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NF-104A AEROSPACE TRAINER EVALUATION

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Project Engineer

ROBERT W. SMITH, Major, USAF
Project Pilot

TECHNICAL REPORT No. 65-37

DECEMBER 1965

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Project Engineer

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Major, USAF
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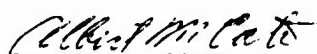
FOREWORD

This report presents the results of the NF-104A evaluation. The tests were conducted from 28 October 1963 to 16 February 1965 at the Air Force Flight Test Center, Edwards Air Force Base, California. Sixty-eight productive flights totaling 42 hours of flying time were required. The tests were conducted under the authority of Headquarters Air Force Systems Command as directed by AFFTC Project Directive 62-39, 3 November 1961. The Project Number was 9060 and the program structure was ESP 922A-62B02.

It is believed that this report contains the first generalized analysis of actual zoom climb performance data. Acknowledgement is given to Mr. Willie L. Allen, Chief of the Fighter Engineering Branch, whose supervision made this analysis possible. The authors also

wish to acknowledge Mr. Robert G. Hoey and Captain Charles D. Zaleski of the Flight Research Division, and 1st Lieutenant Gordon W. Fox and 1st Lieutenant Jack H. Pierson of the Fighter Engineering Branch for their invaluable assistance during the flight test evaluation.

This technical report has been reviewed and is approved.
23 November 1965


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ABSTRACT

The NF-104A Aerospace Trainers were designed to provide students of the Aerospace Research Pilot School with experience in rocket engine operation, reaction control operation, and precise zoom climb profile flying. This report presents the results of the flight test evaluation which was conducted to determine whether the aircraft had acceptable flying qualities for this design mission. It was necessary to incorporate many development items into the aircraft and impose flight restrictions in order to attain a reasonable degree of safety. The aircraft were restricted to a maximum pullup Mach number of 2.15, and a maximum pitch angle of 50 degrees. At the apex of a zoom maneuver the aircraft were restricted to a minimum dynamic pressure of 20 pounds per

square foot and a minimum Mach number of 1.0. The limitations at the apex were placed on the aircraft after it was determined that the J79 engine gyroscopic effects coupled with rapid rate of change of yaw or pitch angles could cause loss of control at very low dynamic pressures. Observing these limitations the aircraft can attain altitudes in excess of 100 000 feet and provide approximately 90 seconds of zero g flight and reaction control experience. The results of the test program showed that successful utilization of these aircraft as Aerospace Trainers requires: (1) strict observance of limitations imposed on the aircraft; (2) intimate familiarization of flight characteristics of the aircraft, and mission profile; and (3) meticulous maintenance of the aircraft and its systems.

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LIST OF ABBREVIATIONS AND SYMBOLS -----

<u>Symbol</u>	<u>Definition</u>	<u>Units</u>
AARS	Attitude and Azimuth Reference System	---
A/B	afterburner	---
A_e	rocket engine nozzle exit area	in. ²
AO	automatic observer	---
APC	automatic pitch control	---
A_T	rocket engine throat area	in. ²
a_{z_w}	air mass acceleration normal to flight path	ft per sec ²
CEC	Consolidated Electrodynamics Corporation	---
C_F	rocket engine thrust coefficient	dimensionless
cg	center of gravity	percent MAC
C_L	lift coefficient	dimensionless

C_{L_α}	rate of change of lift coefficient (C_L) with angle of attack (α)	per degree
C_{l_β}	rate of change of rolling-moment coefficient (C_l) with sideslip angle (β)	per degree
C_{n_β}	rate of change of yawing-moment coefficient (C_n) with sideslip angle (β)	per degree
Dn	down	---
EGT	exhaust gas temperature	degrees C
E/W	specific energy	ft
$(E/W)_R$	specific energy of rocket engine	ft
F_{ex}	excess thrust	lb
F_g	gross thrust	lb
F_n	net thrust	lb
fpm	feet per minute	ft per min
F_R	rocket engine thrust	lb
g	acceleration due to gravity	ft per sec ²
GN_2	gaseous nitrogen	---
H	altitude	ft
H_c	calibrated pressure altitude	ft
H_2O_2	hydrogen peroxide	---
Δh	tapeline altitude gained from pullup to peak	ft
ΔH_{pc}	position correction to be added to pressure altitude	ft
IMN	indicated Mach number	dimensionless
J79-GE-3B	General Electric turbojet engine	---
k	specific kinetic energy at pullup plus the rocket engine specific energy input from pullup to burnout, divided by the total specific energy at the peak	dimensionless
kt	knots	kt
KIAS	knots indicated airspeed	knots
LR121-NA-1	North American Aviation liquid fuel rocket engine	---
M	Mach number	dimensionless
MAC	mean aerodynamic chord	in. or ft
M_{ic}	indicated Mach number corrected for instrument error	dimensionless

