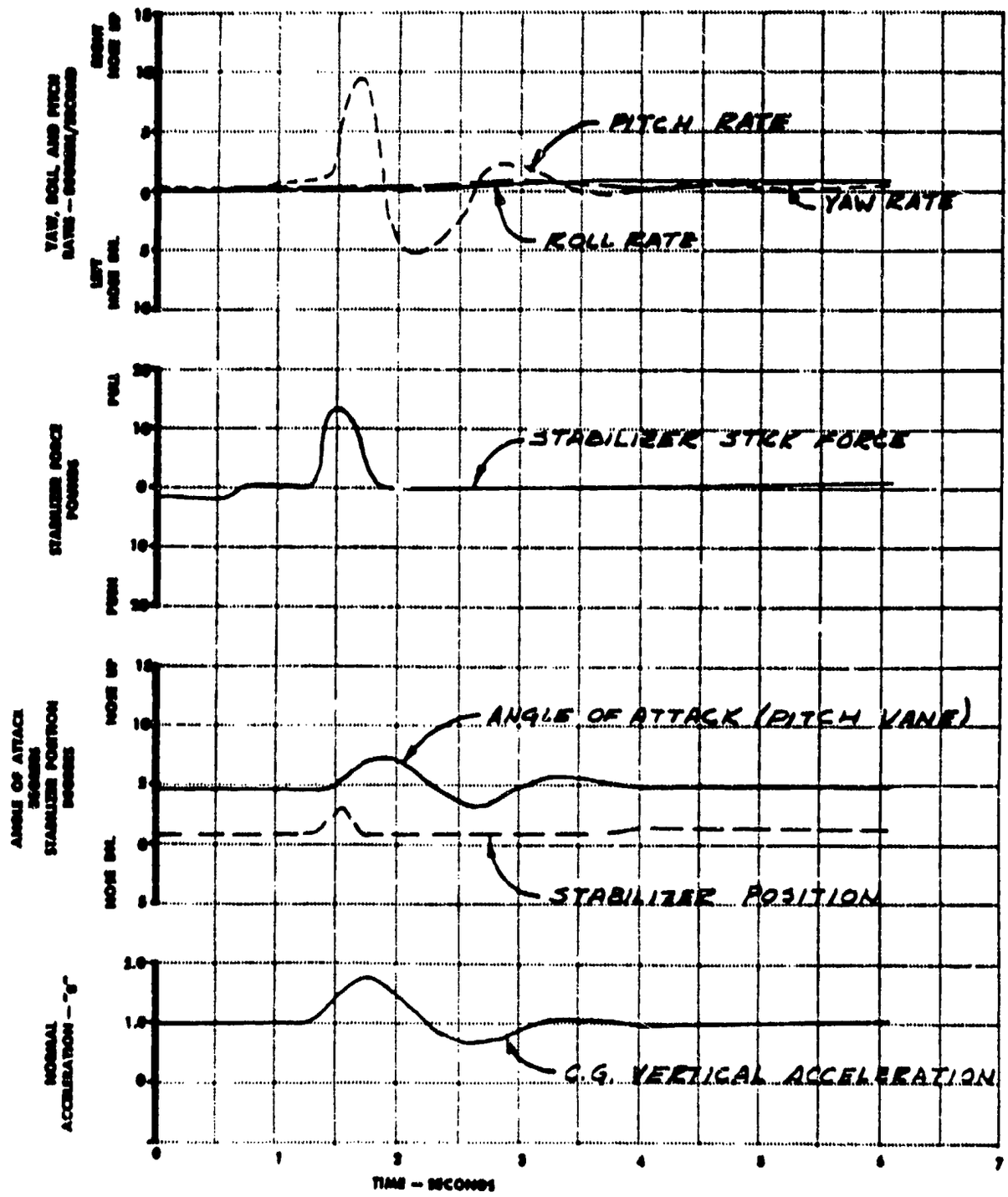


FIGURE 29 | DYNAMIC LONGITUDINAL STABILITY

CONFIG. CRUISE

TRIM CONDITIONS: CAS 315 KNOTS | CG 12.2 %MAC | STABILIZER 0.4 TE. UP

ALTITUDE 34850 FEET | WEIGHT 14,800 POUNDS | MACH NO. .92

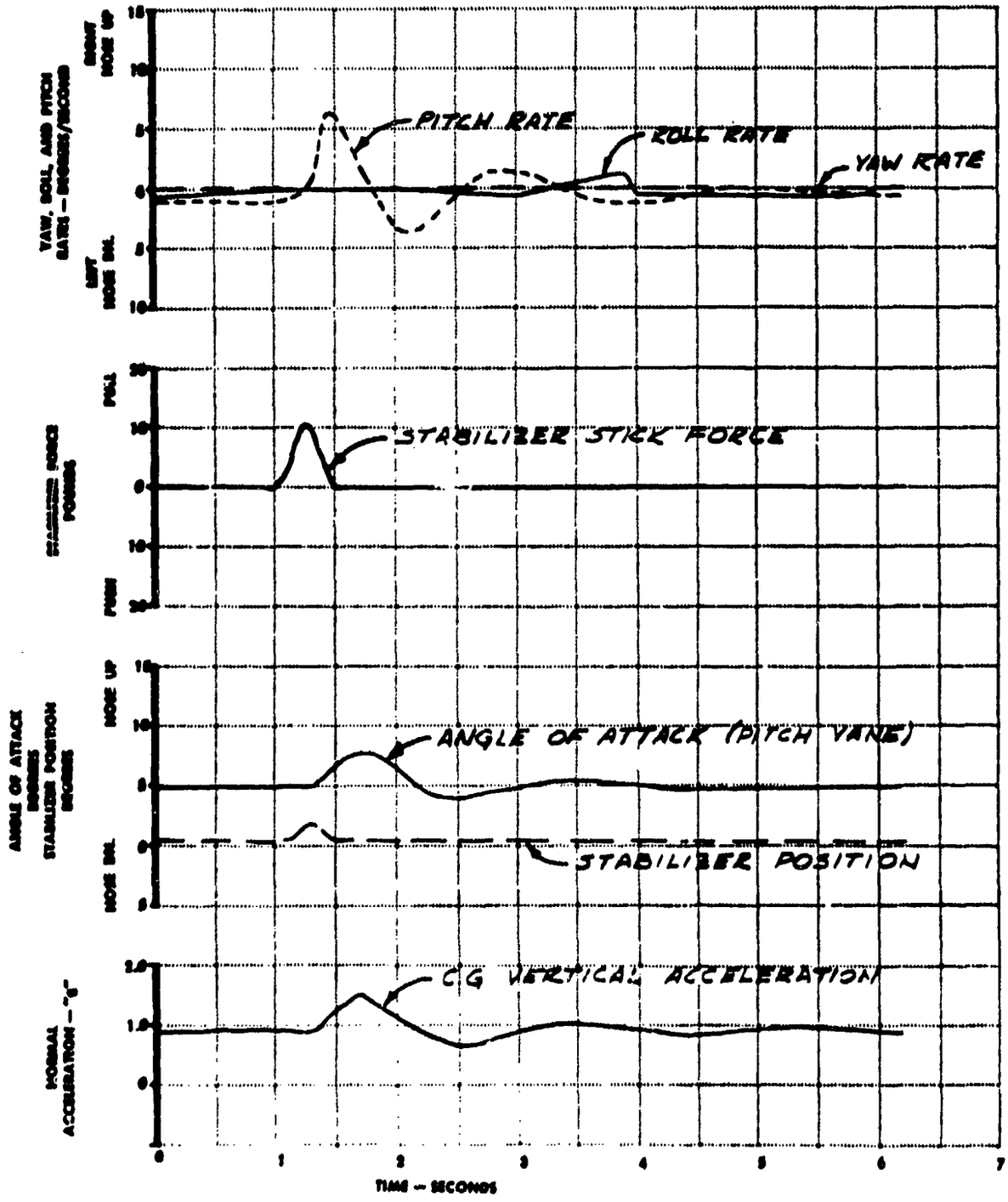


FIGURE 30	DYNAMIC LONGITUDINAL STABILITY

CONFIG. CRUISE

TRIM CONDITIONS: CAS 315 KNOTS; CG 12.2 %MAC; STABILIZER 0.4° TE UP; ALTITUDE 34850 FEET; WEIGHT 14800 POUNDS; MACH NO. .92

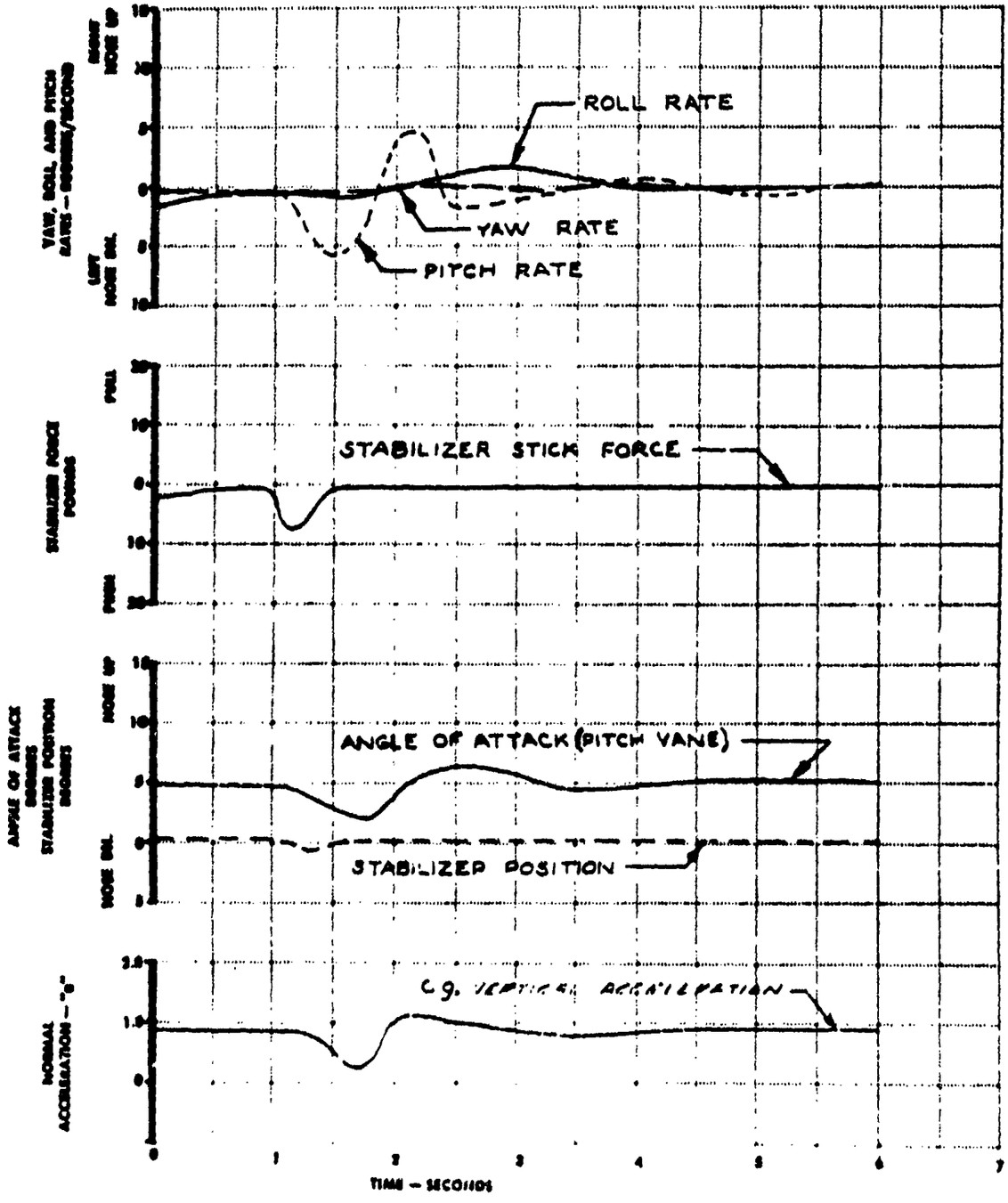
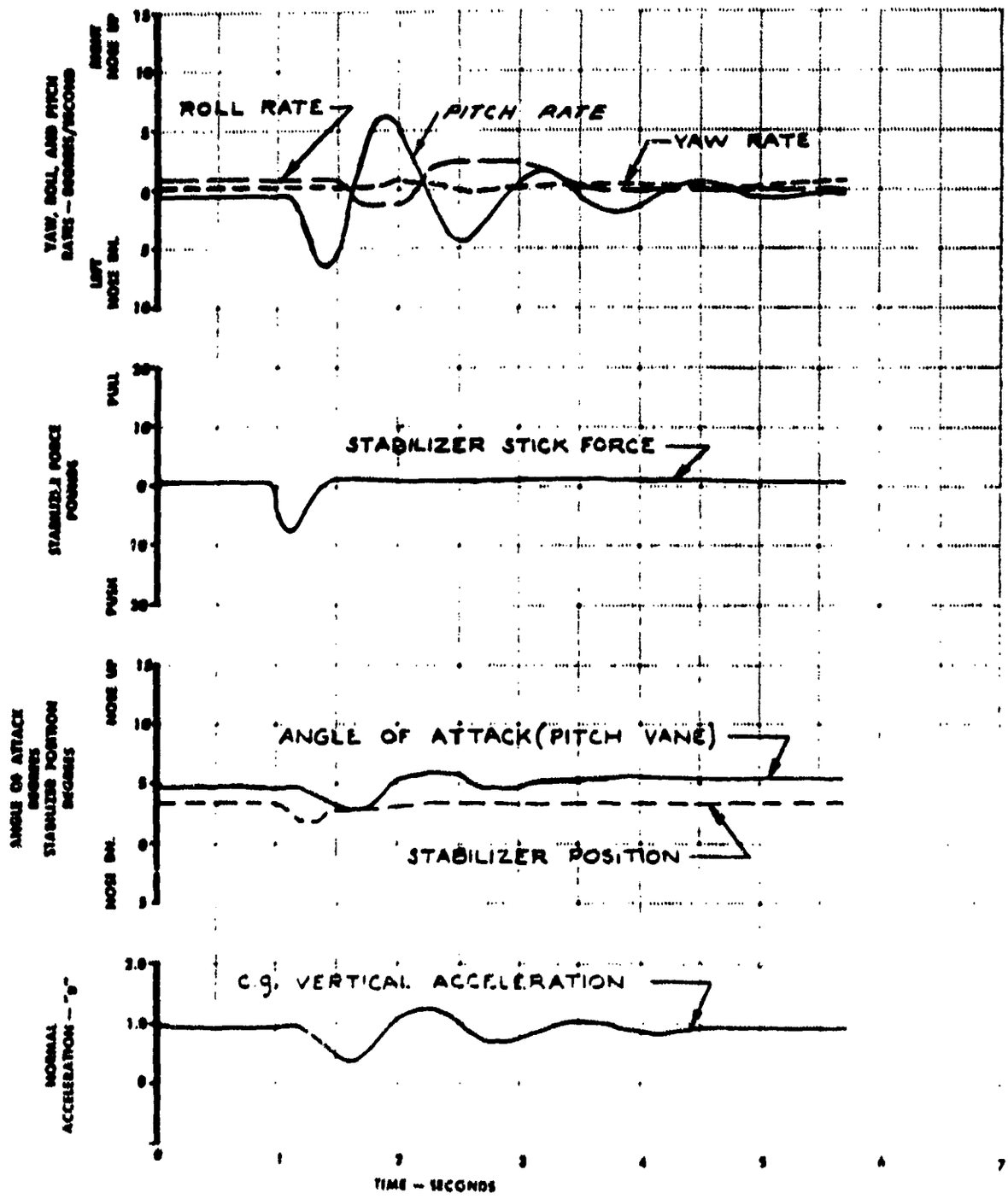


FIGURE 31	DYNAMIC LONGITUDINAL STABILITY

CONFIG. CLEAN

TRIM CONDITIONS: CAS 435 KNOTS; CG 9.5 %MAC; STABILIZER 4.0 TE UP; ALTITUDE 45,000 FEET; WEIGHT 15,740 POUNDS; MACH NO. 1.475



CONFIG. CLEAN

FIGURE 32 DYNAMIC LONGITUDINAL STABILITY

TRIM CONDITIONS: CAS 450 KNOTS; CG 9.5 %MAC; STABILIZER 4.0 TE UP; ALTITUDE 43,500 FEET; WEIGHT 15,740 POUNDS; MACH NO. 1.478

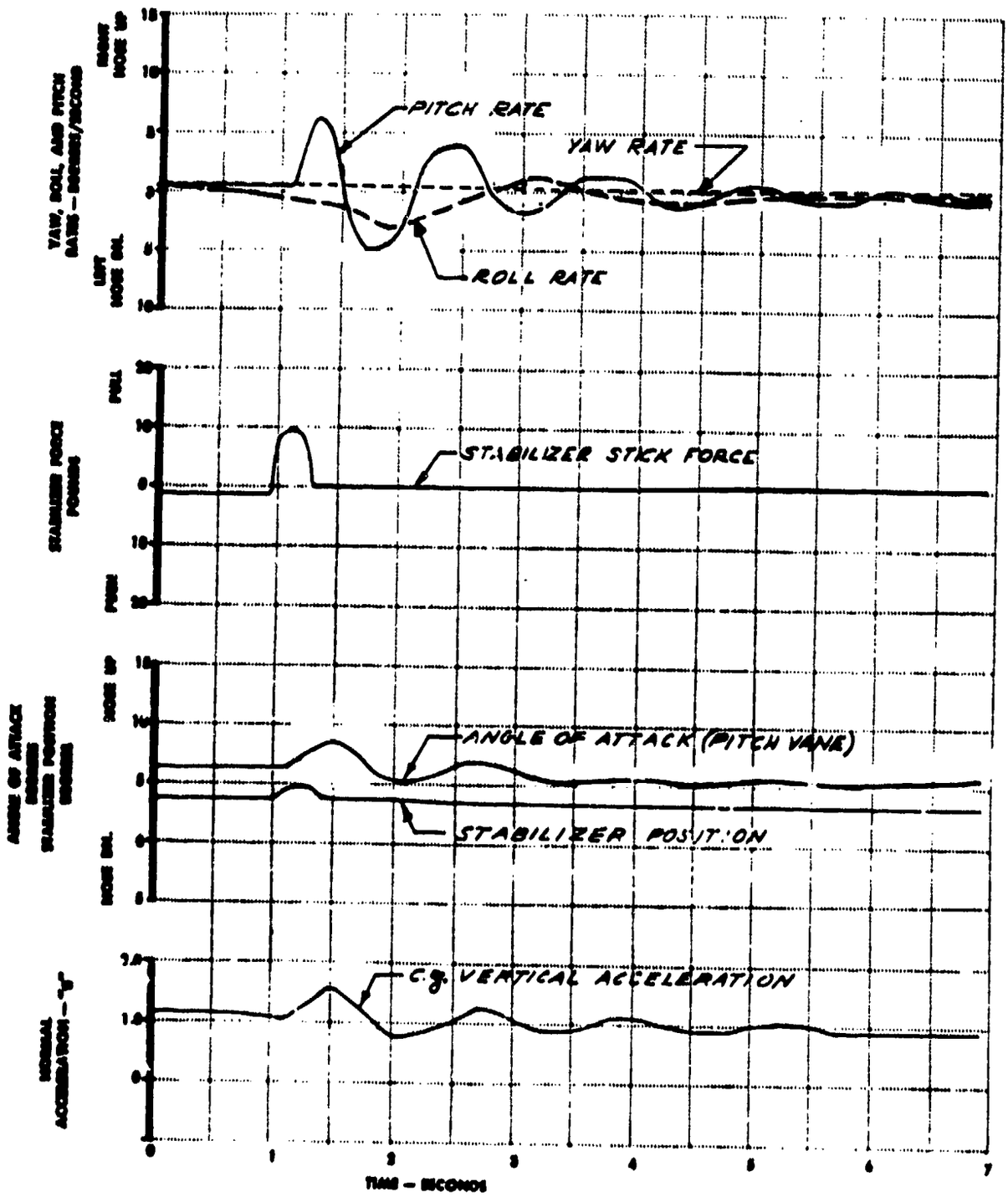


FIGURE 33	DYNAMIC LONGITUDINAL STABILITY

CONFIG. CLEAN

TRIM CONDITIONS: CAS 550 KNOTS; CG 10.55 %MAC; STABILIZER 2.7° T.E. UP; ALTITUDE 45,000 FEET; WEIGHT 15,720 POUNDS; MACH NO. 1.862

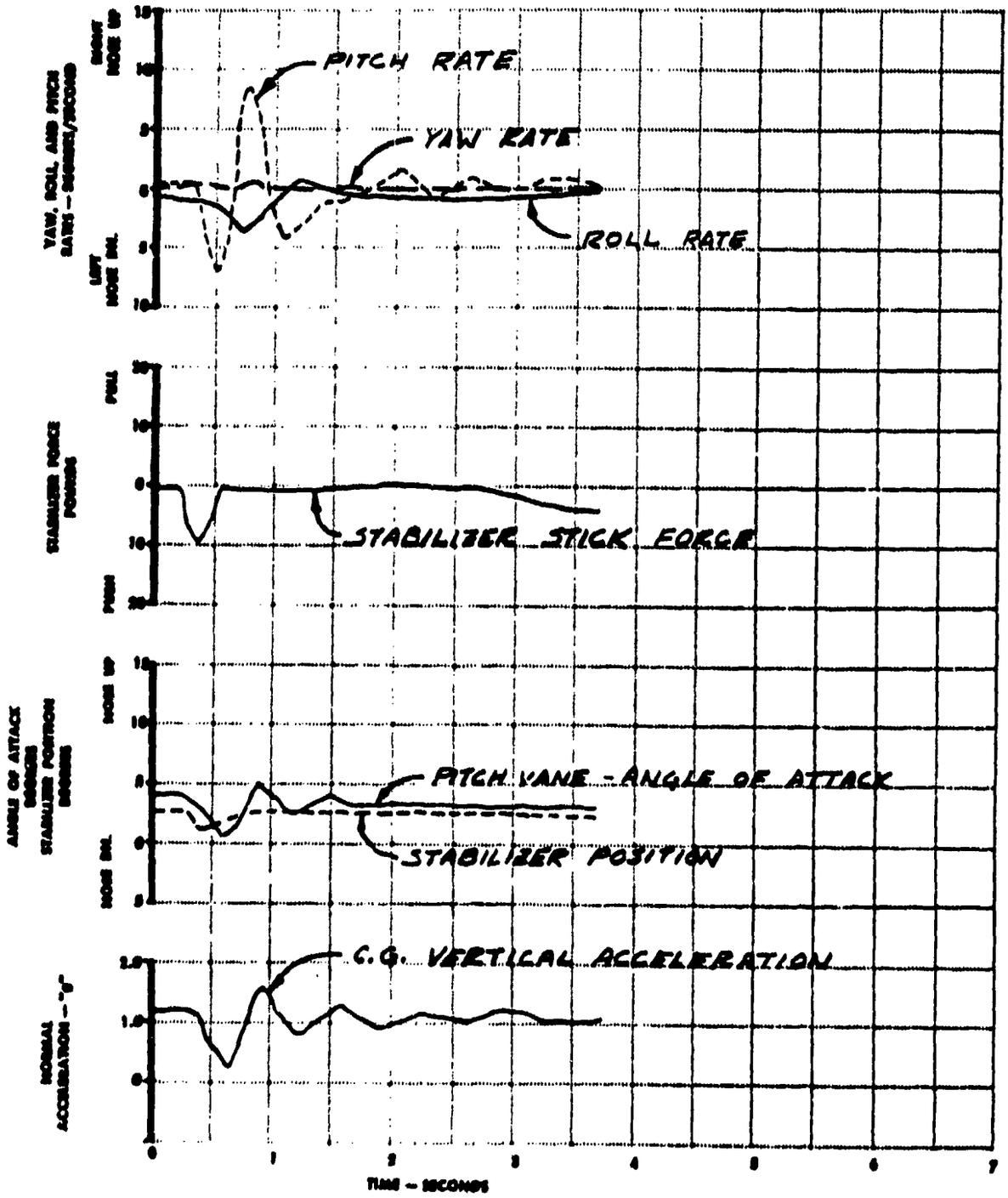
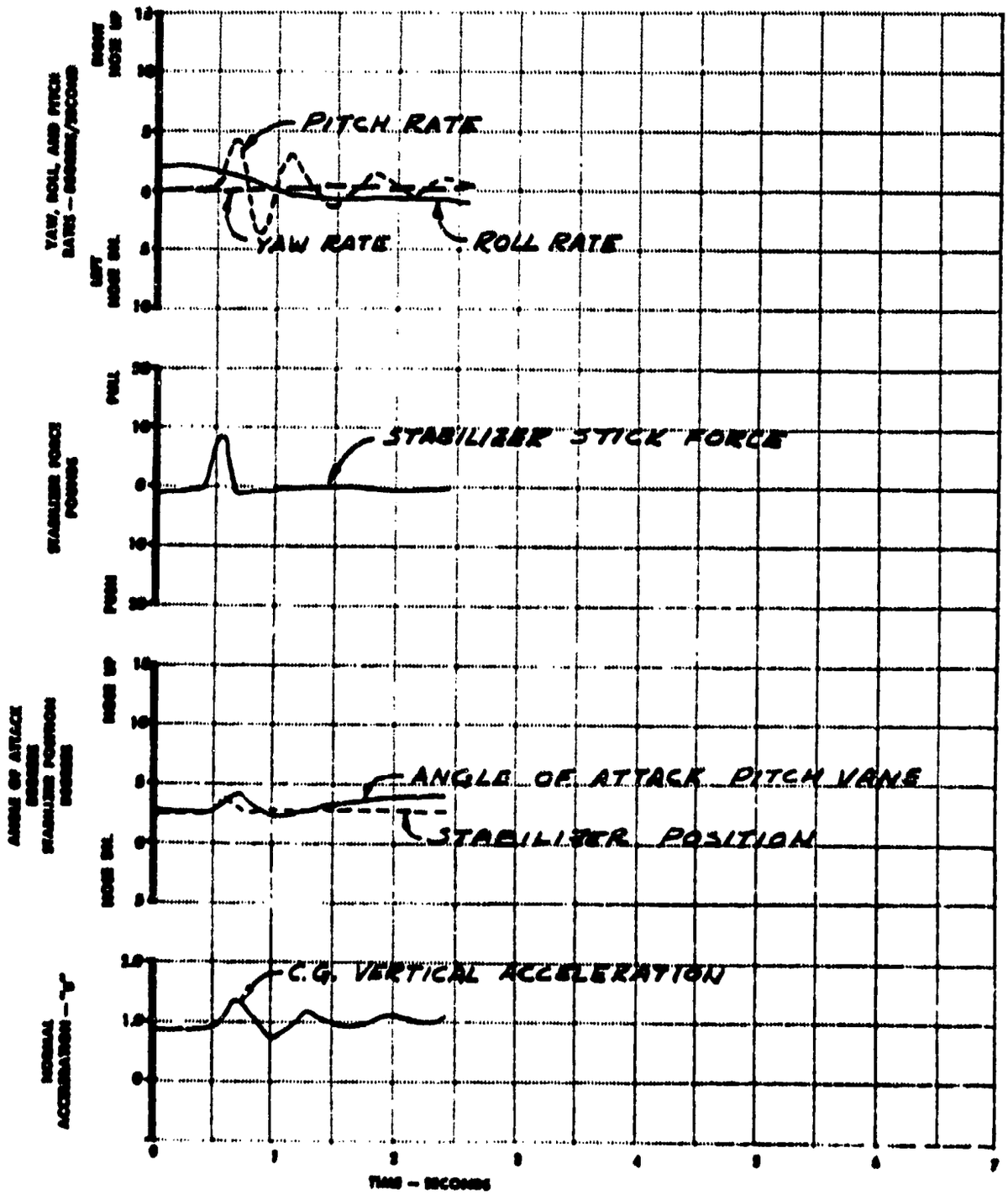


FIGURE 34 | DYNAMIC LONGITUDINAL STABILITY

CONFIG. CLEAN

TRIM CONDITIONS: CAS 550 KNOTS | CG 10.55 %MAC | STABILIZER 27° T.E. UP

ALTITUDE 45,000 FEET | WEIGHT 15,720 POUNDS | MACH NO. 1.862



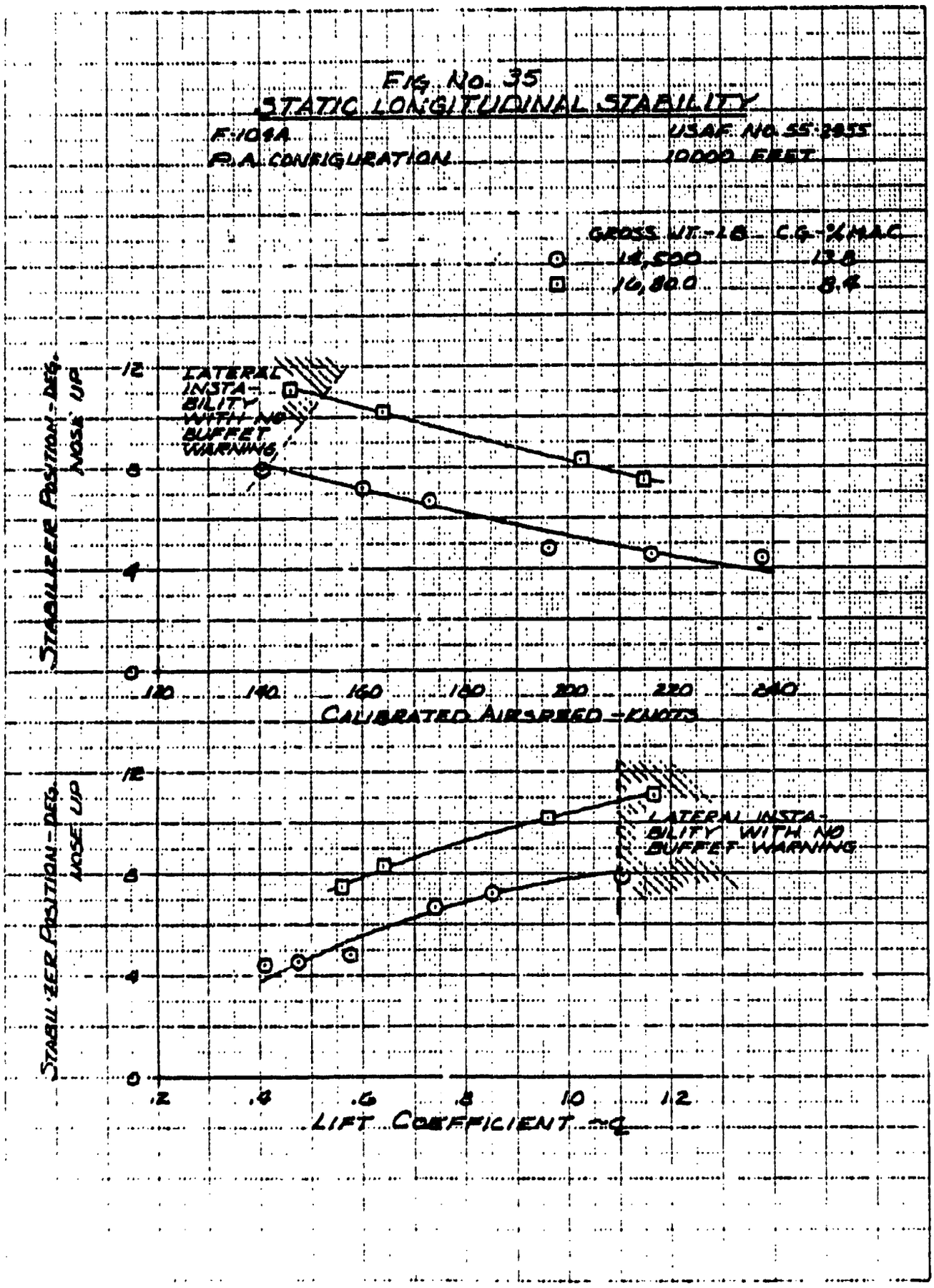


FIG. No. 35
STATIC LONGITUDINAL STABILITY
F-104A
P.A. CONFIGURATION
USAF NO. 55-2855
10000 FEET
GROSS WT.-LB
C.G.-% MAC
14,500
13.8
16,800
8.4
LATERAL INSTA-BILITY WITH NO BUFFET WARNING
STABILIZER POSITION - DEG. NOSE UP
12
8
4
0
120
140
160
180
200
220
240
CALIBRATED AIRSPEED - KNOTS
LATERAL INSTA-BILITY WITH NO BUFFET WARNING
STABILIZER POSITION - DEG. NOSE UP
12
8
4
0
.2
.4
.6
.8
1.0
1.2
LIFT COEFFICIENT - C_L

FIG. NO. 36

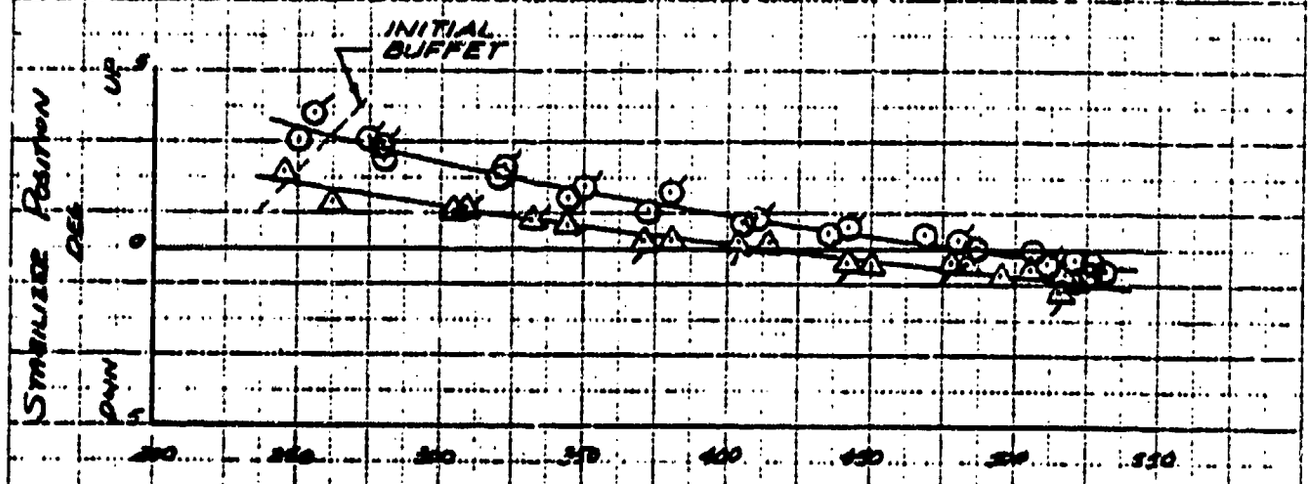
STATIC LONGITUDINAL STABILITY

F-104A USAF NO 56-2935

CRUISE CONFIGURATION 10,000 FT.