ΔM_{pc}	position correction to be added to Mach number	dimensionless
MSL	mean sea level	---
N	engine rotor speed	rpm
NAMT	nautical air miles traveled	nautical miles
n_{fp}	load factor	dimensionless
P_a	ambient pressure	psi or in. Hg
$P_{cd/s}$	rocket engine combustion chamber pressure	psia
pct	percent	pct
P_{t_2}	compressor inlet total pressure	in. Hg
q	dynamic pressure ($q = 1/2 \rho V_t^2$)	psf
RAT	ram air turbine	---
R/C	rate of climb	fps or fpm
RCS	reaction control system	---
rpm	revolutions per minute	rev per min
S	wing area	ft ²
SMAMA	Sacramento Air Materiel Area	---
S/N	serial number	---
t	time	sec
t	temperature	deg C
T	temperature	deg K
t_a	ambient temperature	deg C
T_a	ambient temperature	deg K
TE	trailing edge	---
t_{t_2}	compressor inlet total temperature	deg C
T_{t_2}	compressor inlet total temperature	deg K
V	velocity	fps
V_g	ground speed	fps
V_{ic}	indicated airspeed corrected for instrument error	kt
V_o	initial speed along the flight path referenced to a fixed point on the surface of the earth	fps
V_p	horizontal speed at the peak referenced to a fixed point on the surface of the earth	fps

V_t	true airspeed	kt
ΔV_{pc}	position correction to be added to airspeed (V_{ic})	kt
V_w	wind component along the flight path	fps
W	weight	lb
W_f	JP-4 fuel flow	lb per hr
YAPS (head)	yaw and pitch sensing (head)	---
γ	initial flight path angle from horizontal	deg
γ_{eff}	effective initial flight path angle	deg
δ_a	ambient pressure ratio $P_a/29.92$	dimensionless
δ_{t_2}	inlet pressure ratio, $P_{t_2}/29.92$	dimensionless
∂	partial derivative	dimensionless
θ	pitch angle	deg
θ_a	ambient temperature ratio $T_a/288.16$	dimensionless
λ	thrust inclination angle	radian
ρ	density of the air	lb-sec ² per ft ⁴

Subscripts

s	standardized data
t	test data



**TOP VIEW OF NF-104A SHOWING NOSE BOOM INSTALLATION,
RCS MOTORS, J79 ENGINE INLET CONES, 2 ft WINGTIP
EXTENSIONS AND ROCKET ENGINE INSTALLATION.**

INTRODUCTION -----

A flight test evaluation of the NF-104A Aerospace Trainer was conducted by the Air Force Flight Test Center (AFFTC) to determine if the aircraft had acceptable flying qualities for its design mission. Sixty-eight productive flights, totaling approximately 42 hours of flying time, were made during the period 28 October 1963 through 16 February 1965.

Three F-104A aircraft had been modified into the Aerospace Trainer configuration to allow flight in regimes and under conditions not obtainable with available operational airplanes. The mission of the Aerospace Trainers was to provide students of the Aerospace Research Pilot School with experience in the following:

1. Flying a precise zoom climb profile.
2. Throttleable rocket engine operation.
3. Reaction control system (RCS) operation.
4. Shifting from reaction to aerodynamic controls.
5. Zero normal acceleration (g) conditions.
6. Full pressure suit flying.
7. Reentry effects.
8. Power-off low lift to drag ratio (L/D) landings.