GROSS WT-LB	CG - % MAC
17400	6.65
15100	11.70

FLAGS DENOTE DECELERATION POINTS



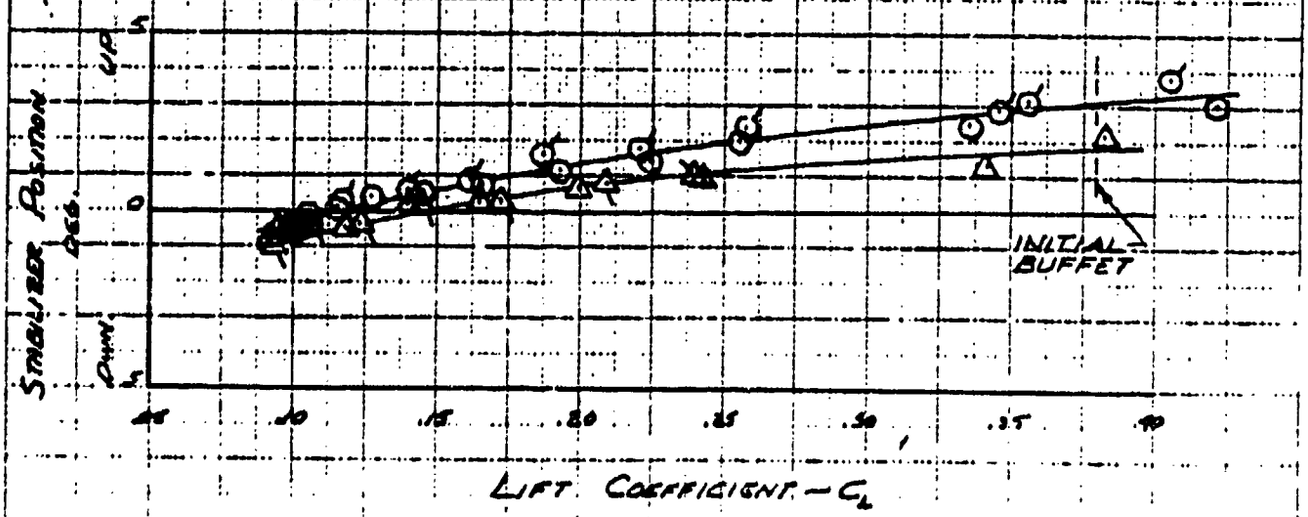


FIG NO 37

STATIC LONGITUDINAL STABILITY

F-104A USAF NO 55-2955

CLEAN CONFIGURATION 35000 FEET

	GROSS WEIGHT	CG - % MAC
START	17180	6.8
FINISH	16000	9.0

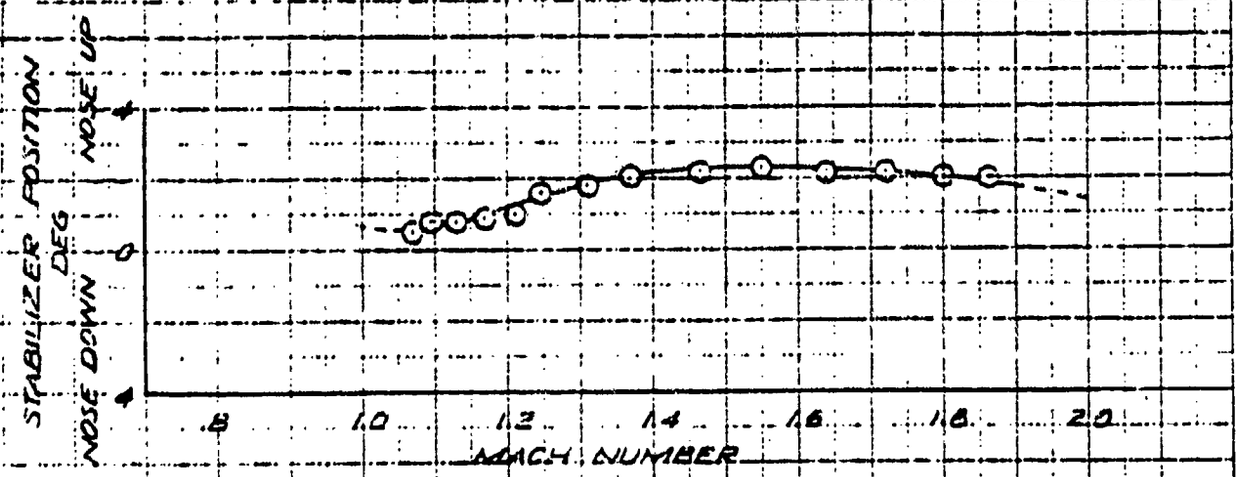


FIG. NO. 35

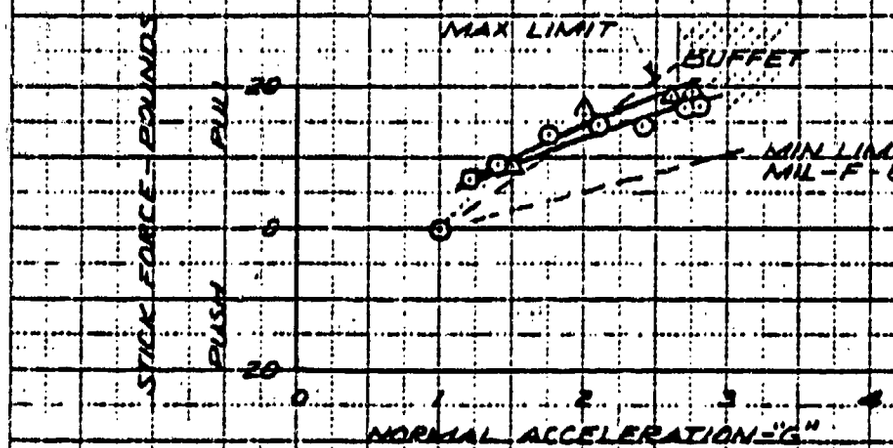
MANEUVERING FLIGHT CHARACTERISTICS

F-104A USAF NO 55-2955

CRUISE CONFIGURATION 10000 FEET

TRIM CONDITIONS

SYMBOL	MACH NO	STAB ADS-DEG	GROSS WT-LB	CG-%MAC
△	0.71	1.1 NU	17300	6.8
⊙	0.71	0.9 NU	14900	11.95



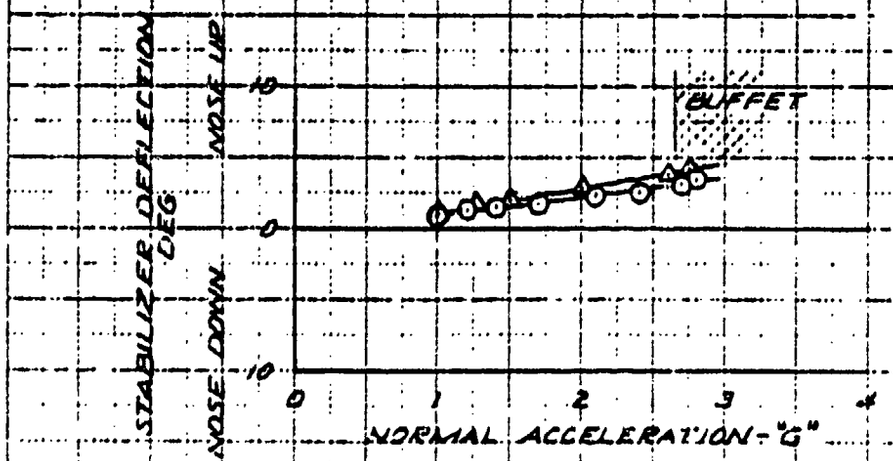


FIG. NO. 34

MANEUVERING FLIGHT CHARACTERISTICS

F-104A USAF NO. 55-2955

CRUISE CONFIGURATION 35000 FEET

TRIM CONDITIONS

MACH NO	STAB POS-DEG	GROSS WT-LB	CG-% MAC
0.93	2 NU	14800	12.2

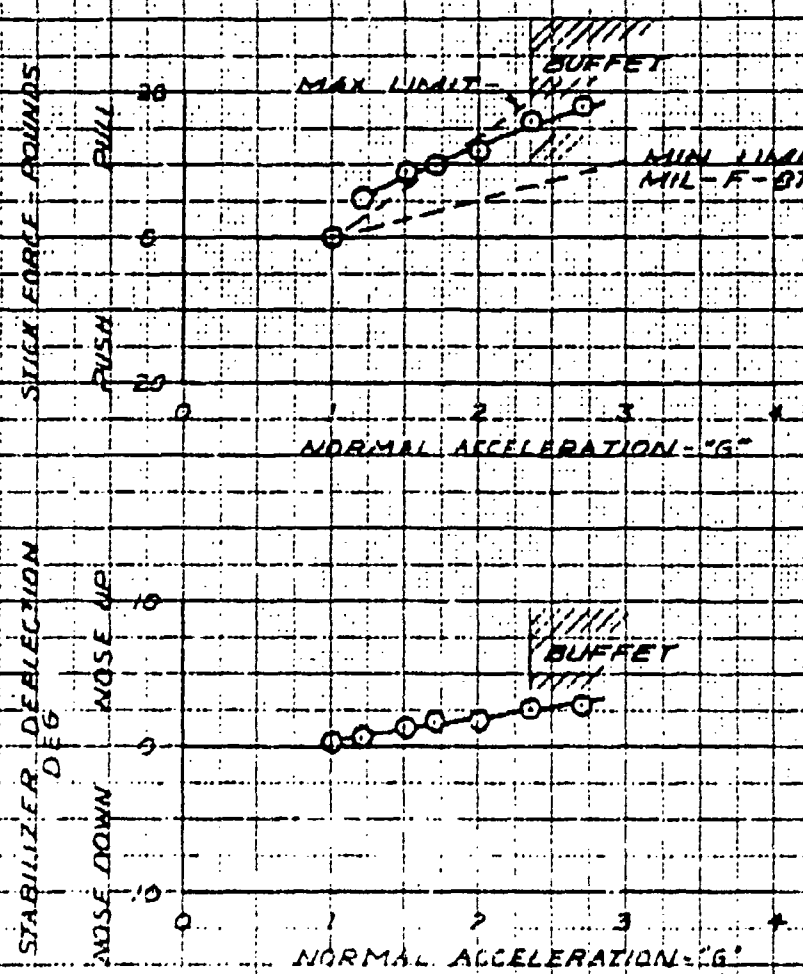


FIG NO. 40

MANEUVERING FLIGHT CHARACTERISTICS

F-104A USAF NO 55-2955

CRUISE CONFIGURATION 45000 FEET

TRIM CONDITIONS

MACH NO	STAB. POS.-DEG	GROSS WT-LB	CG - % MAC
1.49	3.2 NU	15200	11.2

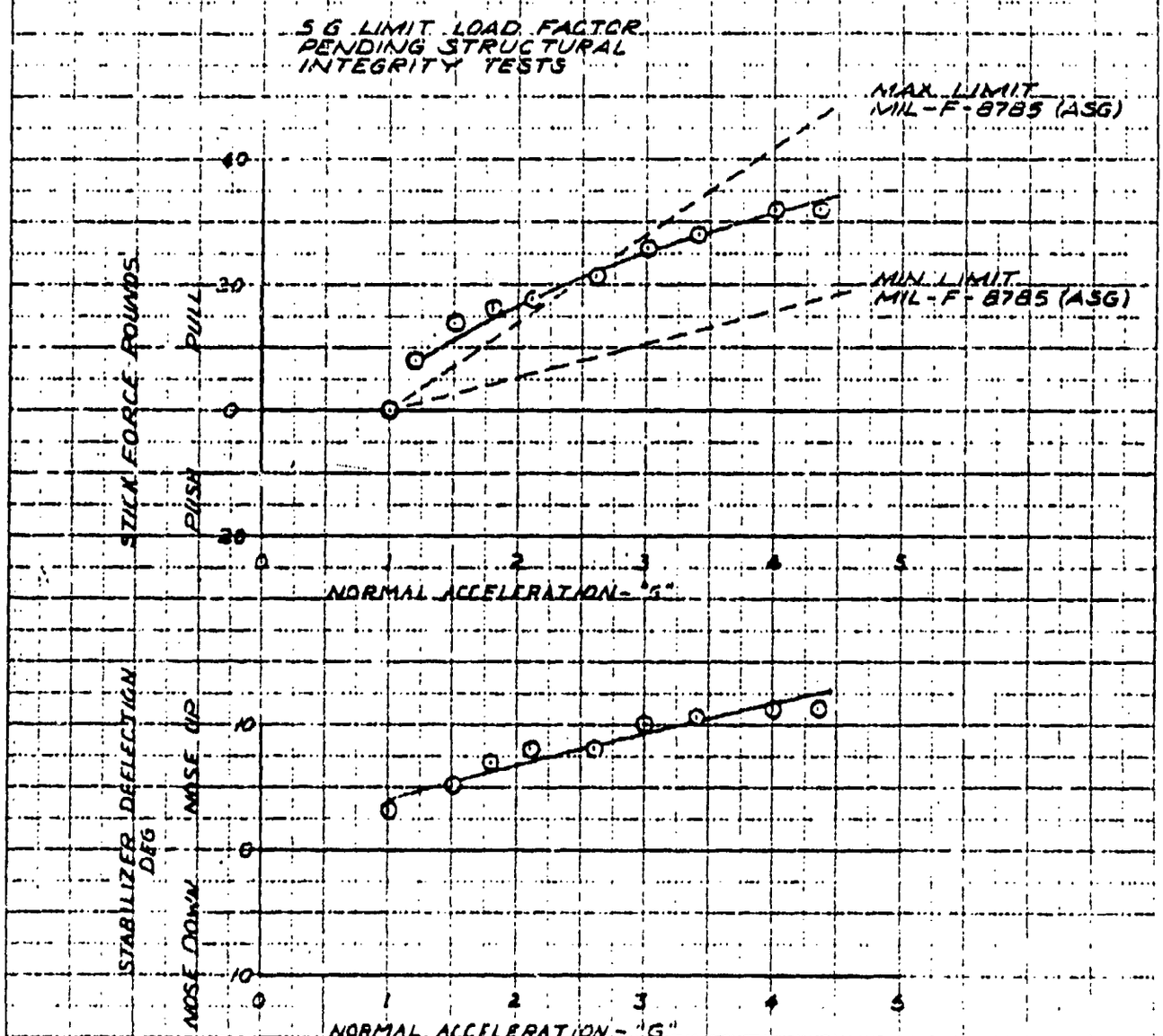


FIG NO 41

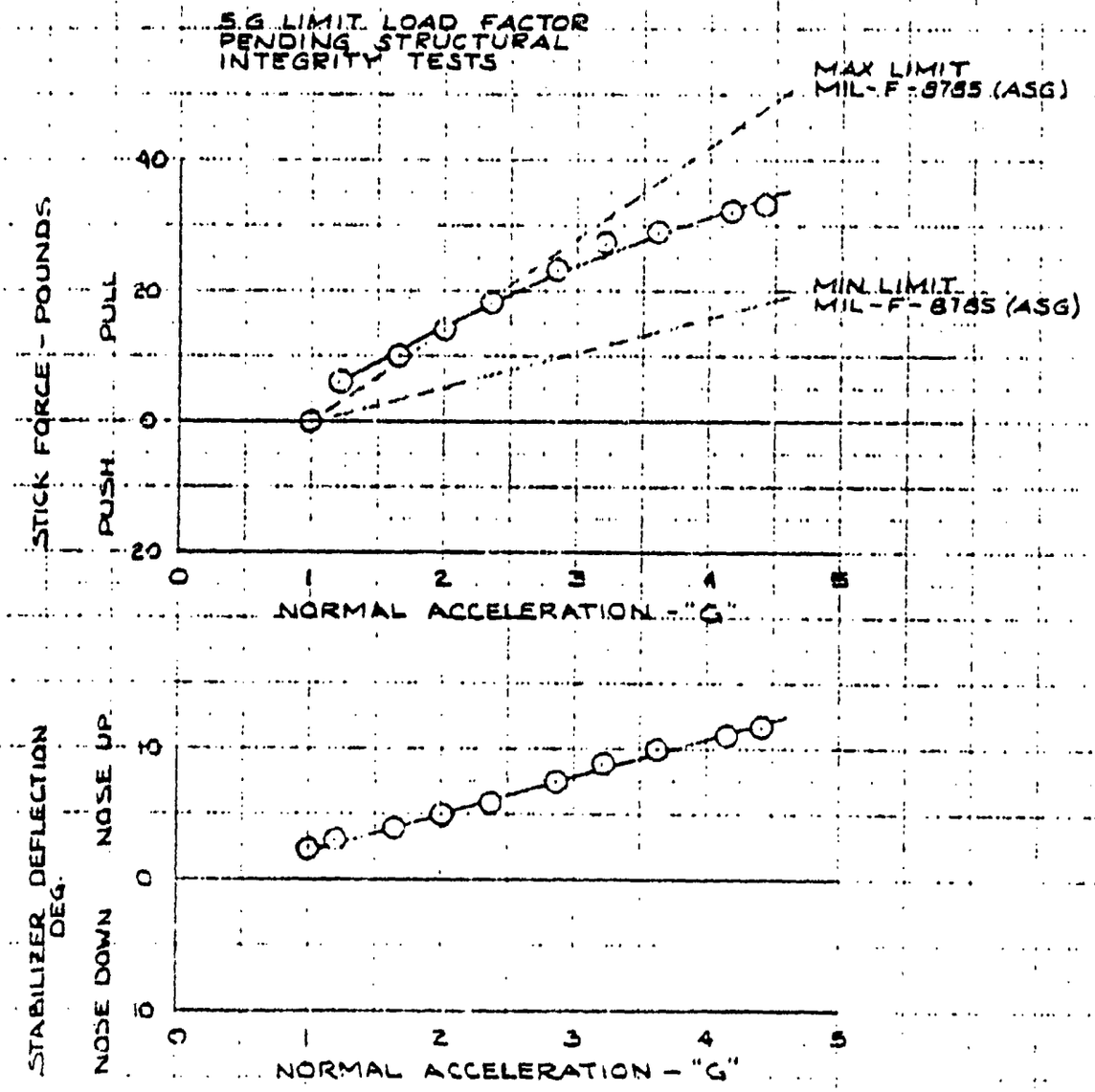
MANEUVERING FLIGHT CHARACTERISTIC

F-104A USAF NO 55-2955

CRUISE CONFIGURATION 45,000 FT

TRIM CONDITIONS

MACH NO	STAB. POS. - DEG	GROSS WT - LB	C.G. - % MAC
1.91	2.4 NU	15,400	10.7



RGUM 42	DYNAMIC DIRECTIONAL STABILITY

CONFIGURATION POWER APPROACH

TRIM CONDITIONS CAS 185 KNOTS CG 9.4 %MAC

MACH NO. .337 ALTITUDE 10610 FEET WEIGHT 15,975 POUNDS

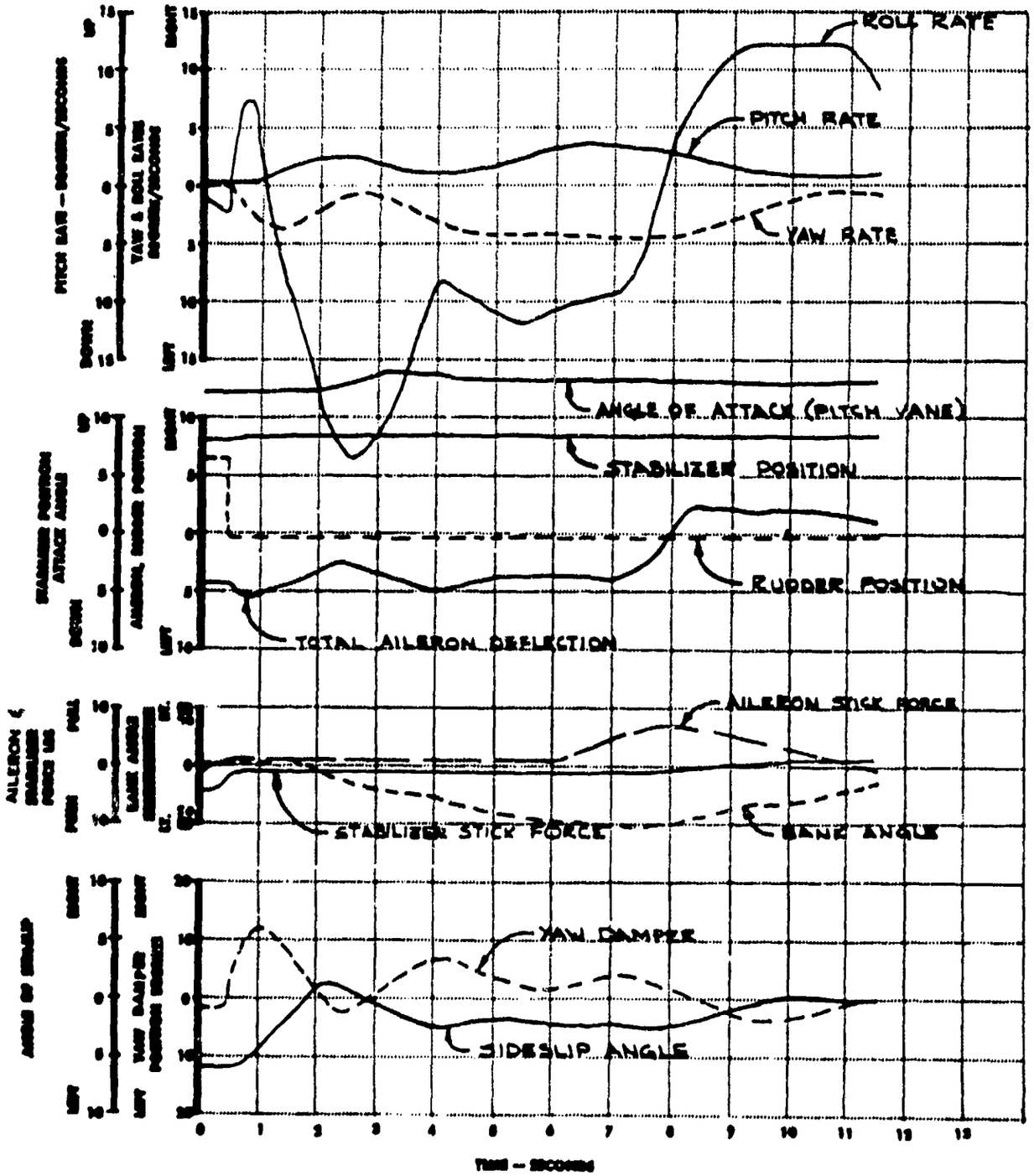


FIGURE 43 | DYNAMIC DIRECTIONAL STABILITY

CONFIGURATION CRUISE

TRIM CONDITIONS CAS 335.5 KNOTS CG 7.8 %MAC

MACH NO. .718 ALTITUDE 10700 FEET WEIGHT 16750 POUNDS

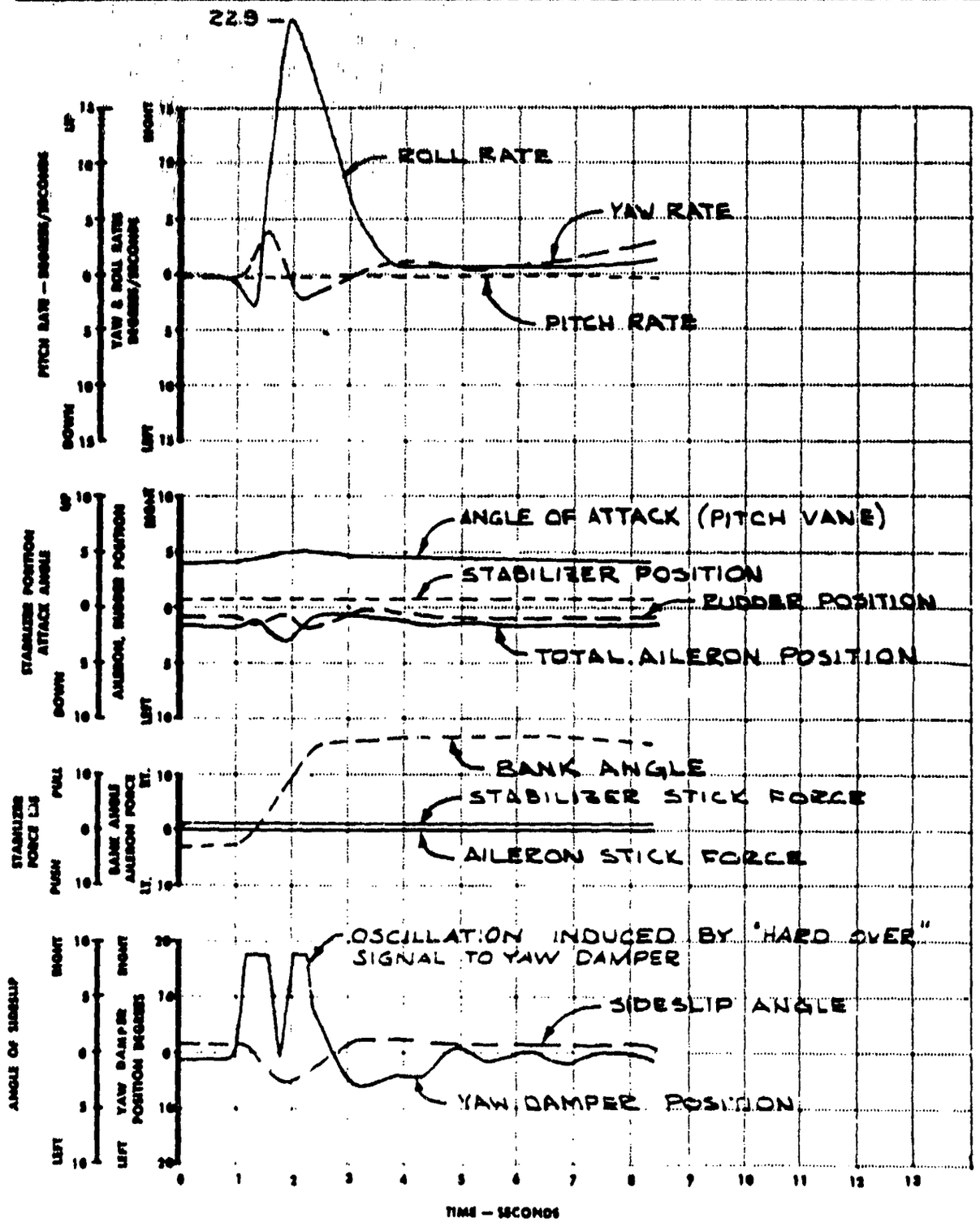


FIGURE 44	DYNAMIC DIRECTIONAL STABILITY

CONFIGURATION CRUISE

TRIM CONDITIONS CAS 320 KNOTS CG 13.1 %MAC

MACH NO. .916 ALTITUDE 34500 FEET WEIGHT 14500 POUNDS

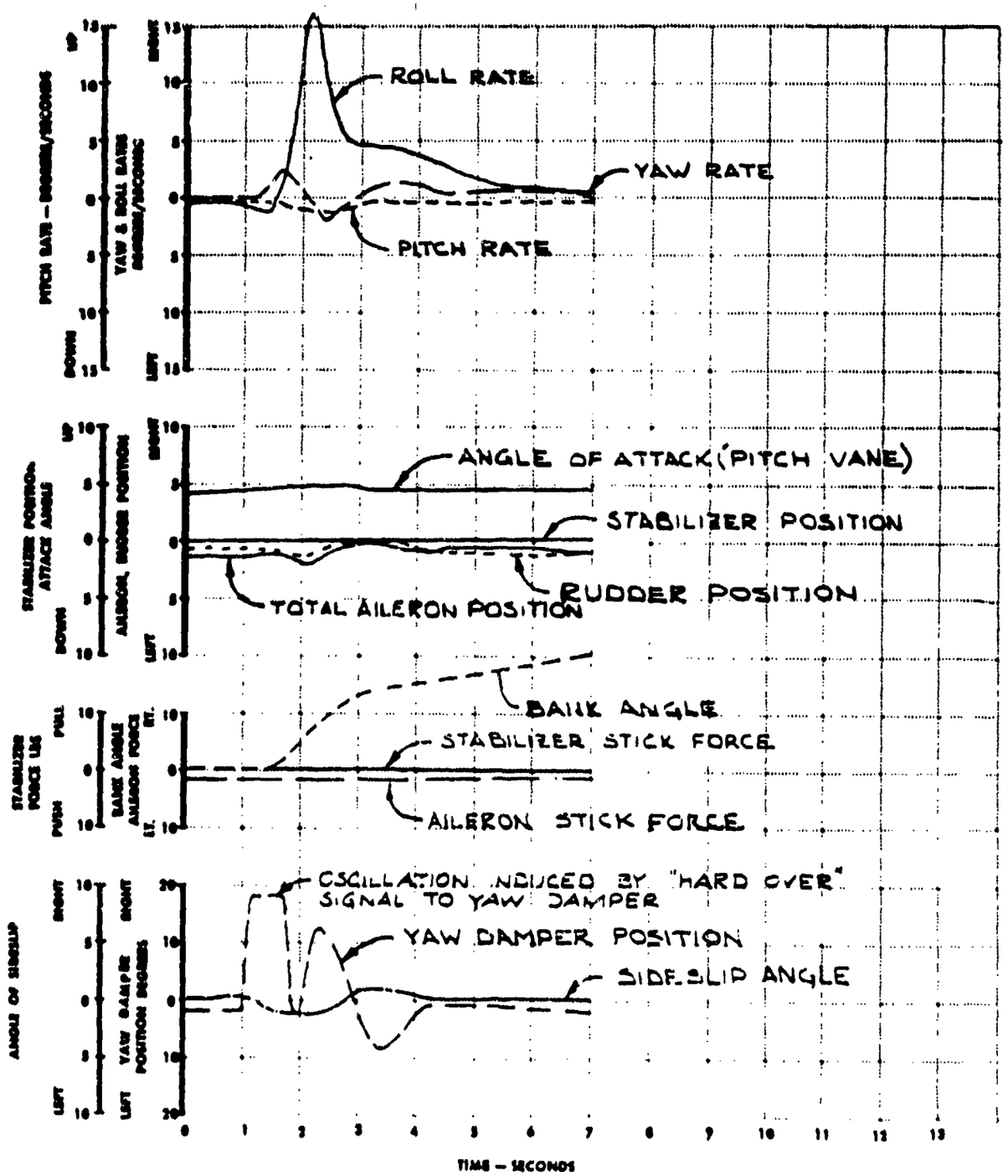


FIGURE 45	DYNAMIC DIRECTIONAL STABILITY

CONFIGURATION CLEAN

TRIM CONDITIONS CAS 470 KNOTS CG 8.55 %MAC

MACH NO. 1.492 ALTITUDE 42000 FEET WEIGHT 16,440 POUNDS

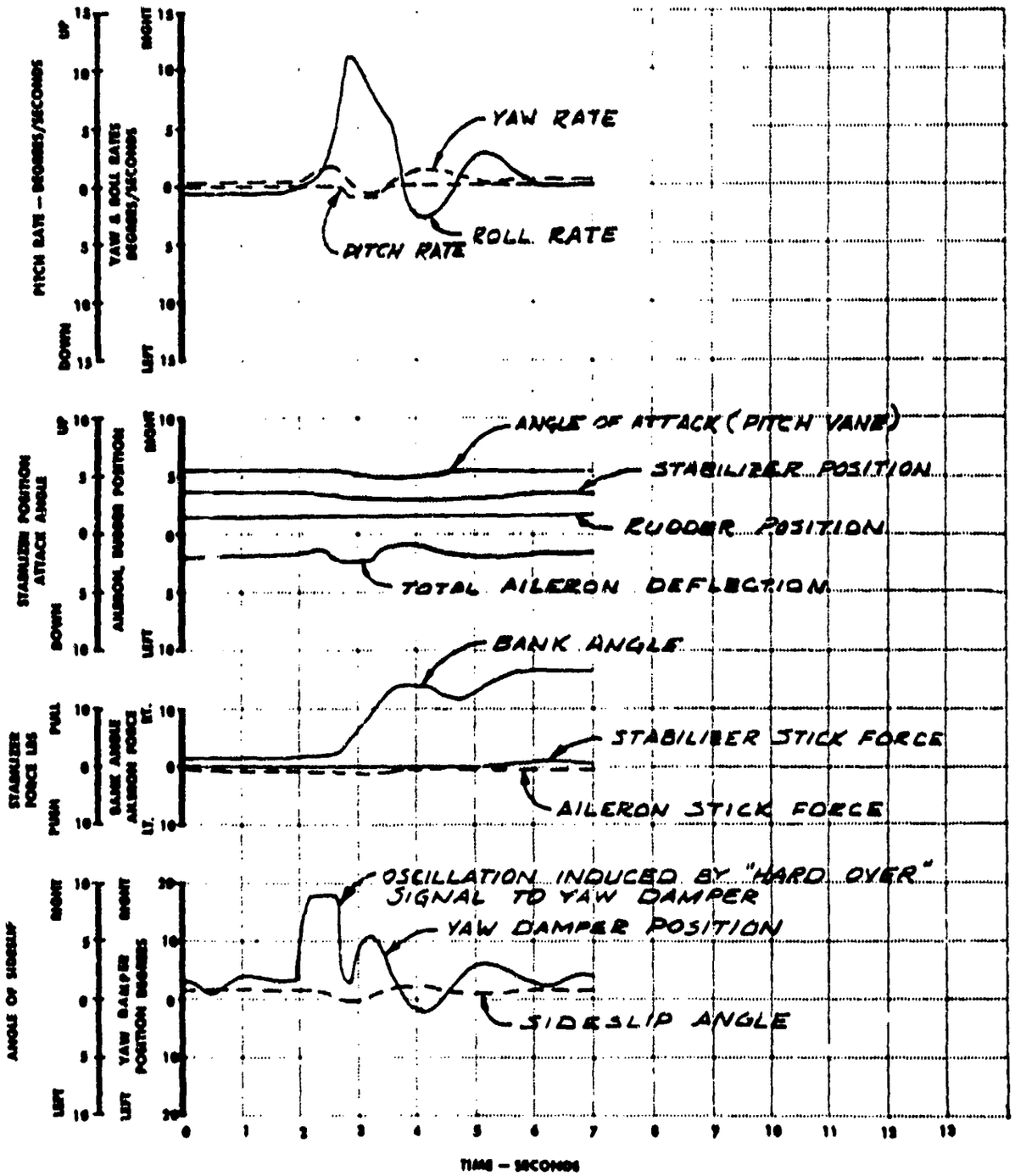
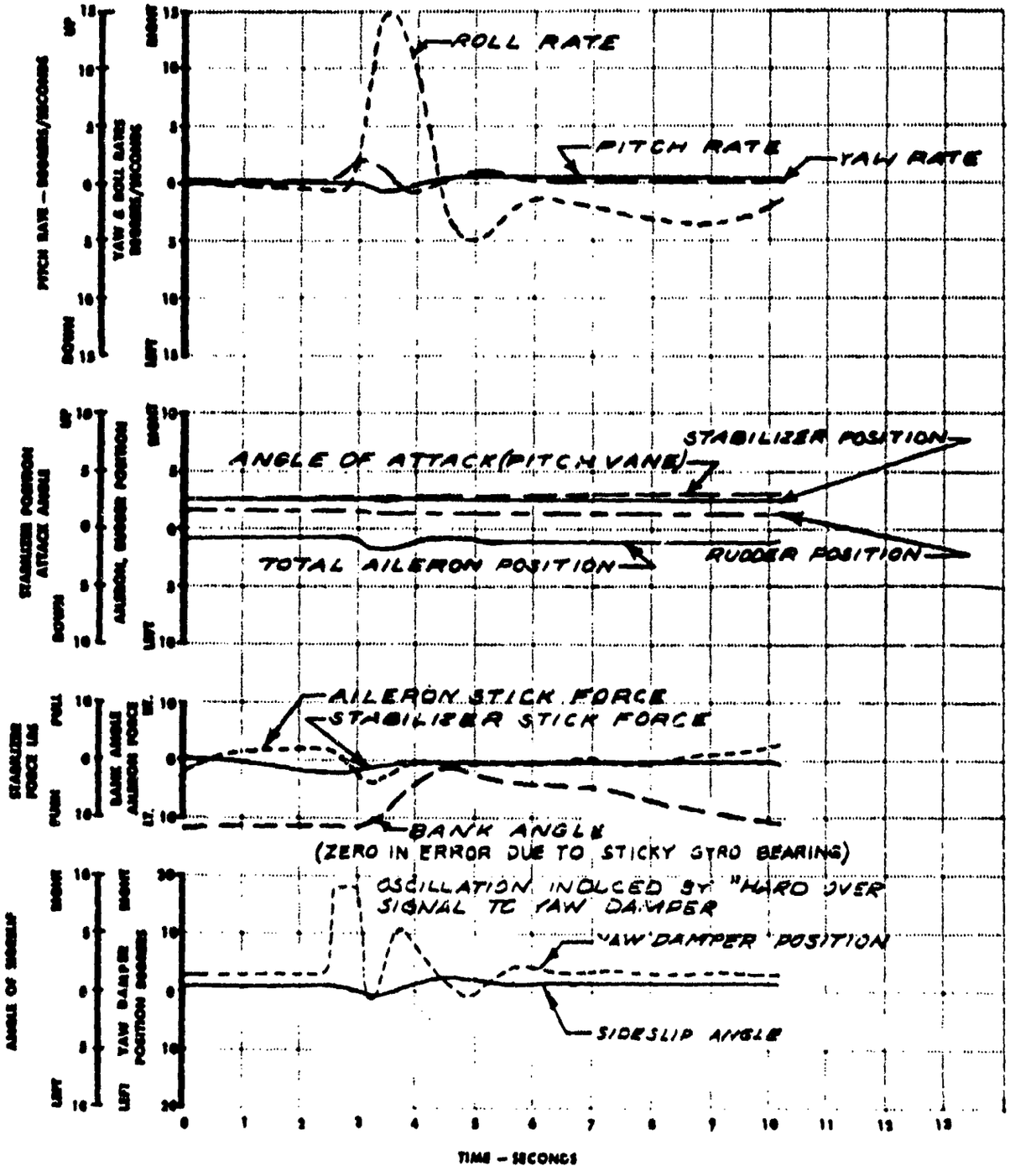


FIGURE 46	DYNAMIC DIRECTIONAL STABILITY

CONFIGURATION CLEAN

TRIM CONDITIONS CAS 560 KNOTS CG 11.0 %MAC

MACH NO. 1.895 ALTITUDE 45,000 FEET WEIGHT 15,140 POUNDS



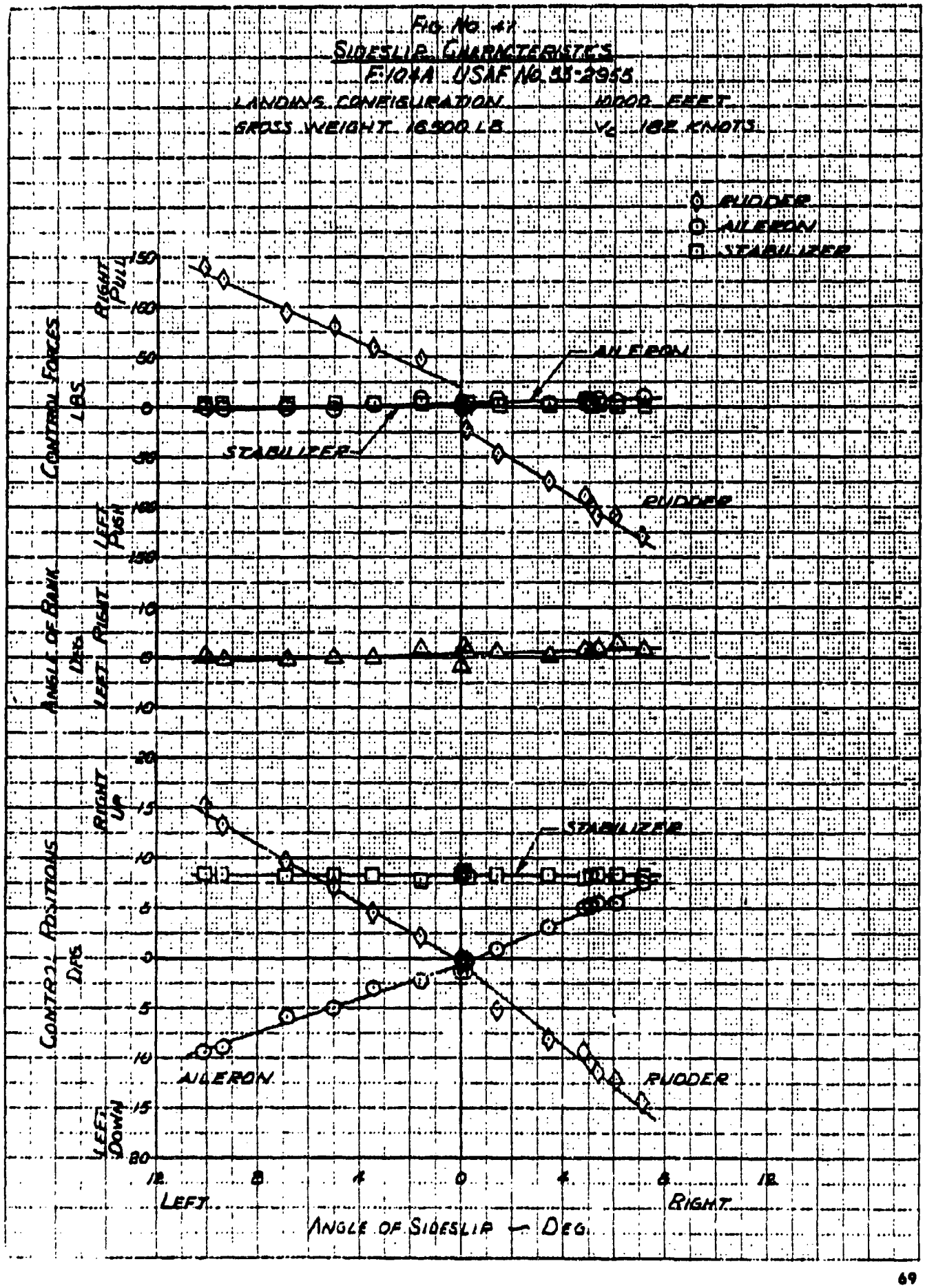


FIG. NO. 41
SIDESLIP CHARACTERISTICS
F-104A USAF NO. 55-2955
LANDING CONFIGURATION
10000 FEET
GROSS WEIGHT 16500 LB
V_c 182 KNOTS
RUDDER
AILERON
STABILIZER
CONTROL FORCES LBS
RIGHT PULL
LEFT PUSH
ANGLE OF BANK DEG.
RIGHT
LEFT
CONTROL POSITIONS DEG.
RIGHT UP
LEFT DOWN
150
100
50
0
50
100
150
10
0
10
20
15
10
5
0
5
10
15
20
12
8
4
0
4
8
12
LEFT
RIGHT
ANGLE OF SIDESLIP — DEG.

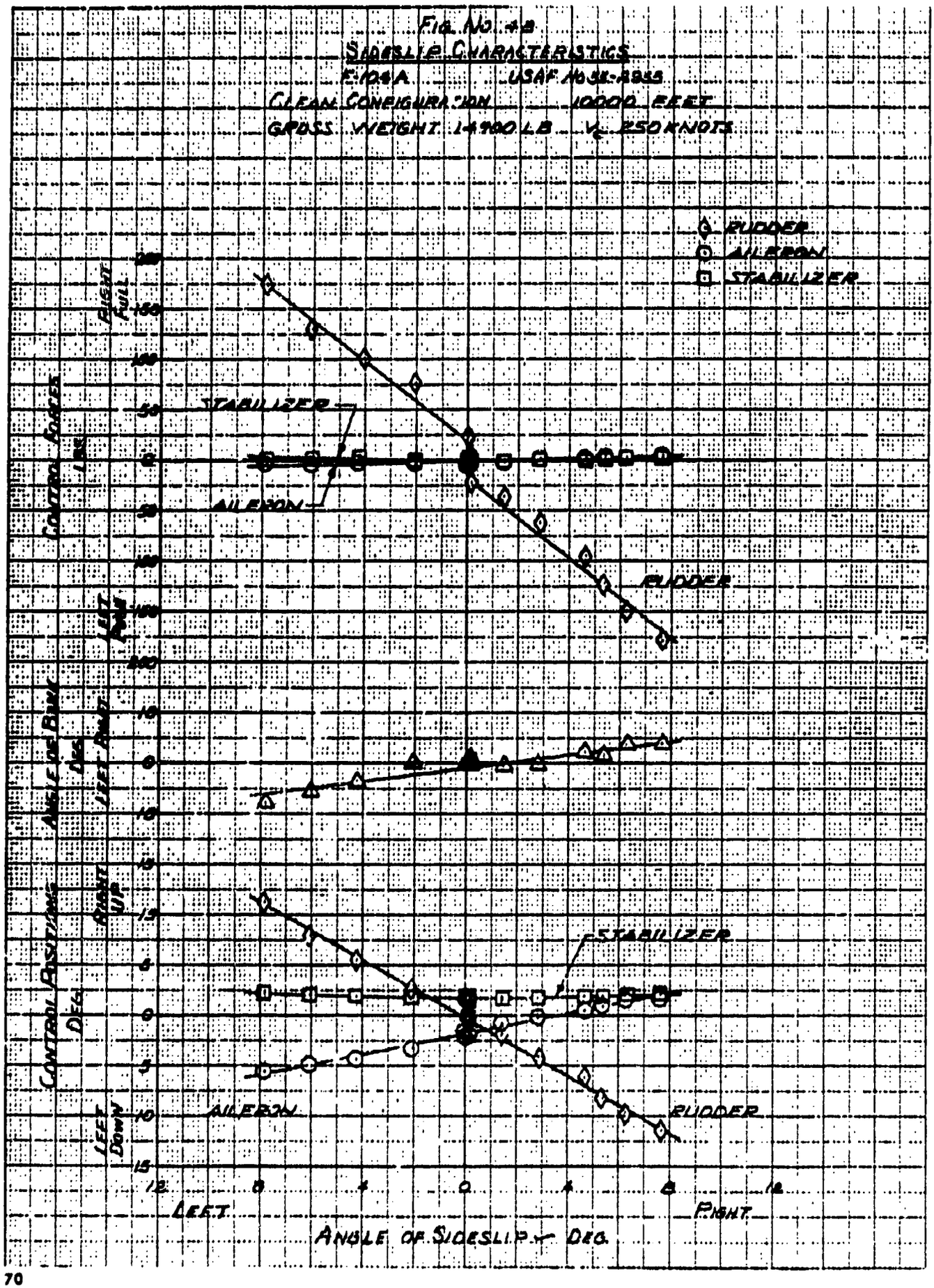


FIG. NO. 48
SIDESLIP CHARACTERISTICS
F-104A USAF No 56-8855
CLEAN CONFIGURATION 10000 FEET
GROSS WEIGHT 14900 LB V_c 250 KNOTS
RUDDER
AILERON
STABILIZER
CONTROL FORCE LBS
RIGHT FULL
LEFT FULL
STABILIZER
AILERON
RUDDER
ANGLE OF BANK DEG
LEFT WING
RIGHT WING
CONTROL POSITIONS DEG
RIGHT UP
LEFT DOWN
STABILIZER
AILERON
RUDDER
LEFT
RIGHT
ANGLE OF SIDESLIP — DEG

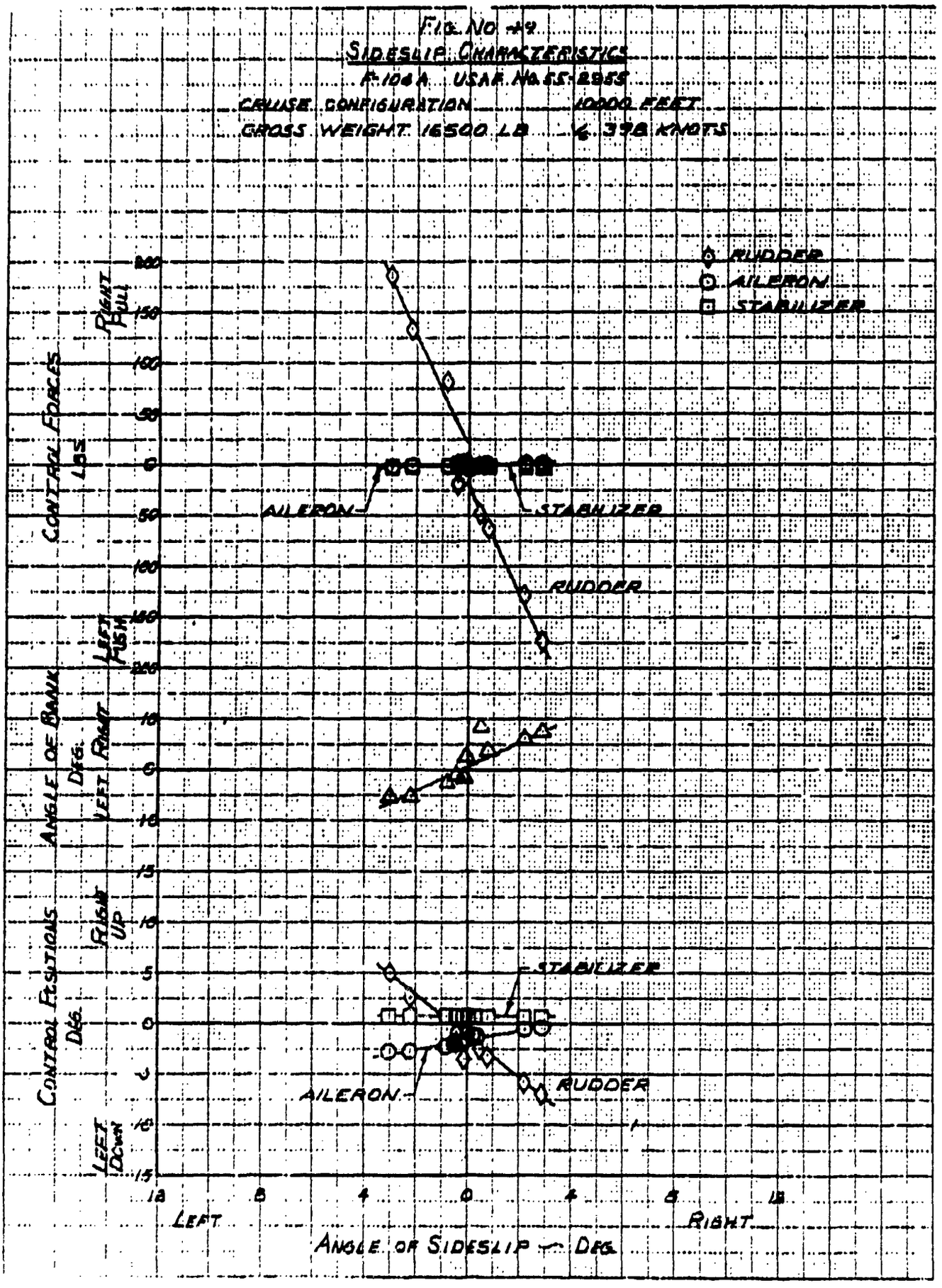


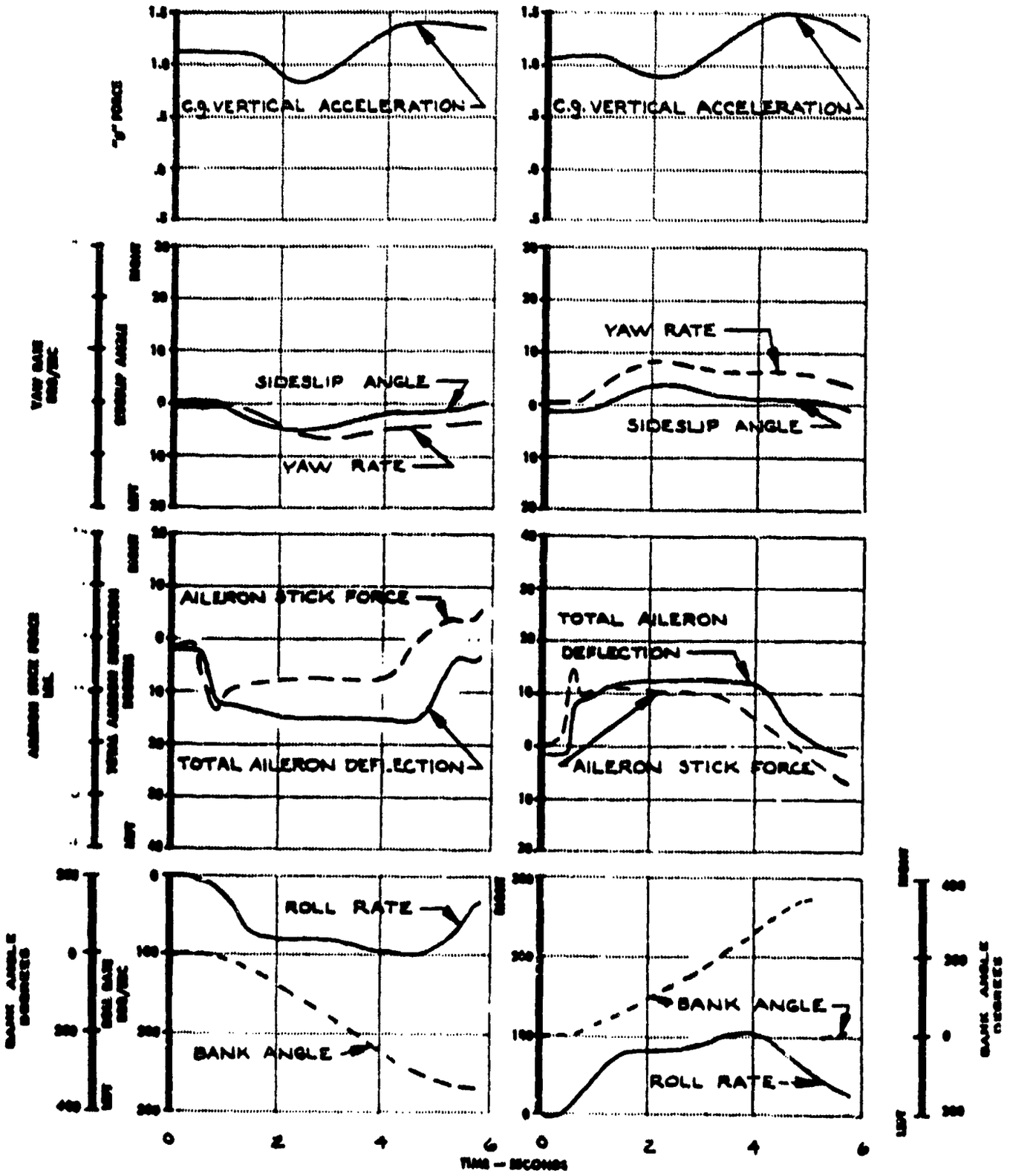
FIG. NO. 49
SIDESLIP CHARACTERISTICS
F-104A USAF No. 55-2955
CRUISE CONFIGURATION 10000 FEET
GROSS WEIGHT 16500 LB 398 KNOTS
RUDDER
AILERON
STABILIZER
CONTROL FORCES LBS
RIGHT PULL
LEFT PUSH
ANGLE OF BANK DEG.
LEFT
RIGHT
CONTROL POSITIONS DEG.
RIGHT UP
LEFT DOWN
AILERON
STABILIZER
RUDDER
LEFT
RIGHT
ANGLE OF SIDESLIP — DEG.