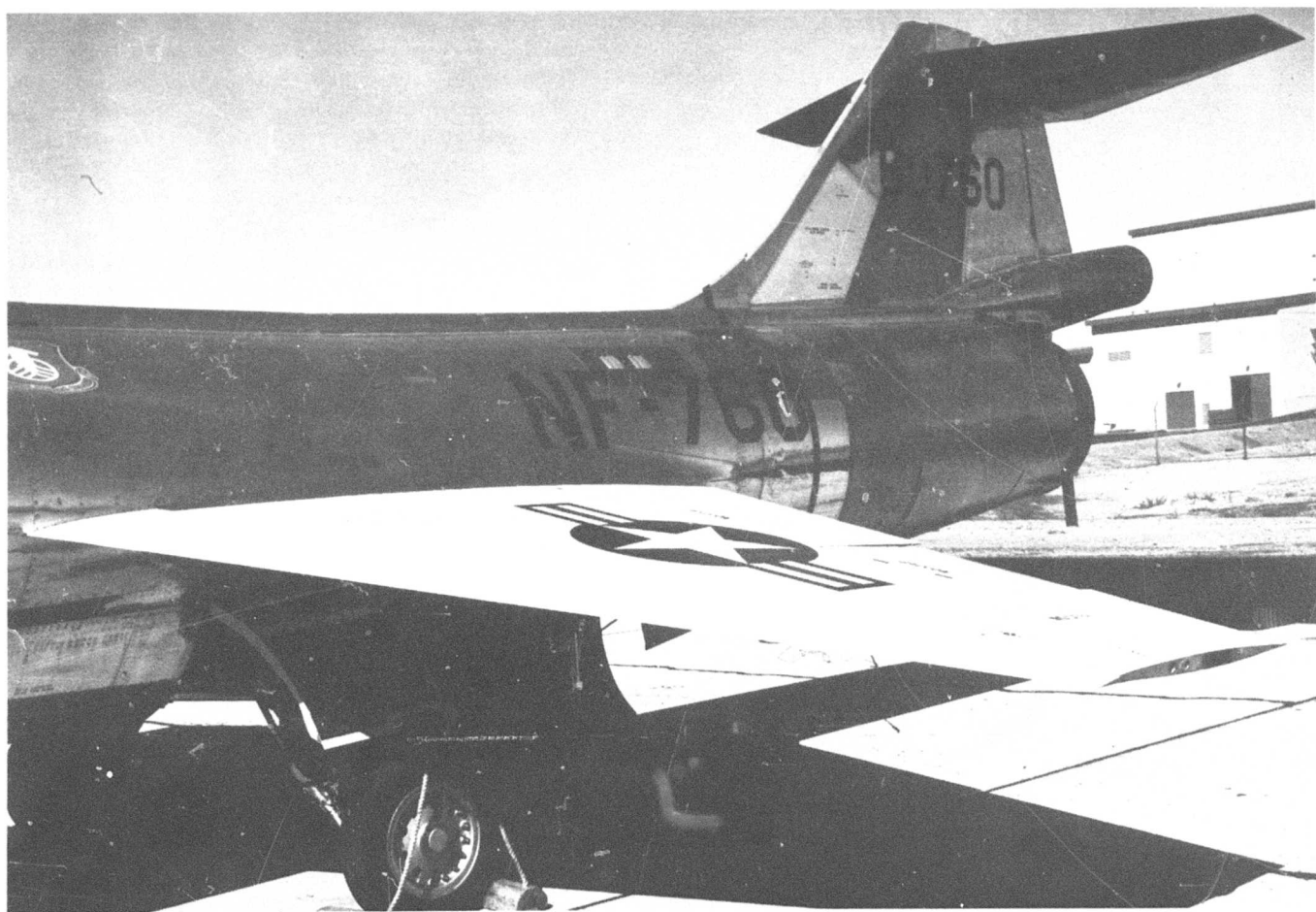


**LR121-NA-1 ROCKET ENGINE INSTALLATION
AND MODIFIED F-104G EMPENNAGE
ASSEMBLY**

Among the modifications that were originally incorporated into the Aerospace Trainers was an LRL21-NA-1 liquid fuel rocket engine in addition to the J79-GE-3B turbojet engine. The propellants for the rocket engine consisted of hydrogen peroxide oxidizer and JP-4 fuel. The rocket engine nacelle was located at the base of the vertical stabilizer. To minimize pitching moments due to rocket engine thrust, the rocket engine mounting was canted so that the line of thrust passed through the aircraft center of gravity (cg) at burnout. An RCS provided attitude stabilization when the dynamic pressure (q) was too low for conventional aerodynamic control effectiveness. Twenty-four inch

wingtip extension panels were provided to accommodate the RCS roll motors. Gaseous nitrogen was used for auxiliary pressurization. The engine air inlet cones were modified for better ram recovery at high Mach numbers, and the electrical load center door incorporated an improved ram air scoop to reduce flow pressures at higher airspeeds. The vertical stabilizer and power rudder consisted of a modified TF-104C unit.

A limited Category I flight test program was conducted by the Lockheed-California Company, prior to the evaluation conducted by the AFMTC. The Category I flight test program consisted of 41 flights in approximately 25 hours of flying