CONFIG CLEAN

FIGURE 50 AILERON ROLLS

TRIM CONDITIONS CAS 305 KNOTS RUDDER FIXED

ALTITUDE 8500 FEET MACH NO. .536



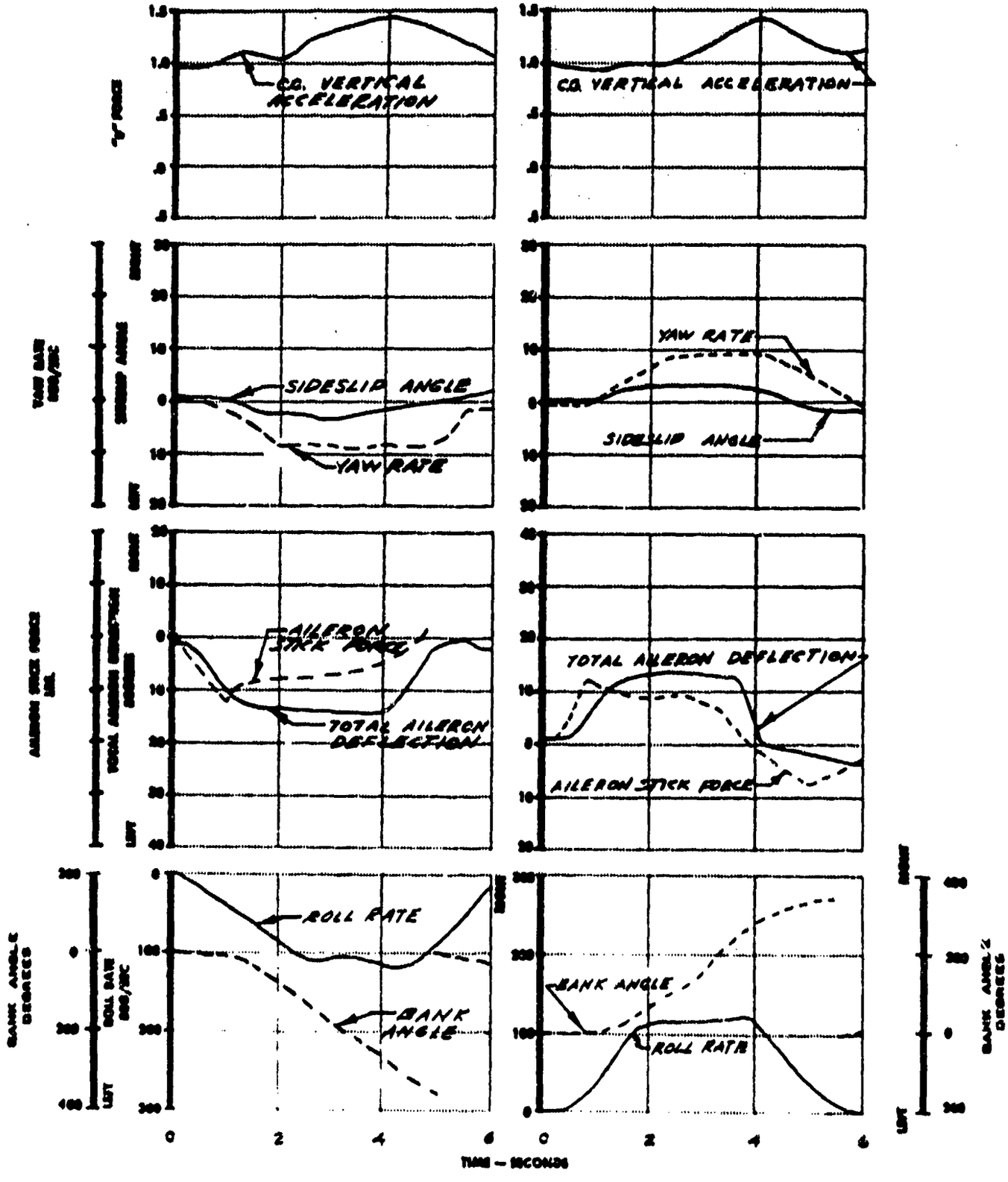
CONFIG. CRUISE

FIGURE 51 | AILERON ROLLS

TRIM CONDITIONS

CAS 280 KNOTS | RUDDER FIXED

ALTITUDE 39 500 FEET | MACH NO. .90



CONFIG. CRUISE

FIGURE 52 | AILERON ROLLS

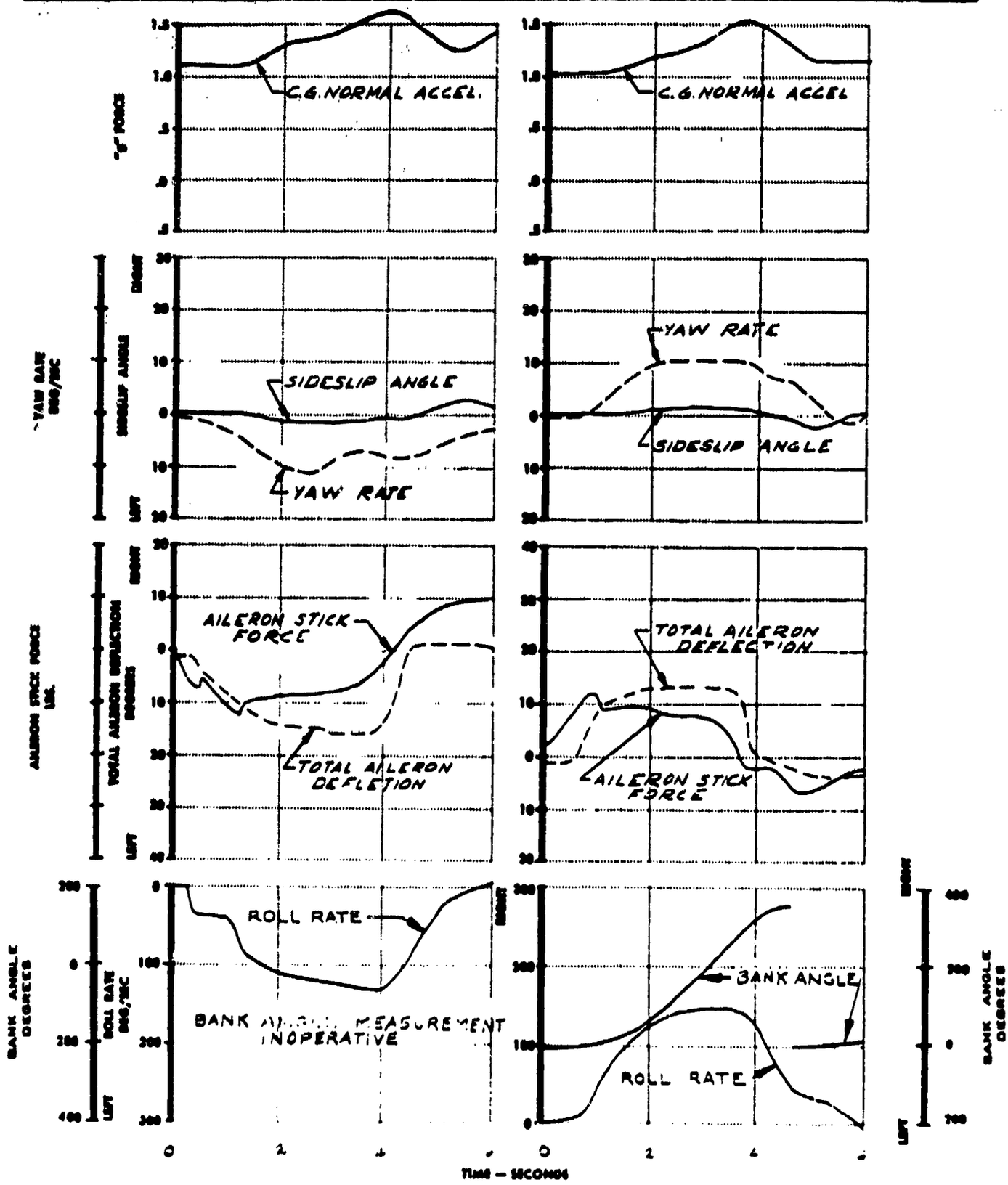
TRIM CONDITIONS

CAS 290 KNOTS

RUDDER PEDAL 1/3 DEFL

ALTITUDE 39000 FEET

MACH NO. .92



CONFIG. CRUISE

FIGURE 53 | AILERON ROLLS

TRIM CONDITIONS

CAS 290 KNOTS

RUDDER PEDAL 2/3 DEFL

ALTITUDE 38500 FEET

MACH NO. .912

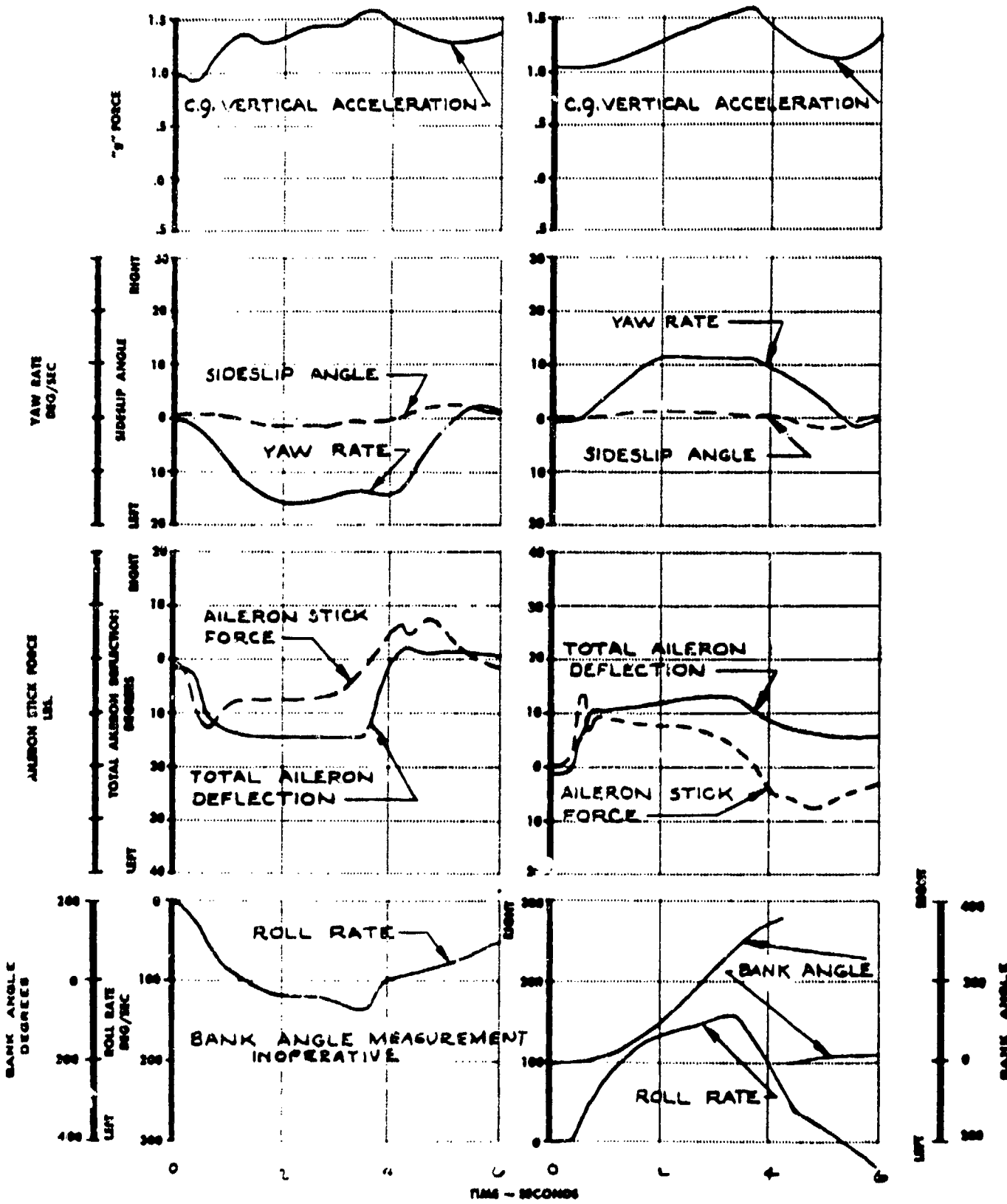
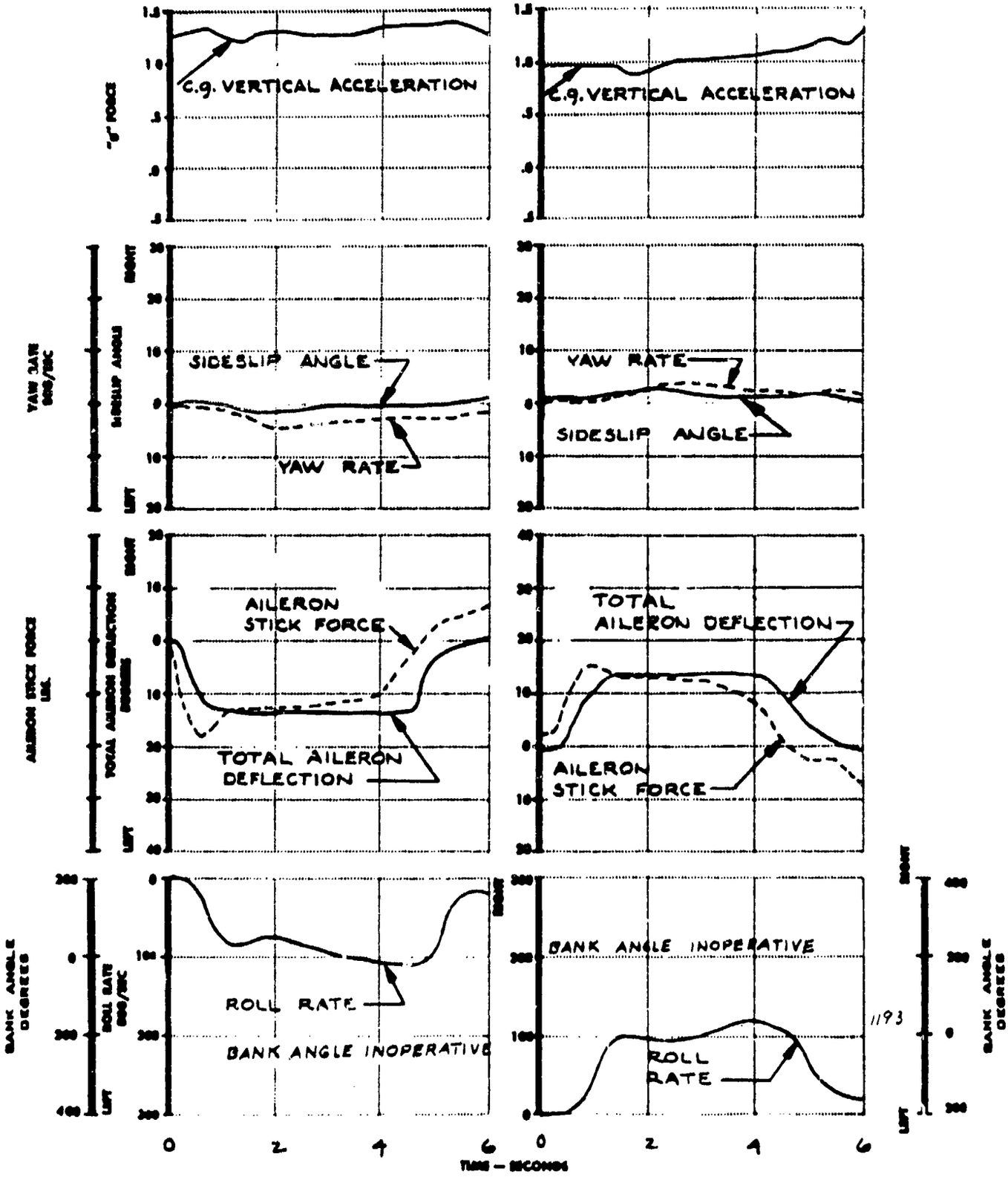


FIGURE 54 AILERON ROLLS

CONFIG CLEAN

TRIM CONDITIONS CAS 610 KNOTS RUDDER FIXED

ALTITUDE 40,000 FEET MACH NO 1.865



1193

FIG NO 55

AILERON ROLL CHARACTERISTICS

F-104A USAF NO 55-2955

POWER APPROACH CONFIGURATION RUDDER FIXED

V_c 160 KNOTS 10000 FEET

NOTES:

MAXIMUM STATIC AILERON DEFLECTION = 30°

ROLLS MADE FROM APPROXIMATELY 60° THROUGH 60°

DATA PLOTTED AT PEAK RATE OF ROLL

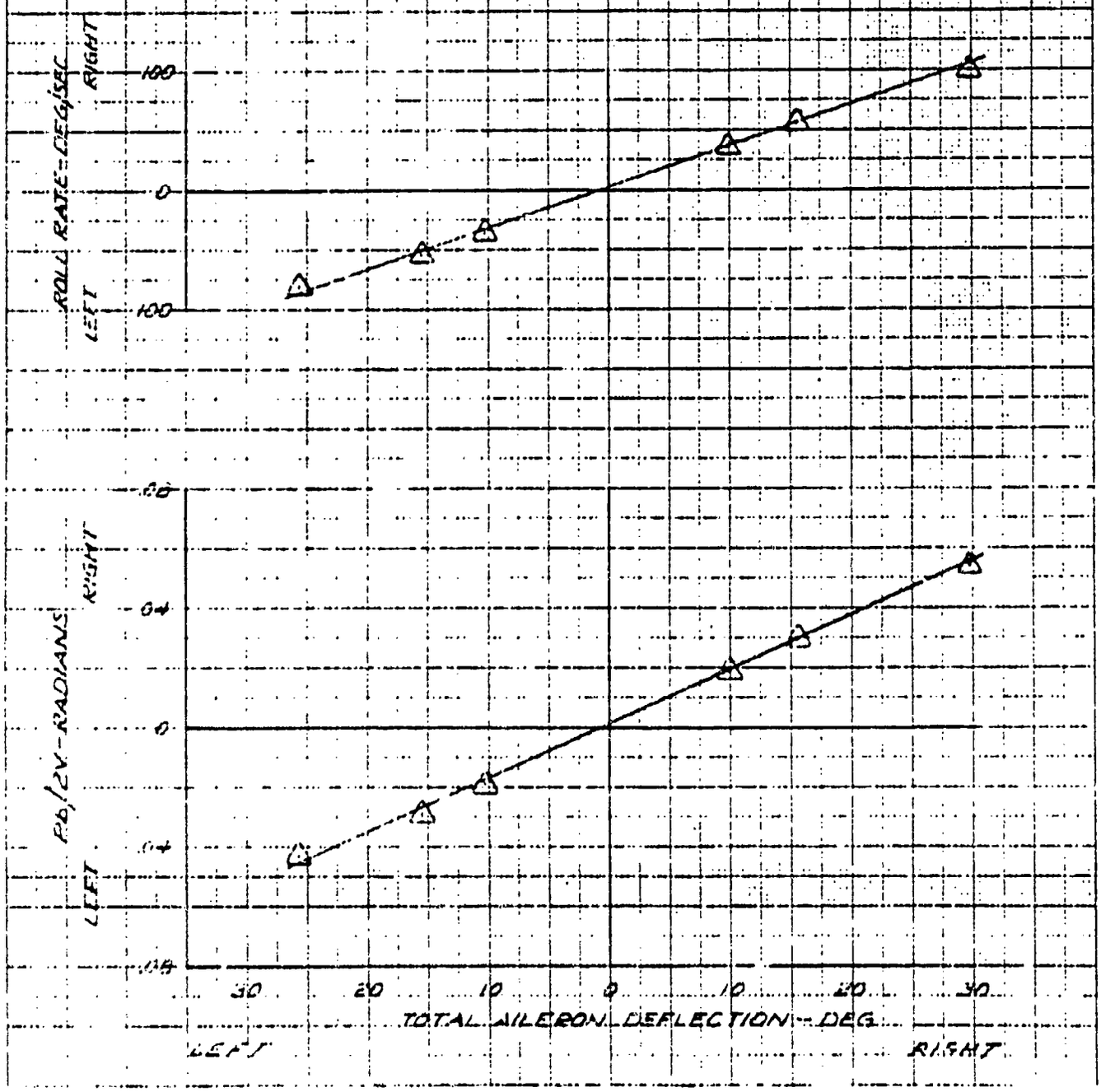


FIG NO 58

AILERON ROLL CHARACTERISTICS

F-104A USAF NO 55-2955

GEAR DOWN, TAKE-OFF FLAPS RUDDER FIXED

V_c 220 KNOTS 10000 FEET

NOTES:

MAXIMUM STATIC AILERON DEFLECTION ~30°

ROLLS MADE FROM APPROXIMATELY 60° THROUGH 60°

DATA PLOTTED AT PEAK RATE OF ROLL

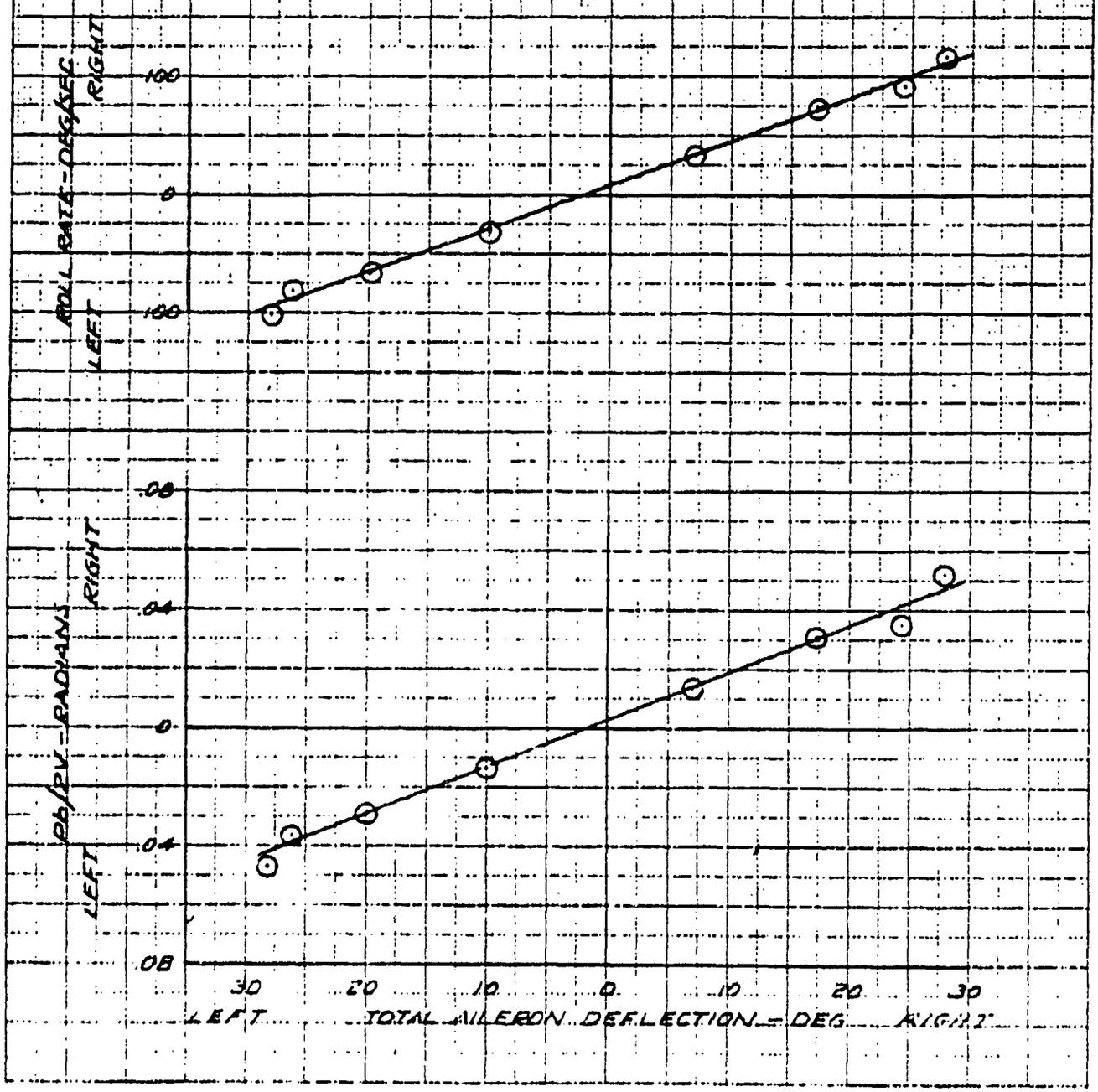


FIG NO 57

AILERON ROLL CHARACTERISTICS

F-104A USAF NO 55-2955

CLEAN CONFIGURATION RUDDER FIXED

10000 FEET

NOTES:

MAXIMUM ALLOWABLE AILERON DEFLECTION = 15°

ROLLS MADE THROUGH 360°

DATA PLOTTED AT PEAK RATE OF ROLL

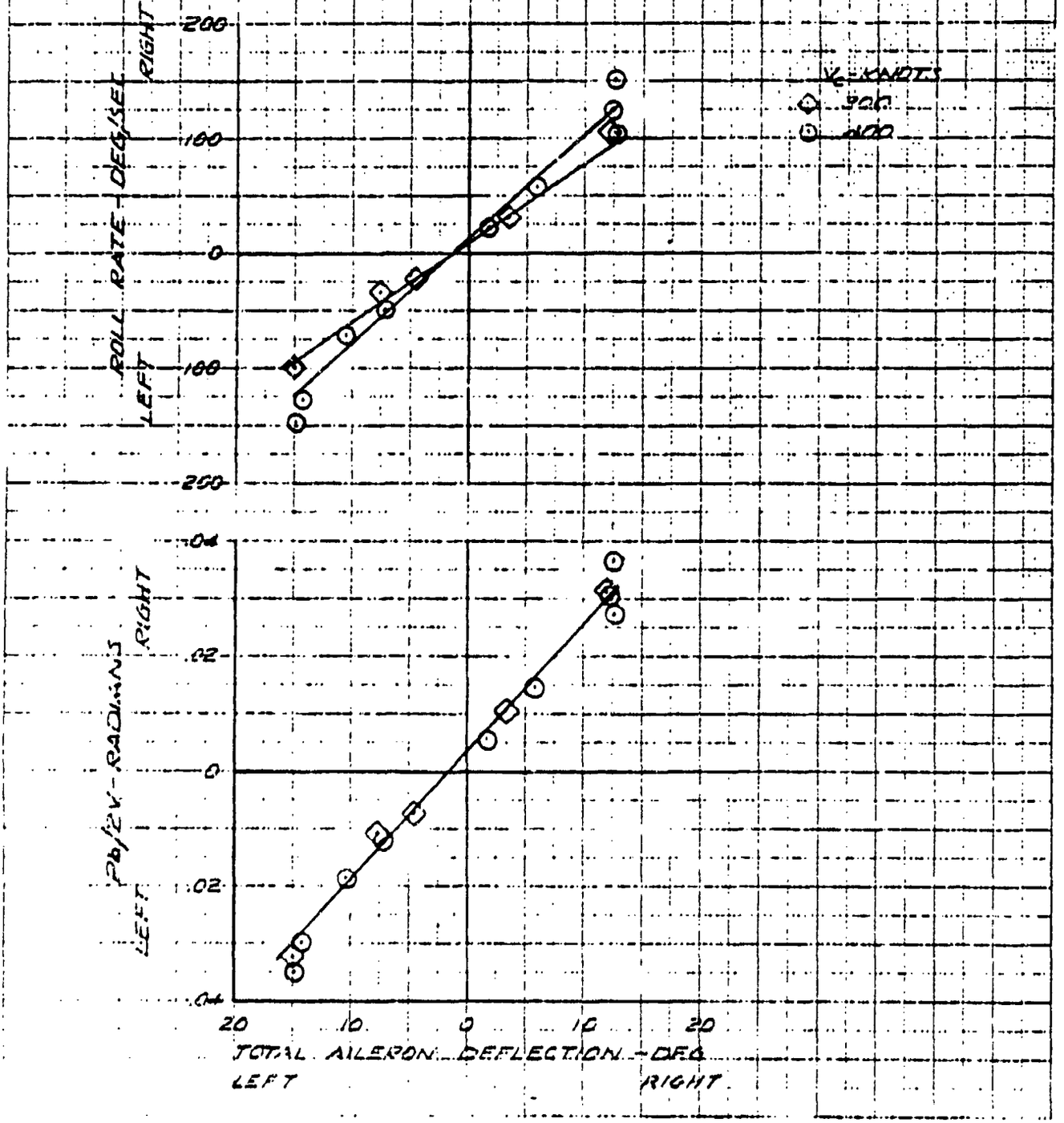


FIG NO. 59

AILERON ROLL CHARACTERISTICS

F-104A USAF NO 55-2955

CLEAN CONFIGURATION RUDDER FIXED

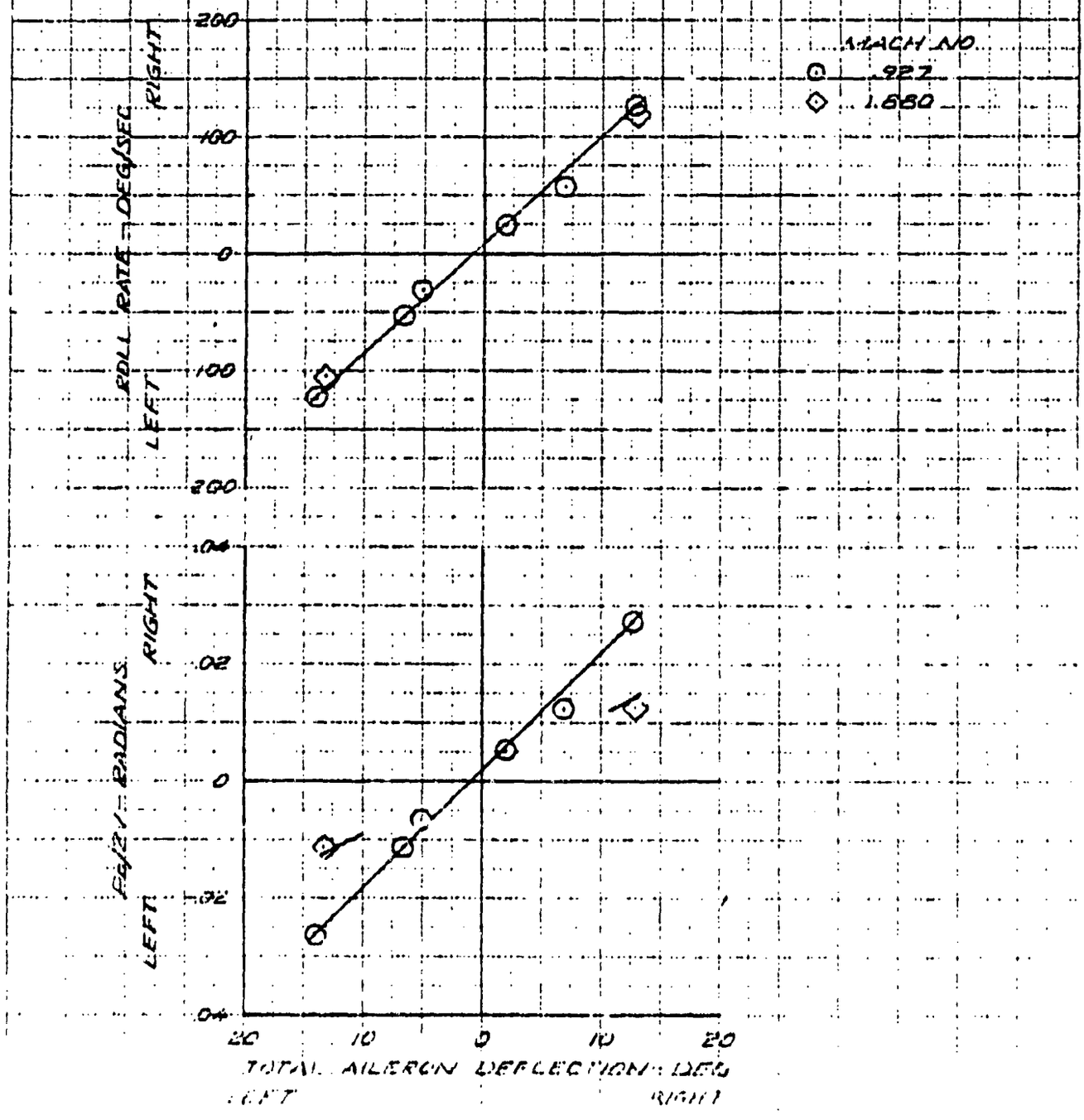
40000 FEET

NOTES:

MAXIMUM ALLOWABLE AILERON DEFLECTION ~15°

ROLLS MADE THROUGH 360°

DATA PLOTTED AT PEAK RATE OF ROLL



THIS PAGE LEFT BLANK FOR CONVENIENCE IN PRESENTING PLOTS

FIGURE 59 STALL TIME HISTORY

CONFIG POWER APPROACH

TRIM CONDITIONS

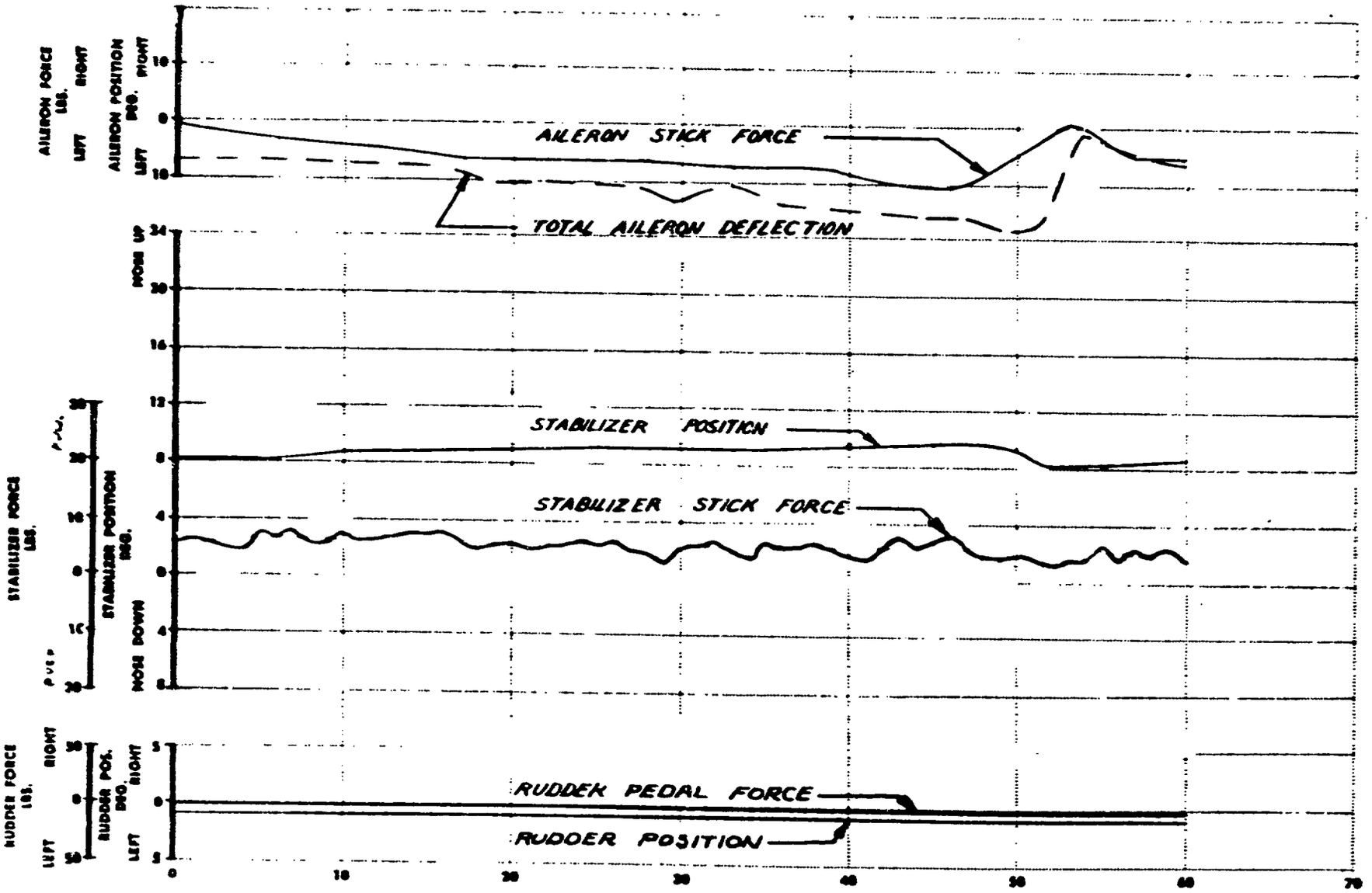
CAS 180 KNOTS

CG 12.6 %MAC

ALTITUDE 30,000 FEET

WEIGHT 19,910 POUNDS

STABILIZER 2.0 T.E. UP



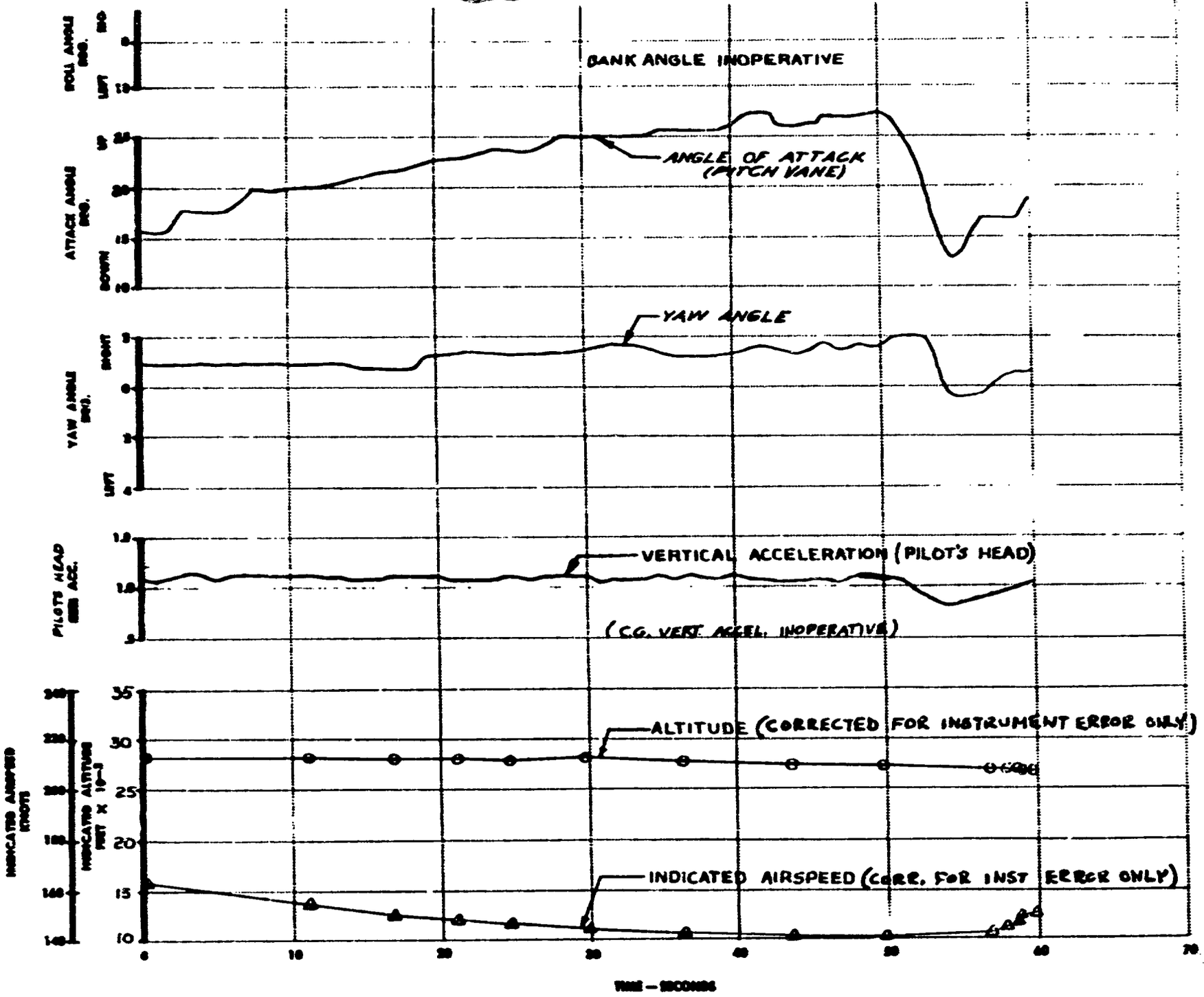
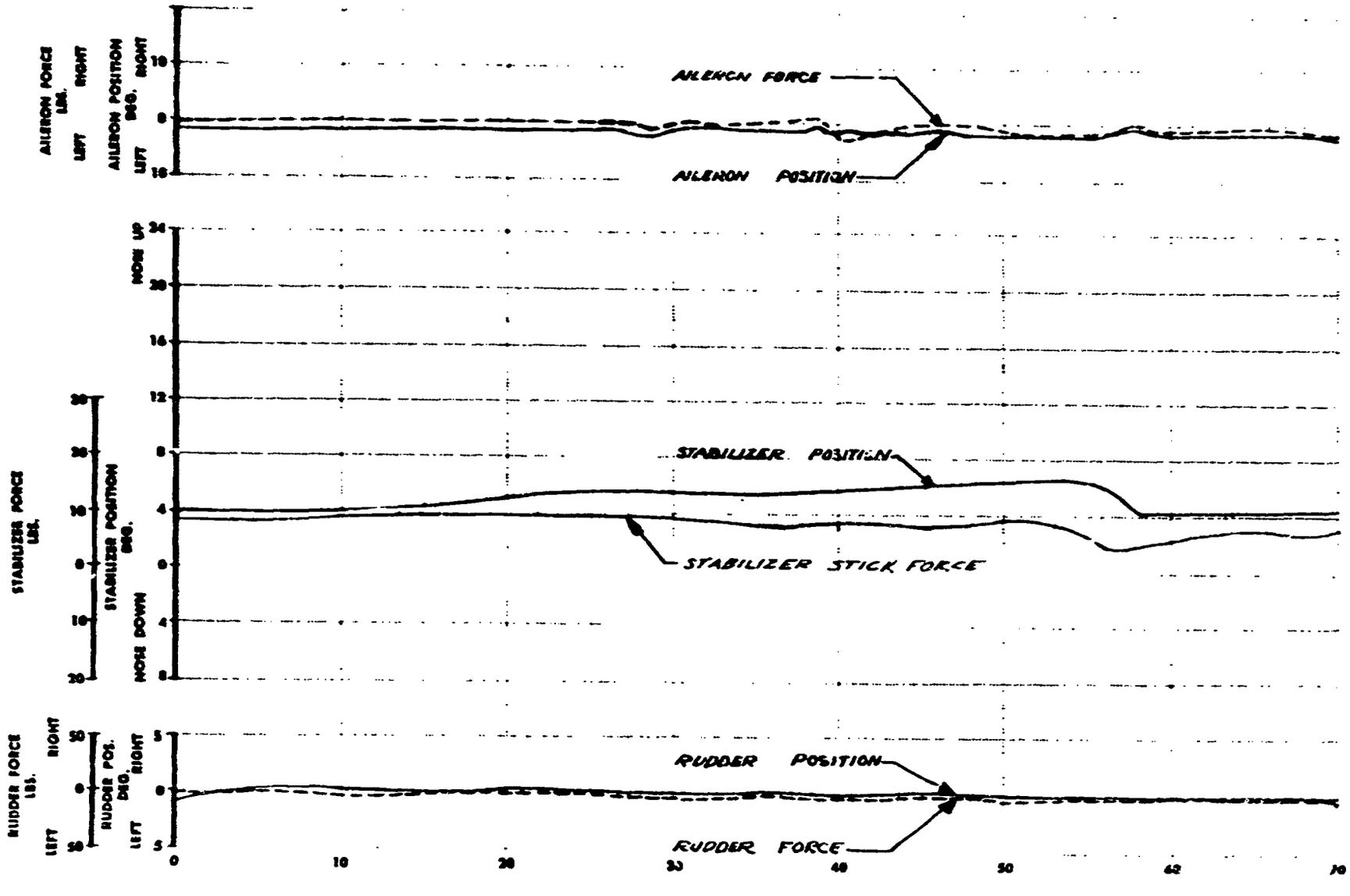