WINGTIP INSTALLATION

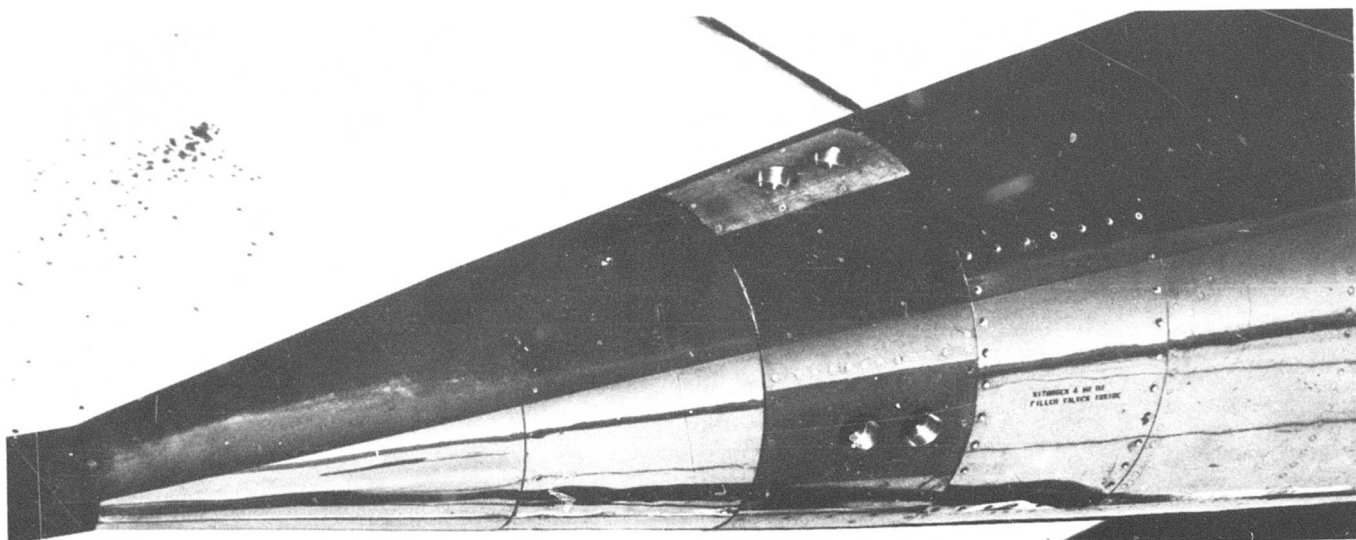
time, during the period of July 1963 through October 1963 (reference 1).

Early in the AFFTC test program one of the aircraft was destroyed after entering a flat spin at the apex of a zoom climb maneuver. As a result of this accident, and the many other problems encountered, numerous modifications as detailed in this report were incorporated into the remaining Aerospace Trainers.

At the completion of the AFFTC test program a Partial Flight Manual (T.O. 1F-104A-1-2), for the

Aerospace Trainer, was generated by the AFFTC and printed at the Sacramento Air Materiel Area (SMAMA) (reference 2).

A complete, up-to-date, instrumentation calibration book for each of the NF-104A aircraft was forwarded to the Aerospace Research Pilot School on 2 June 1965, and the Partial Flight Manual Appendix I, which contained information to allow complete zoom climb mission planning, was transmitted on 15 June 1965. This document was also printed at SMAMA.



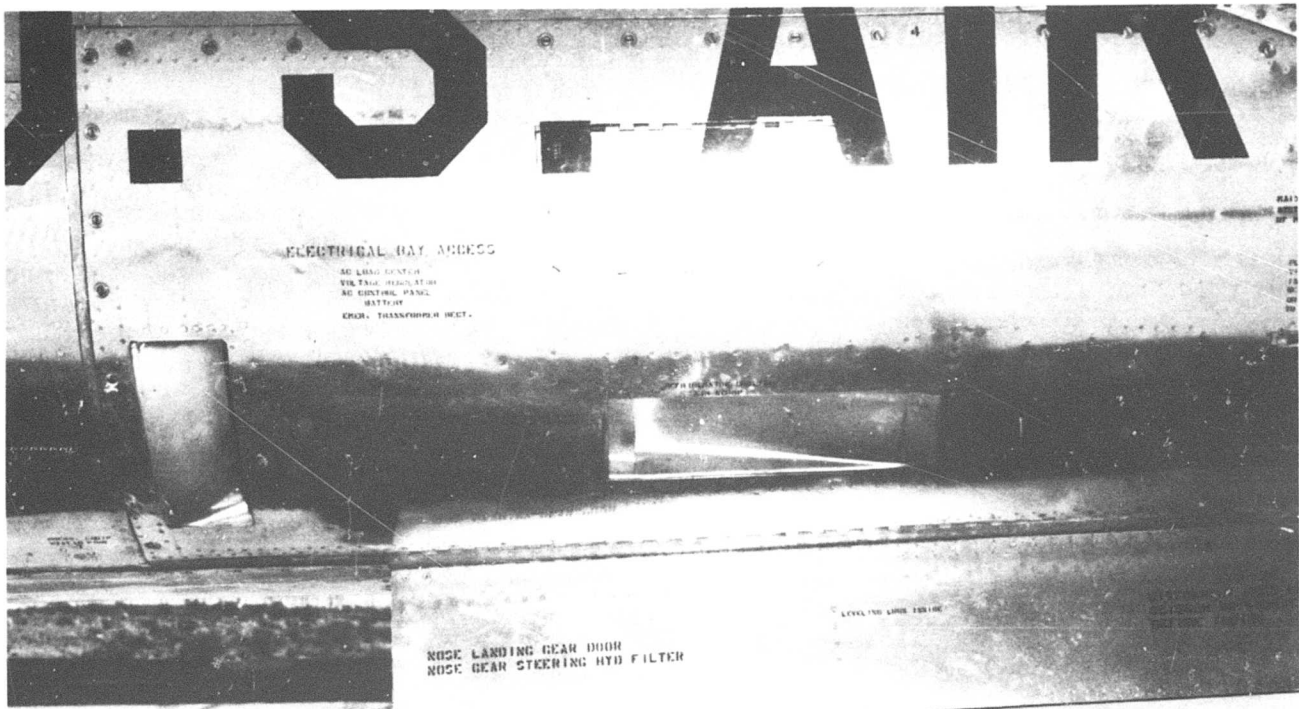
EXHAUST PORTS FOR RCS YAW AND PITCH MOTORS