BANK ANGLE INOPERATIVE
ROLL ANGLE DEG.
RIGHT
LEFT
ANGLE OF ATTACK (PITCH VANE)
ATTACK ANGLE DEG.
UP
DOWN
YAW ANGLE
YAW ANGLE DEG.
RIGHT
LEFT
VERTICAL ACCELERATION (PILOT'S HEAD)
PILOTS HEAD VERT ACC.
(C.G. VERT. ACCEL. INOPERATIVE)
ALTITUDE (CORRECTED FOR INSTRUMENT ERROR ONLY)
INDICATED AIRSPEED (CORR. FOR INST ERROR ONLY)
INDICATED AIRSPEED KNOTS
INDICATED ALTITUDE FEET X 10-3
TIME — SECONDS

FIGURE 60 STALL TIME HISTORY

CONFIG. CLEAN

TRIM CONDITIONS CG 11.4 %MAC WEIGHT 15150 POUNDS

CAS 336 KNOTS ALTITUDE 30640 FEET STABILIZER 1° T.E. UP



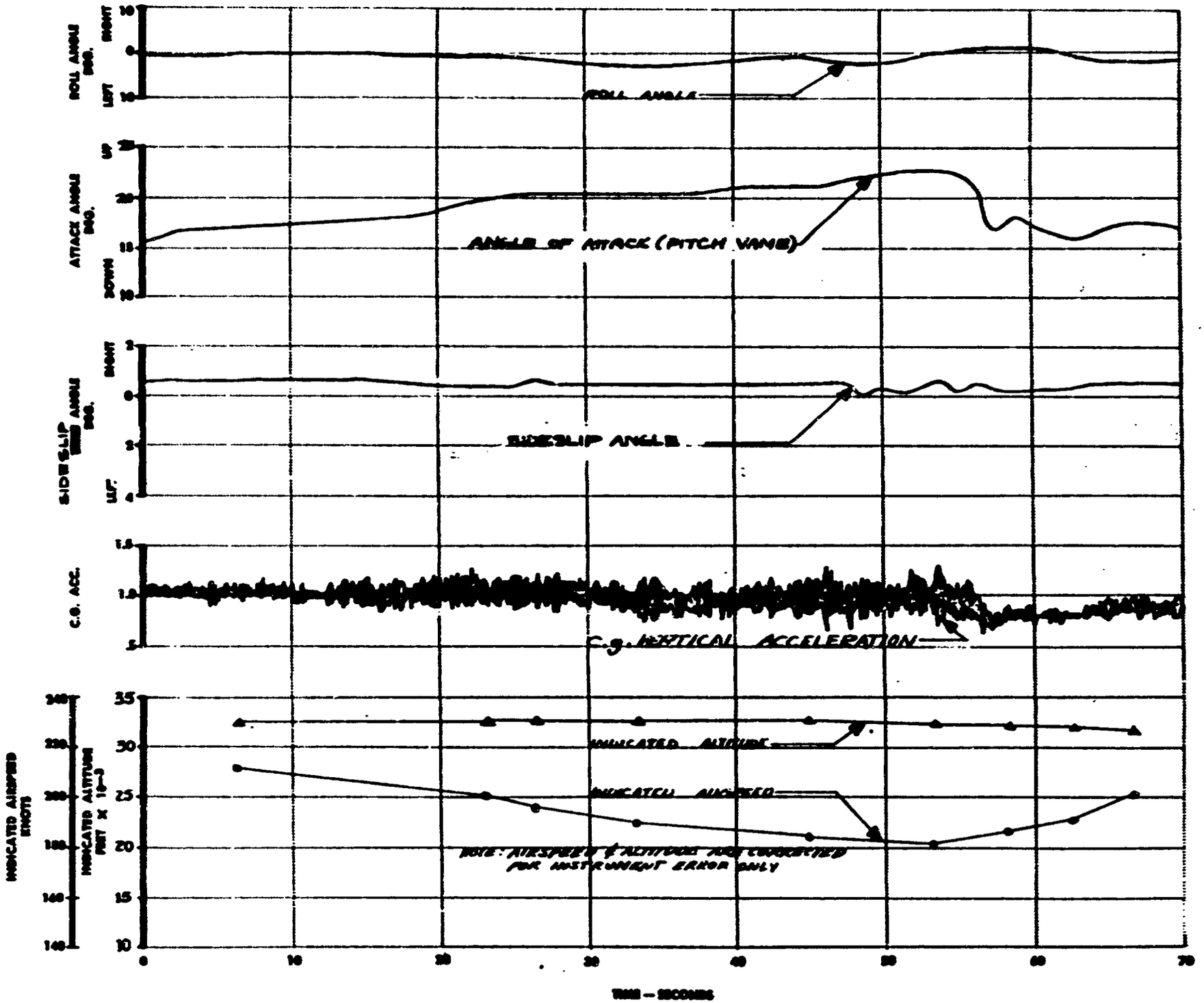

ROLL ANGLE DEG.
RIGHT
LEFT
ROLL ANGLE
ATTACK ANGLE DEG.
UP
DOWN
ANGLE OF ATTACK (PITCH VANE)
SIDESLIP ANGLE DEG.
RIGHT
LEFT
SIDESLIP ANGLE
C.G. ACC.
C.g. VERTICAL ACCELERATION
INDICATED AIRSPEED KNOTS
INDICATED ALTITUDE FEET X 10^-3
INDICATED ALTITUDE
INDICATED AIRSPEED
NOTE: AIRSPEED & ALTITUDE ARE CORRECTED FOR INSTRUMENT ERROR ONLY
TIME – SECONDS

FIG NO 6

AILERON CONTROL FRICTION

E-104A USAF NO 55-2965

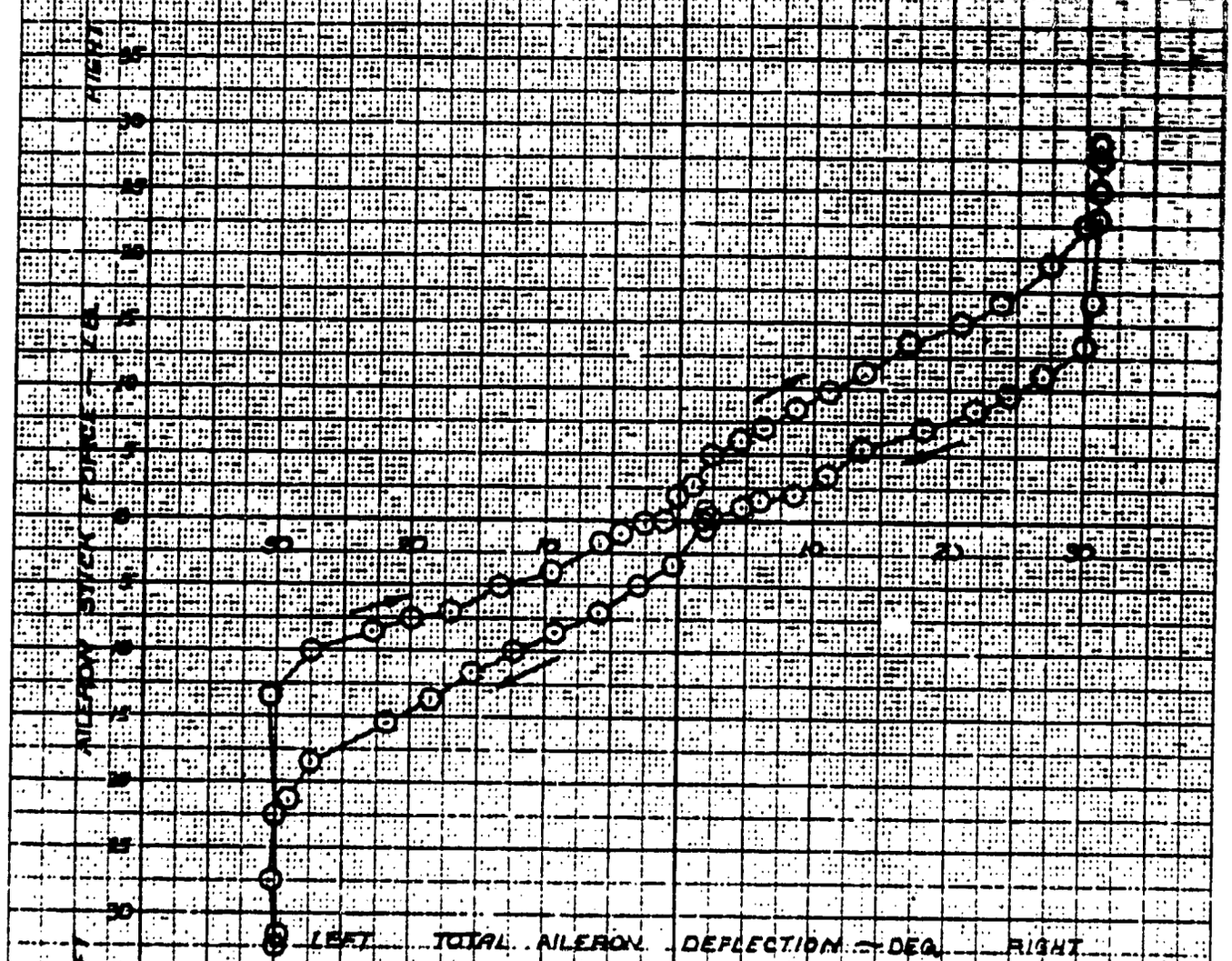


FIG NO 62

STABILIZER CONTROL FRICTION

F-104A USAF NO. 55-2955

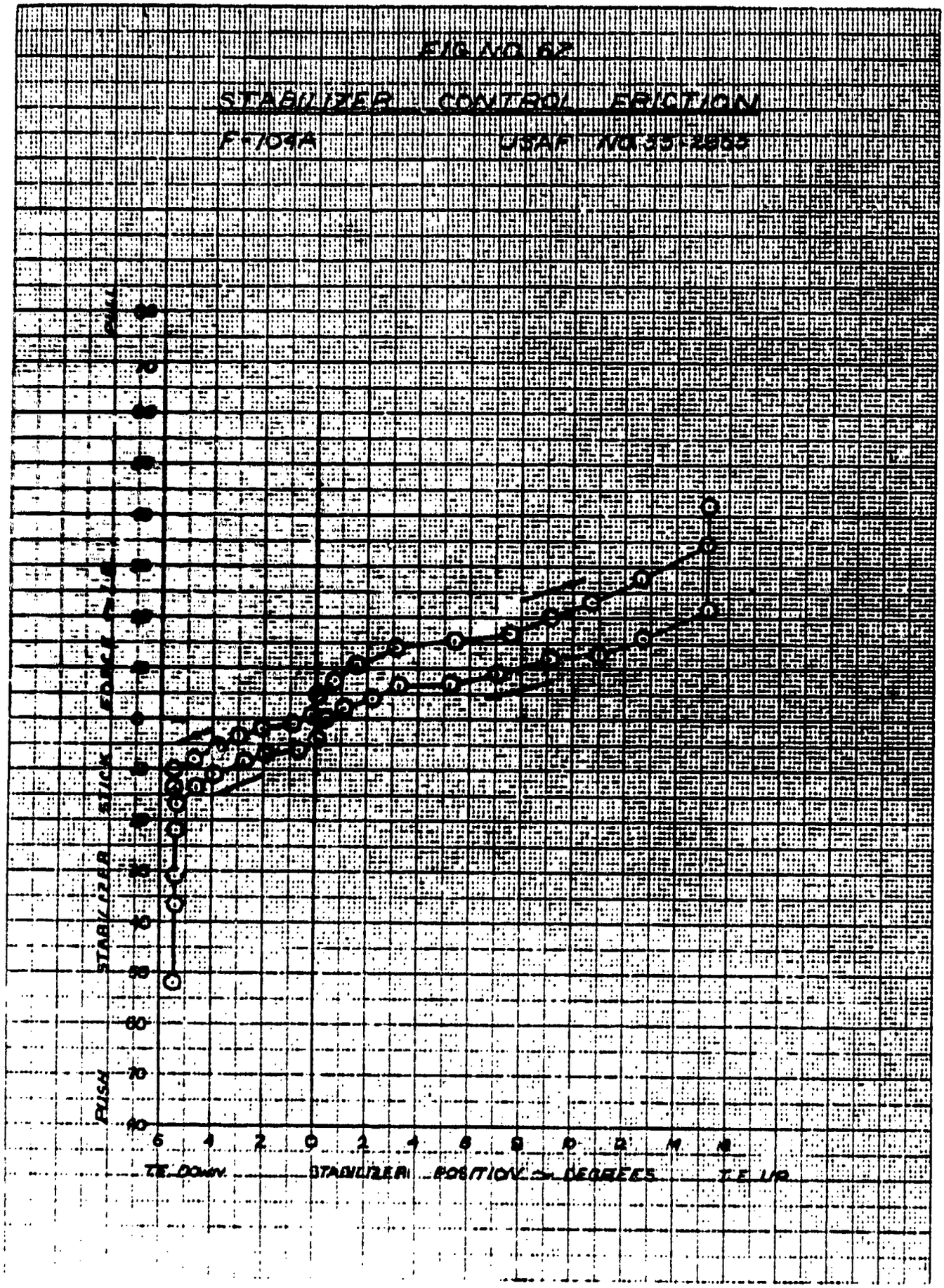
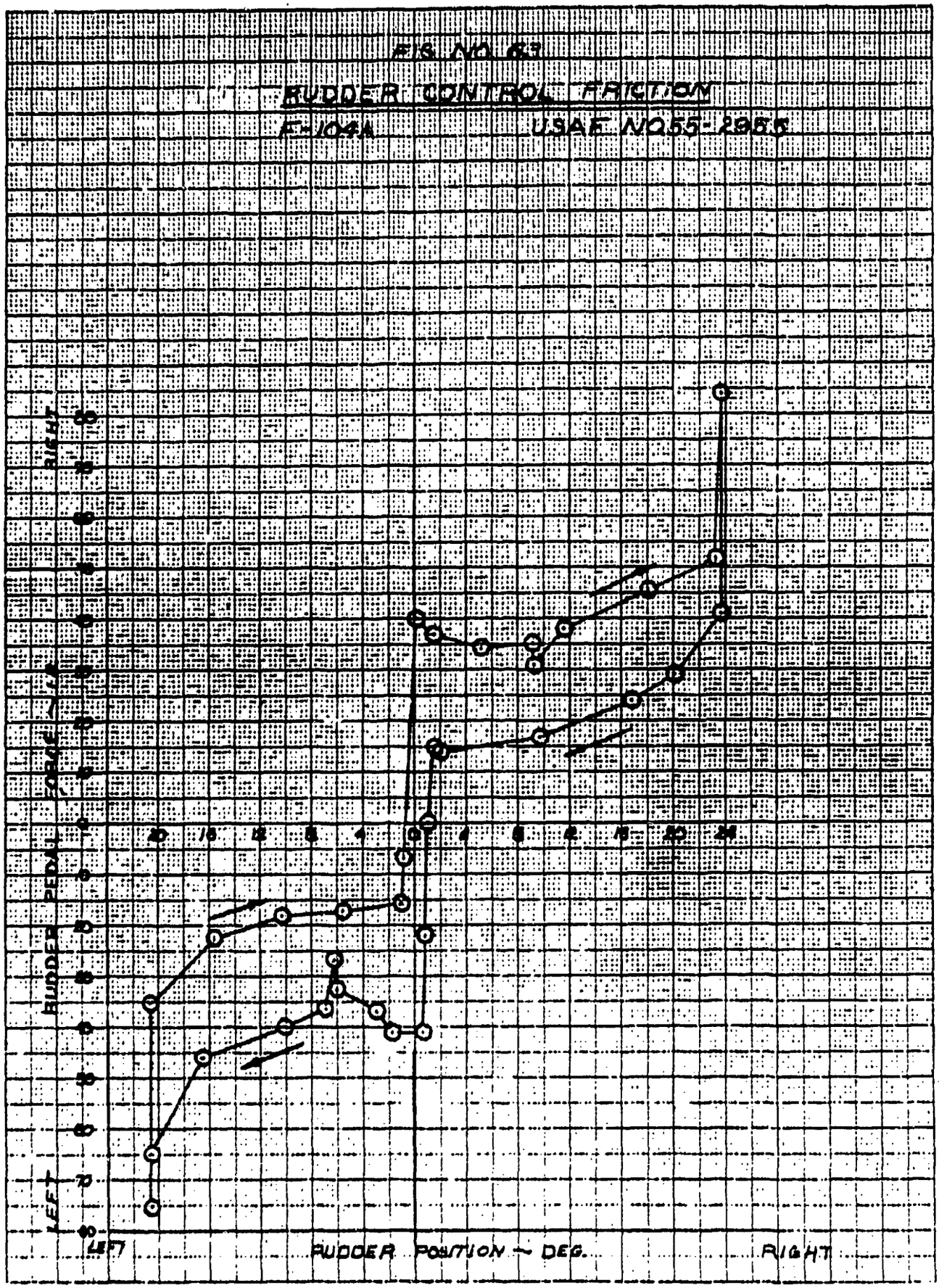


FIG. NO. 63

RUDDER CONTROL FRICTION

F-104A USAF NO. 55-2955



APPENDIX II
general aircraft information

dimensions

GENERAL DIMENSIONS:

Length	54.77 ft.
Height	13.49 ft.
Span	21.94 ft.

WING:

Area	196.1 sq. ft.
Span	21.94 ft.
Aspect ratio	2.45
Taper ratio	0.378
Sweepback	18.1 at 25%
Dihedral	— 10°
Incidence, root and tip	0°
Airfoil section	Modified bi-convex 3.4% thick

AILERONS:

Area, total	9.46 sq. ft.
Deflection	± 15°

TRAILING EDGE FLAPS (PER SIDE):

Area	11.55 sq. ft.
Average chord	30.2 in.
Deflection limit	45°

NOSE FLAPS (PER SIDE):

Area	8.50 sq. ft.
Mean chord	12.14 in.
Deflection limit	30°

SPEED BRAKES:

Area (projected)	8.25 sq. ft.
Deflection limit	60°

FUSELAGE:

Frontal area	25.0 sq. ft.
Length	615 in.
Fineness ratio	9.09

HORIZONTAL TAIL:

Area	48.2 sq. ft.
Deflection limit	
control	+ 5 to — 17°
trim	+ 2 to — 11°

VERTICAL TAIL:

Area, total	35.1 sq. ft.
Area, rudder	4.3 sq. ft.
Rudder deflection	± 25°
Area, yaw damper	1.00 sq. ft.
Yaw damper deflection	± 20°

operational limitations existing during phase II tests

V-G LIMITS:

$C_{L_{max}}$ reduced 20 percent above .9 Mach number

Load factor of 5g pending structural integrity tests

AILERON ROLLS:

One g and 42 percent aileron deflection

LIMIT SPEEDS:

Over 40,000 feet	2.0 Mach number
30,000 feet	1.57 Mach number
20,000 feet	1.24 Mach number
10,000 feet	0.96 Mach number

LANDING GEAR EXTENDED:

Operating	260 knots IAS and 3.0g
Locked down	295 knots IAS and 3.0g

WING FLAPS EXTENDED:

15° – 15°	295 knots IAS and 3.0g
30° – 45°	240 knots IAS and 3.0g

COMPRESSOR INLET TEMPERATURE:

121°C

CG LIMITS:

2 to 20% MAC

power plant

ENGINE:

Manufacturer: General Electric

Designation: YJ79-3

ENGINE RATING:

	% RPM	Net Thrust – lb.	Net SFC lb/hr-lb.
Maximum	100	15,600	1.972
Military	100	10,000	0.840

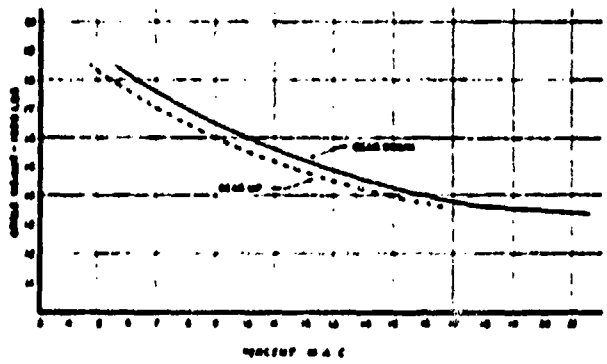
weight and balance

WEIGHT:

Basic weight, including instrumentation, ballast, oil, and residual fuel	13,291 lb.
Pilot	250 lb.
Fuel (762 gallon at 6.5 pound per gallon)	4,95[illegible] lb.
Gross weight at engine start	18,49[illegible] lb.