**REACTION CONTROL
THRUST REACTOR**

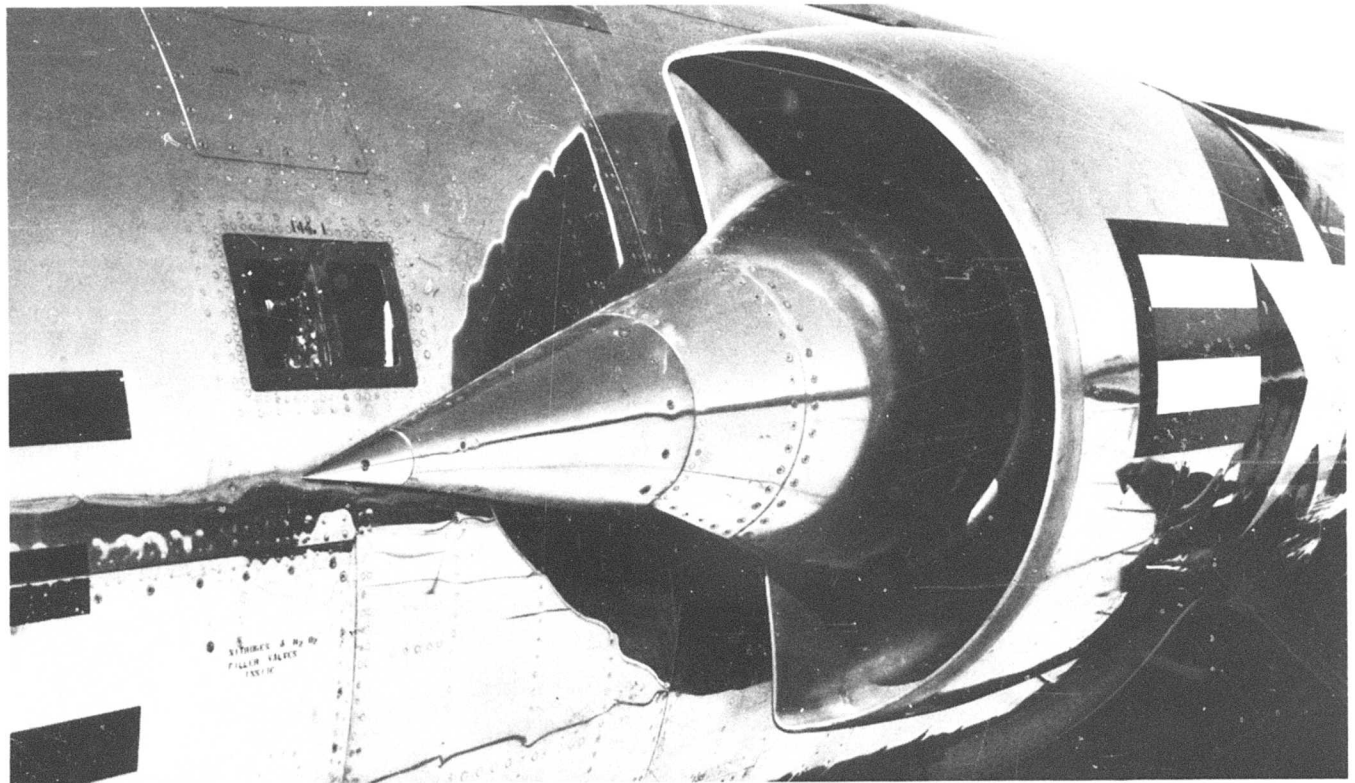
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**ACTION
VERFLOW**

EXHAUST PORT FOR RCS ROLL MOTOR



ELECTRICAL LOAD CENTER RAM AIR SCOOP MODIFICATION



J79 INLET CONE MODIFICATION

TEST AND EVALUATION-----

■ Preflight, Starting and Ground Handling

The NF-104A preflight, starting, and ground handling characteristics were similar to the F-104A except for the systems peculiar to the Aerospace Trainer. A discussion of these items is contained in the Aerospace Trainer Partial Flight Manual.

■ Takeoff Performance

Takeoff tests were conducted with the NF-104A to evaluate the elevator effectiveness because of the increased forward cg condition over that of the F-104A. In a fully serviced configuration the gross weight of the NF-104A was 22 000 pounds. The load on the nose gear was approximately 5000 pounds, and the gear down cg position was at minus 4.1 percent of the mean aerodynamic chord (MAC) (typical F-104A cg at takeoff was plus 2 percent MAC). Because of the heavy gross weight, all takeoffs were made with maximum afterburner power. Military power was set and the brakes were released when the throttle was advanced to the afterburner range. At 90 knots indicated airspeed (KIAS) the control stick was moved aft to decrease nosewheel loads during the high speed portion of the ground roll. Initial aircraft rotation was slow because of the forward cg. When aircraft rotation began, back pressure on the stick was partially released and the aircraft flown off the ground. The wing flaps were raised at 260 KIAS or above. The acceleration with maximum thrust was continued to 400 KIAS, then the throttle was slowly retarded to military power.

The nosewheel lift-off speed was 185 KIAS and the main gear lift-off speed was about 210 KIAS.

The tires were limited to a ground speed of 239 knots, therefore the aircraft was restricted to upwind takeoffs. Main gear lift-off speed was 12 knots higher and the ground roll distance was 1000 feet longer for the NF-104A than the F-104A Flight Manual values at the same weight. This decreased performance was caused by the more forward cg position for the NF-104A. The results of these tests are presented in figure 1, appendix I. (R 1)¹

■ Subsonic Climb Performance

Data were obtained during the subsonic climb portion of the zoom climb mission to determine the time, distance, and fuel required to climb to the acceleration altitude. The climbs were made in a clean configuration with military power. The climb schedule used was 450 KIAS to 0.86 indicated Mach number (IMN) and then 0.86 IMN was held until the desired altitude was reached. The 450 KIAS portion of the climb schedule was less than the 0.86 IMN recommended in the F-104A Flight Manual because at the recommended speeds airframe buffet was encountered at the lower altitudes. The aircraft traveled 89 nautical air miles and required approximately 10 minutes and 970 pounds of fuel to climb from sea level to a pressure altitude of 35 000 feet on a standard day. After reaching acceleration altitude it was necessary to cruise approximately 10 nautical miles further to reach the appropriate turn around point.

¹Numbers indicated as (R 1), etc., represent the corresponding recommendation number as tabulated in the Conclusions and Recommendations section of this report.

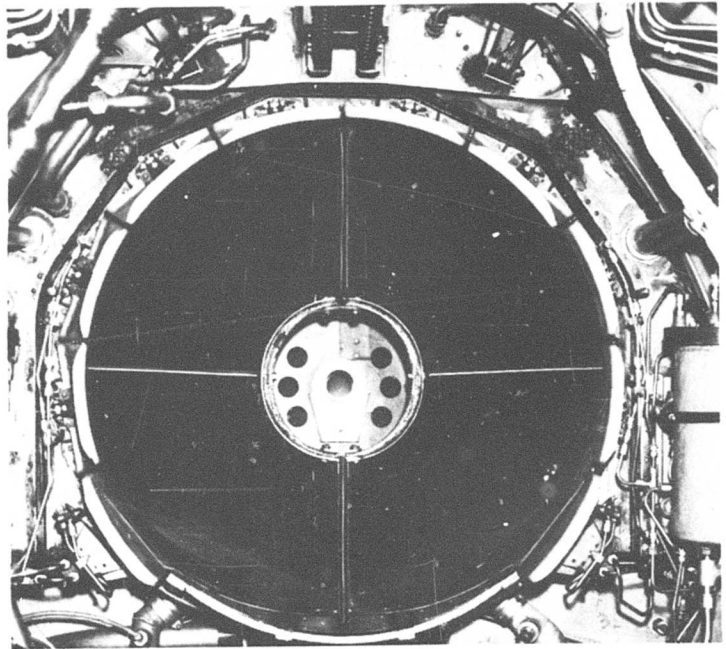
The methods employed in reducing the test day climb data to standard conditions are discussed in appendix I and the data are presented in figure 2.

■ Level Acceleration Performance

Level acceleration tests were conducted to determine the time, distance, and fuel required to accelerate from 0.9 Mach number to the speed at which the zoom maneuver was initiated. Accelerations were made with maximum afterburner thrust and also with maximum afterburner plus the rocket engine thrust.

The acceleration portion of the zoom climb mission required large quantities of fuel because of the low excess thrust from about 1.0 to 1.5 Mach number. It was determined from the analog simulation that for the most favorable fuel utilization, the rocket engine should be operated from 1.1 to 1.4 Mach number and ignited again at a Mach number one-tenth less than pullup Mach number. The fuel saved by using this technique gave approximately a 2 percent MAC more forward cg which improved the stability at peak altitude and increased the fuel remaining for landing. Using the rocket engine in this manner did not significantly impair the zoom climb performance of the aircraft.

At 1.6 Mach number the J79 engine "B" bypass flaps were opened, which increased the bypass airflow around the engine. This action resulted in more stable operation of the engine air inlet system, which alleviated compressor stalls and duct spillage, and permitted speeds in excess of that possible with the normal bypass flap configuration. Exhaust gas temperature (EGT) was monitored at 615 degrees C after T_2 reset occurred at a compressor inlet temperature (CIT) of 79 degrees C.