BALANCE:

The graph below shows the relationship between cg position and weight.



instrumentation

Test data was recorded by means of a photo-recorder and an oscillograph located in the electronic compartment. Instrumentation used during these tests is noted below.

PHOTO-RECORDER

- Altimeter
- Airspeed
- Tachometer
- Free air temperature
- Exhaust gas temperature
- Vertical accelerometer
- Stop watch
- Oscillograph counter
- Fuel totalizers
- Exhaust nozzle area
- Power lever position
- Turbine discharge total pressure
- Fuel temperatures
- Heading

OSCILLOGRAPH

- Stabilizer position
- Stabilizer stick force
- Left and right aileron positions
- Aileron stick force
- Rudder position
- Rudder pedal force
- Angle of attack
- Angle of sideslip
- Angle of bank
- Vertical accelerometer
- Roll rate
- Yaw rate
- Pitch rate
- Engine fuel flow
- Afterburner fuel flow

APPENDIX III

human factors evaluation

An evaluation of the aircraft was made by personnel of the Human Factors Branch. The report which was prepared as a result of this evaluation makes up the contents of this Appendix.

Introduction

A limited evaluation of the cockpit design features of the F-104A was conducted throughout the Phase II Flight Test Program. As a result of this evaluation there have been a number of deficiencies noted varying in degree of importance. Isolation of the difficulties was made through pilot comment and through independent investigations. The cockpit evaluation must of necessity be considered incomplete in that many of the production model instruments and essential controls, such as the radar on the armament panel, were not installed or utilized in the Phase II aircraft. This commentary will therefore cover two categories of items, i.e., evaluation of deficiencies occurring as a result of standard installations, and those where a comparison of production aircraft indicates a potential problem area based on previous flight test experience and principles of human engineering. It should also be noted that a number of the deficiencies outlined below are presently being considered for corrective action in later aircraft; however, the purpose of this report is to point out and record actual and potential areas where emphasis should be placed to improve simplicity and efficiency in the operation of the aircraft by the pilot.

general evaluation

Cockpit Accessibility: The principle of simplicity and accessibility is well illustrated in the F-104A cockpit; particularly when an evaluation is made on a comparative basis. Although the full range of pilot sizing was not encountered during Phase II testing, the pilot was near the median in physical size and reportedly experienced no difficulty in reaching all controls with or without shoulder harness locked. With the exception of a few circuit breakers all console control panels and essential flight controls are forward of the seat neutral reference point. Similarly, all forward controls and panel areas are within easy reaching distance. With one minor exception that being the snap-up standby compass, there should be no difficulties encountered as a result of the linear distance of controls from the pilot's position.

Emergency Escape System: The F-104A is presently equipped with a model B-2 type ejection seat. Sled runs wherein this type of seat has been experimentally ejected have resulted in the disintegration of the seat at speeds in the range of 600 knots. Studies of the physical tolerances of the human being indicate that the present ejection seat assembly installed in

Flight Test F-104A aircraft, and contemplated for the Phase IV and VI aircraft, is not adequate to provide for emergency escape by the pilot throughout the full performance range. This condition has been recognized by the Air Force and the contractor and improved ejection seat assemblies are scheduled for later production models. Pilots using the B-2 type seat should be fully appraised of its limitations. Introduction of improved emergency escape equipment should be given high priority.

The safety pin installed in the initiator between the foot guides on the front of the ejection seat provides for a ground safetying feature. However, it has been noted that in other aircraft with pins in this position, the warning streamer frequently falls beneath the seat structure which tends to mitigate its warning function and increases the possibility of the safety pin being inadvertently overlooked. This was prevented during Phase II testing by the addition of a bungee cord on the loose end to allow the streamer to be looped over the end of the control stick. This is a local Lockheed modification; however, it has merit and should be applied to safety pin-warning flag assemblies employed in this aircraft.

The parachute support filler utilized in the seat pan in connection with the MC-1 aircraft cushion does not appear to be adequate in two areas. Past experience indicates that the support ledge will be too low to effectively remove the weight of the parachute from the back and shoulders of the pilot, especially with the B-5 type chute. Examination of the filler item also indicates that the plastic material rapidly degenerates under wear encountered from flight or ground maintenance activities. The introduction of complete seat cushion survival kit assemblies into the seat pan may eliminate the need for this filler in later model aircraft. Until that time it should be made adequate to meet its purpose.

The ejection seat is equipped with a lap belt tie down strap installed on the forward part of the seat pan. The mechanism for adjustment of this strap is almost completely inaccessible to the pilot once he is positioned in his seat. The adjustment device is located below the "D" ring attachment fittings and also behind the foot guides. Unless the pilot is able to make the necessary fitting of this strap the lap belt will not provide the expected bodily restraint against accelerative forces. A potential method of simplifying the adjustment of this tie down strap is noted in the recommendations.

The canopy design of this aircraft does not incorporate an explosive jettison device for emergency *external* removal of the canopy. Provisions for external rescue entrance is made through a manual external canopy release mechanism on the right side of the fuselage. No provision is made for entrance on the left of the fuselage. The existing external release mechanism is identified with lettering indicating its location, and method and direction of operation. These word markings are yellow in color and while sufficient to indicate the nature of the control, are not adequate to immediately alert rescuing personnel to the location of the only available external release on the right side.

Pilot's handbooks normally contain a section on emergency procedures which includes such data as the ejection sequence, limiting speed, lap belt

separation times, and general caution data. The performance of this aircraft and the limitations on ejection imposed by the present ejection system requires that greater detail be developed regarding potential flight conditions of the aircraft wherein ejection would in all probability be extremely hazardous or unsuccessful. Graphic data should be included in the handbook to illustrate "time to go" and altitude factors which will affect a pilot's decision to abandon the aircraft or remain until a lower speed is reached. Data should be based on aircraft in uncontrolled dives at various angles, drag inducing conditions, power situations, etc., from various initial altitudes. This data should then be combined with required seat ejection times, lap belt and seat separation times to provide the pilot with realistic conditions where emergency separation from the aircraft may become necessary.

- *Oxygen Equipment:* The F-104A is presently fitted with an MB-2 type high pressure oxygen regulator. The panel for this installation occupies a relatively large portion of the right forward console area. The introduction of liquid oxygen equipment into subsequent aircraft and the use of integrated oxygen equipment will result in the elimination of a need for this regulator. Future regulators will be required to deliver 100% oxygen at all times to meet the pressure needs of improved pressure helmets and as a result many of the features of the present regulator are no longer satisfactory. Removal of this panel and the introduction of a 3-inch panels containing an oxygen quantity gauge and essential control valves will provide additional console space in this area.

  Present oxygen provisions, as regarding emergency oxygen and personal leads, are not compatible with the new concepts of integrating this equipment into the seat cushion-survival kit assembly. There is a requirement to incorporate the oxygen regulator into the seat cushion-survival kit assembly. Personal leads to the pilot, (oxygen and suit pressure), as well as communications and face plate heat should be routed into the survival kit and integrated with personal leads from the partial pressure suit, pressure helmet, and emergency oxygen cylinder.

  It is noted that the lead from the ship's oxygen supply to the pilot's mask is located to the left rear of the pilot. When the pilot employs a pressure suit with this arrangement the hose must of necessity cross the body to the manifold fitting on the right of the body. As the contractor has the alternative of placing this connection on either side of the seat there seems to be little argument for retaining it in a position where as it crosses the body it will add to the already heavy clutter of straps and hoses on the pilot. It would seem self evident that with the introduction of fittings for the MC-4 Partial Pressure Suit simplicity and utility of personal leads will be improved if connections are on the right of the pilot.

  Flights during Phase II testing were made with a modified MC-1 partial pressure suit. Missions of one hour and fifteen minutes were completed with an indication that adequate oxygen was available. The oxygen duration chart in the handbook would tend to indicate that sufficient quantities of oxygen will be available, particularly since the addition of more bottles which increases the supply to approximately 29.7 cu ft of available free gas at sea level under standard temperature conditions. However, experiment in altitude chambers using the MC-4 type pressure suit indicates that this equipment will create a greatly increased oxygen consumption rate when compared to that normally experienced with standard equipment. For example, at ground level the consumption rate using the MA-2 pressure helmet (normally used with the MC-4 partial pressure suit was 54.0 cu ft per hour; at 10,000 ft, 32.0 cu ft per hour; at 25,000 ft, 15 cu ft per hour. As the cabin pressurization schedule indicates cabin altitude will not generally exceed 25,000 ft, under normal conditions the oxygen consumption rate will always average greater than 15 cu ft per hour, especially when oxygen is used at ground level. Thus, even with the addition of high pressure bottles to increase the available oxygen to its present quantity, the period during which the pilot can anticipate having adequate oxygen will be greatly reduced when improved altitude protective equipment is employed.

- *Controls:* The drag chute release handle is located on the forward instrument panel slightly to the right and forward of the throttle area. Examination of this installation indicates that there is marginal clearance (a maximum of 1¼ inches) between the left vertical console and the back of the "T". This restricted access to the handle would make it difficult to secure a firm grip through a simple straightforward grasping motion, particularly if pressure gloves were utilized. It is also noted that identification of the drag chute handle as outlined in ARDCM

80-1 has not been met as regards the requirement for symbolic shape coding. This identification as well as a color contrast between the gray of the control and the gray background paneling is more of a requirement because of the relative location of the manual gear release "T" handle located adjacent to the drag chute handle.

The pilot reportedly encountered difficulty when utilizing the wheel brakes. The problem is associated with two different conditions. The simple hydraulic brakes installed in the aircraft first of all require that the pilot employ a relatively high pedal force to secure the braking action necessary. This fact in itself will require a period of familiarization for pilots whose past experience has been more closely associated with power type brakes. In addition, the pilot is required to "pump" the brakes once or twice to secure full braking action and equalize braking forces. When such action is necessary during a ground roll where directional control is essential or where immediate braking is required it may produce potentially hazardous conditions.

Immediately above and forward of the landing gear control is a button labeled, "landing gear down lock override". This switch functions as an emergency override to allow the pilot to raise the gear with the weight of the aircraft still on the wheels. The location of the button is appropriate to its function; however, the miniaturized size of the button and the potential interference that will occur from the raised portion of the vertical console adjacent to it as the pilot attempts activation indicates that minor design changes may be required. A pilot utilizing a pressure glove would probably find it difficult to positively and cleanly activate this button during the potential emergency conditions when ground gear retraction would be essential.

Directly under the main instrument panel and on either side of the armament panel are located a total of four manually activated controls whose functions are such as to categorize them as emergency controls. These are: Landing Gear Manual Release Handle, Ram Air Turbine Extension Handle, Pylon Tank Manual Jettison Handle, and Tip Tank Jettison Handle. At the present time the color of these controls is a standard grey which does not provide the identification or contrast essential for emergency type controls. A standard orange-yellow and black striping should be used to more clearly outline and identify the controls. This is also true of the Hatch Jettison Handle located on the right side of the armament panel.

- *Instruments:* Several of the flight instruments received general criticism as to location and quality of the information presented to the pilot. A comment is also included on one indicator which is scheduled for production installation, but which was not utilized in the Phase II aircraft.

The ME-2 air speed indicator was not felt to be satisfactory primarily because of the "cluttered" nature of the air speed scales, the change in IAS scale from the outer scale to the inner scale at 200 knots, and the difficulty encountered in securing simplified check readings of air speed with minor increments of error. It is noted in the pilots' comments that during approaches and base legs critical speeds are in the scale area where there is a conversion of scale values making it difficult to "peg" a particular value.

The Phase II aircraft contained instrumentation which placed the Critical Inlet Temperature gauge to the left of the primary flight group on the forward instrument panel. This position was felt to be advantageous as compared to the scheduled position on the lower right forward instrument sub-panel. The function of this gauge is closely associated with air speed and its location near the primary flight group would appear to be essential. It is noted that a change in the location of this instrument in production aircraft to the more preferred location has been submitted as an ECP. Some further evaluation may be necessary to determine the functional utility of the information presented by this indicator, especially when a flashing light on the forward instrument panel signals when the temperature limit has been reached. In the event the pilot does not require data which reflects an increase or decrease in actual inlet temperature and needs only to know that he is operating within limits it may be possible to eliminate the instrument and rely only on the "checking" nature of the flashing light signal.

The 2-inch accelerometer on the lower forward sub instrument panel was criticized primarily because of the readability of the scale and the inherent error of the instrument. Past experience in high performance aircraft and flights during Phase II testing indicated that the accelerometer is becoming a prime instrument in many phases of flight. "G" loadings during climbs, turns, etc., are critical in many instances and a more refined presentation

of this type data to the pilot becomes more essential. An improved 3-inch accelerometer is known to be available and would represent a considerable improvement in dial legibility and instrument error over the present indicator.

The flip-up stand-by compass located to the pilot's left, forward of the glare shield, was considered slightly beyond reach of the pilot; however, this is not considered critical as it can be reached with slight effort or slight release of the shoulder harness. It represents the only item where accessibility to the medium sized pilot was marginal.

The Radio Magnetic Indicator is to be utilized in the flight group as a primary directional reference. This instrument has previously been reported as inadequate for aircraft to be utilized under a ground controlled intercept operation. The indicator presentation is so arranged that the heading is read from a moving scale against a fixed index at the top of the instrument; thus check reading of the heading as with a moving pointer indicator is not possible. Turning to a desired new heading with the RMI causes the pilot to first decide which direction he must turn (often times turns are started in the wrong direction because of the ambiguous presentation) and secondly to monitor the moving scale very closely in order to stop the turn at the proper heading. Utilizing an instrument which employs a moving pointer indicator allows the pilot to merely set his desired heading at the top and fly the pointer to a vertical position. Such a capability is a great convenience in combat/defense operations under GCI operation since the pilot is relieved of the task of remembering the directed heading.

- *Warning Lights:* The word warning panel on the right vertical console contains individual caution indicators which are activated whenever a particular condition is to be called to the attention of the pilot. Signals provided by these indicators are more representative of actions which do not necessarily require immediate action by the pilot. At the present time the mask and/or light for this panel is red in color which is a color more generally reserved for signals more critical in degree of significance. The word warning panel color should therefore be corrected to reflect a more accurate indication of the nature of the signal. Pilot comment on the intensity of the lighting on this panel indicates that some adjustment may be required to provide for proper signal strength during unusual lighting conditions, particularly during day light operation.

The "Master Caution" light on the forward instrument panel also presently employs a red mark and/or light whereas the nature of the signal requires that an amber color be associated with this type of signal. Corrective action will require that the color of the light or mask be changed as well as making possible adjustments in the intensity of the signal.

The landing gear warning light panel is so designed that a relatively small green mask is seen by the pilot when the lights are activated. Normally, a "pinpoint" light requires a greater intensity to provide the same stimulus that a light with more area would provide. Under certain lighting conditions the presence of a light with a larger area would be more discernible and alerting to the pilot.

- *Miscellaneous:* Circuit breakers on both the left and right consoles are difficult to identify when they have "popped". The consoles are inclined at an angle of approximately 30° which places them so that they are almost perfectly perpendicular to the pilot's line of sight. As a result the pilot cannot identify a popped breaker except through noting the "dimensional relationship" among adjacent circuit breaker heads or through lowering his head and examining the contrast between the console area and circuit breaker shafts. This latter identification is hindered because of the lack of contrast between the shaft and surrounding console area.

Frequency channelization of the ARC-34 UHF radio is noted on the radio control panel on the left console. In the past this arrangement has been unsatisfactory as pilots cannot read the frequencies assigned, especially under night lighting conditions. The wording and numbers identifying each channel are generally quite illegible except at close range and under ideal lighting. To correct this, channel frequencies were typed on a card which was placed in the snapup compass correction card holder in the Phase II aircraft. In production models a similar

index card holder should be provided for radio channelization where it will be readily accessible to the pilot for ease of reference.

Switch guards on the Stability Augmentation Control panel were identified with word markings indicating their appropriate function in the Phase II aircraft. It was noted that subsequent models did not have this feature. Although lighting on the panels may provide sufficient data for generalized location of switches during night flights, additional word identification etched into the switch covers provides the pilot with an additional assurance that he is selecting the proper equipment during normal day light operations. The three switch guards on the stability augmentation panel, the generator switch guards, and the fuel tank selective jettison switch are all considered as requiring this type of lettered identification on the switch guard proper.

Examination of the cockpit area indicates that the pilot lacks an appropriate space to store maps or reference data desired during flights. All material will therefore have to be retained on his person. The addition of the pressure suit may preclude attaching these materials to the normal flying suit, therefore, loose material will probably be scattered about the cockpit. A portion of the console area should be modified to allow map and data storage.

The radar control panel was not installed in the Phase II aircraft. Examination of a second test vehicle indicated a need for more appropriate identification of the Master Switch-Scope Intensity and Receiver Frequency Control-Receiver Gain control switches. Each pair of controls are located on a common switch axis with a design which has one function slightly stepped above the other. The switches are shape coded to aid in identification and proper selection of the desired function, however, the lack of contrast between the switches and the radar panel proper makes it very difficult to distinguish the controls. If the important switches are painted light colors they will stand out so that they can be easily located and identified as to distinct function.

- *Visibility:* External visibility would appear to be comparable to or exceed other fighter type aircraft. With minor head movement the pilot can secure 360 degree visibility in the horizontal plane. Overhead visibility is also good with the exception of interference from a canopy structure behind the head. Forward over the nose visibility exceeds the established minimums and is approximately 12 degrees below the horizontal plane. Side visibility approximates 34 degrees over the side from a normal head position and more if the head is moved to either side.

Internal visibility of instrument and control panels is aided by the simplified nature of the cockpit, however, the cockpit has not been extensively evaluated under night flight conditions and with the type of lighting fixtures installed there are occasional dark spots, etc., which should be eliminated.

- *Ventilation, Air Conditioning, and Pressurization:* The introduction of personal equipment to the body of the pilot and the tactical requirement for crash or ejection protection may require that the pilot use the ventilating garment of the MD-1 anti-exposure suit. Use of this gear will require that a source of ventilating air be provided either from the air conditioning system or from a separate air source. Until such time as this requirement does not exist a provision should be made to provide the necessary air supply. In the event it is not considered feasible to equip the aircraft with a permanent source of ventilating air the necessary structure and wiring should be included to allow easy installation of a separate blower.

At the present time ground cooling of the cockpit is accomplished through opening the pilot's Ram Air Scoop. The system operates so that to secure the necessary engine cooling during ground operations it is impossible to use the air conditioning system for cockpit ventilation and cooling. This particular condition should be studied and corrective action taken to allow pilot use of the air conditioning system at all phases of ground or flight operation.

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8725 JOHN J. KINGMAN ROAD, SUITE 0944
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FROM: HQ AFMC/SCDP
4225 Logistics Avenue, Room A112
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SUBJECT: Change in Distribution Statement for AFMC Documents

1. Distribution statements on several documents were officially changed to Distribution Statement A in accordance with AFI 61-204, 27 Jul 94, *Disseminating Scientific and Technical Information*. The documents (excluding those marked out in Atch 3) are owned by AFMC and were reviewed by the HQ AFMC History Office and HQ AFMC Public Affairs Office. The documents cleared for public release are listed on three attachments.

2. Please direct further questions to Ms. Lezora Nobles, AFMC STINFO Assistant, HQ AFMC/SCDP, DSN 787-8583.

Patricia T. McWilliams

PATRICIA T. McWILLIAMS
AFMC STINFO Program Manager
Directorate of Communications and Information

Attachments:
1. AFDTC/PA Memo, 11 Jan 95
2. HQ AFMC/PAX 1st Ind, 4 May 00
3. HQ AFMC/PAX Memo, 5 May 00

2. Attachments a through c are part of an internal AFMC/HO review; attachments d and e are requested by Mr. Morris Betry, a private researcher; attachments f through h are requested by Ms. Pat McWilliams (AFMC/SCDP); and attachment i is requested by Mr. Gregory Hughes (ASC/ENFD).

3. The AFMC/HO point of contact for these reviews is Dr. William Elliott, who may be reached at extension 77476.

JOHN D. WEBER
Command Historian

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~~8~~ Attachments:
a. ~~AFSC No. 150.174~~
b. ~~AFSC No. 400.490~~
c. DTIC No. AD-098 048
d. DTIC No. AD-376 934
e. DTIC No. AD-895 879
f. DTIC No. AD-094 838
g. DTIC No. AD-068 388
h. DTIC No. AD-046 931
~~i. AFLC No. R1-120-2~~

1st Ind, HQ AFMC/PAX 4 May 2000

This material has been reviewed for security and policy IAW AFI 35-101. It is cleared for public release.

JAMES A. MORROW
Security and Policy Review
Office of Public Affairs

atch 2