J79 ENGINE BYPASS FLAP INSTALLATION

The addition of the rocket engine thrust increased the potential maximum standard day speed of the aircraft from 2.0 to 2.2 Mach number at a pressure altitude of 35 000 feet. The maximum allowable speed, however, was controlled by a J79 engine CIT limit of 155 degrees C which occurred at 2.18 Mach number on a standard day at a pressure altitude of 35 000 feet. On test days with higher than standard temperatures it was possible to reach the CIT limit before the desired pullup Mach number was reached. Prior to reaching this limit, the rocket engine was ignited and pullup was initiated.

There were no graduations on the CIT gage above 150 degrees C. To more accurately determine maximum allowable CIT, additional graduations are required to include 160 degrees C. (R 8)

Plots of excess thrust and JP-4 fuel flow vs Mach number for 10 degrees C hot, 10 degrees C cold, and standard day conditions

can be seen in figures 3 through 8, appendix I. These data are shown for the J79 engine only and for the J79 plus rocket engine. The data show that the rocket engine thrust was approximately 5700 pounds at a pressure altitude of 35 000 feet.

The acceleration performance of the aircraft, with the rocket engine operating from 1.1 to 1.4 Mach number, is summarized in the following table:

ACCELERATION PERFORMANCE					
NF-104A Aerospace Trainers					
USAF S/N 56-0756 and 56-0760					
Weight = 20 350 lb at 0.9 Mach number					
Pressure Altitude 35 000 ft					
Rocket Relight Mach Number	Pullup Mach Number	Time to Accelerate (sec)			
		-40 °C	-50 °C	-60 °C	-70 °C
1.8	1.9	269	224	184	145
1.9	2.0	314	250	202	160
2.0	2.05	----	277	222	175
2.0	2.10	----	281	226	180
2.0	2.15	----	287	231	183
Rocket Relight Mach Number	Pullup Mach Number	Distance Traveled (NM)			
		-40 °C	-50 °C	-60 °C	-70 °C
1.8	1.9	66	48	38	32
1.9	2.0	75	55	44	35
2.0	2.05	----	65	50	41
2.0	2.10	----	67	52	42
2.0	2.15	----	69	53	43
Rocket Relight Mach Number	Pullup Mach Number	Fuel Used to Accelerate (lb)			
		-40 °C	-50 °C	-60 °C	-70 °C
1.8	1.9	1560	1310	1140	1000
1.9	2.0	1820	1520	1300	1120
2.0	2.05	----	1740	1450	1250
2.0	2.10	----	1820	1510	1300
2.0	2.15	----	1870	1550	1340

Notes:

1. Numbers indicate time, distance, and total JP-4 fuel required to accelerate from 0.9 Mach number to pullup Mach number at the ambient temperatures shown.
2. Rocket engine operated from 1.1 to 1.4 Mach number.

■ Zoom Climb Performance

Zoom climb maneuvers were conducted to determine a reasonably safe flight profile which would meet the training requirements of the Aerospace Research Pilot School and to obtain sufficient Flight Manual information to permit complete preplanning of a zoom maneuver.

All zooms were initiated by increasing load factor at the rate of 1 g per second until 3.5 g was reached. This normal acceleration was recommended by the contractor as being optimum for a 65-degree pitch attitude. Testing to determine the optimum normal acceleration to transition from level flight to a given pitch angle was not accomplished during this evaluation. The 3.5-g pullup was held until the desired pitch angle was indicated on the Attitude and Azimuth Reference System (AARS). This pitch angle was held until the angle of attack planned for the particular mission was reached. This angle of attack was then flown until the peak altitude was attained, and then 8 degrees angle of attack was held during reentry. In this report the word reentry was used to describe that portion of the zoom maneuver from the apex through dive recovery. Although speed brakes were not required their use decreased both the altitude lost and the maximum Mach number attained during reentry. The speed brakes were extended as reentry was started. In order to increase the maximum allowable Mach number from 1.6 to 2.0 with the speed brakes deflected, blocks were installed which reduced the maximum deflection of the speed brakes from 60 degrees to 52 degrees. The speed brakes were retracted during the turn toward the landing area to increase the gliding distance.

The J79 engine was allowed to run as long as possible during the zoom climb without exceeding the

EGT limitations. It was determined that the throttle should be retarded to military power at 63 000 feet tapeline altitude or an EGT of 625 degrees C, whichever occurred first, and that the J79 engine be shutdown at 80 000 feet tapeline altitude or an EGT of 610 degrees C. These tapeline altitudes were called out to the pilot by the ground radar monitor.

A wide range of performance data were obtained by varying the pullup Mach number from 1.92 to 2.18 and by varying the maximum pitch angle from 27 to 61 degrees. These data were used to construct plots of aircraft pitch angle, "k" factor, aircraft ground speed at the peak, and increase in tapeline altitude from pullup to peak all versus initial effective flight path angle. These plots, shown in figures 9 through 12, appendix I, were then used to construct the Zoom Climb Performance chart contained in the Partial Flight Manual and figure 13, appendix I of this document. By using this chart it can be determined that on a no-wind standard day a zoom climb maneuver conducted at the maximum allowable pitch angle of 50 degrees would have to be restricted to a maximum pullup Mach number of 2.1. These initial conditions would result in the minimum allowable q of 20 psf, a minimum Mach number of 1.22 at a maximum tapeline altitude of 105 500 feet.

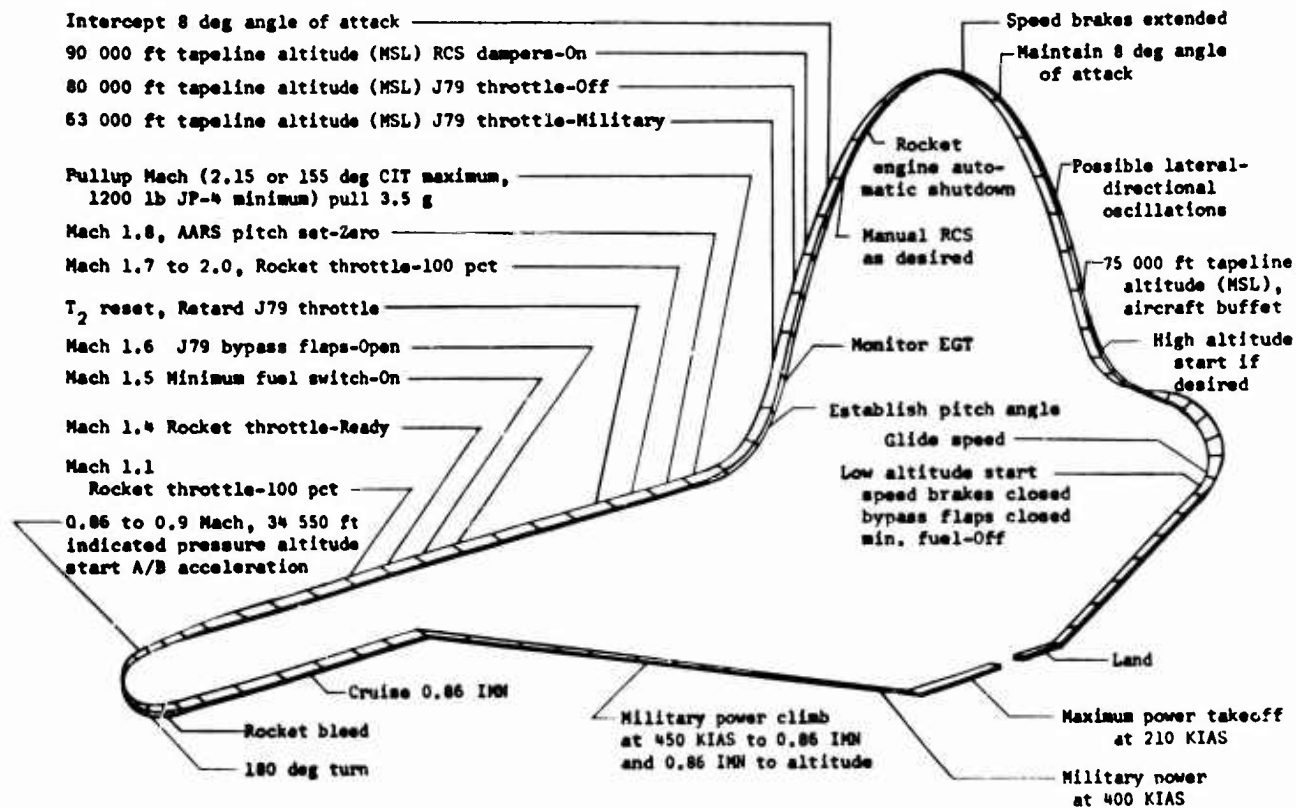
As a result of these flight tests it was found that, in general, Mach number had a tendency to increase after pullup so a maximum standard day pullup Mach number limit of 2.15 was arbitrarily established to prevent the J79 en-

gine from exceeding the CIT limit of 155 degrees C. On a day colder than standard it was possible to exceed the 2.15 Mach number limit before reaching the 155 degrees C CIT limit. On a day warmer than standard it was possible to reach the CIT limit before reaching the 2.15 Mach number limit. A maximum allowable pitch angle of 50 degrees was established to conform with the maximum allowable pullup Mach number of 2.15, and the minimum allowable conditions of 1.0 Mach number and a q of 20 psf at the peak of the zoom. The criteria for establishing the minimum Mach number and q are discussed in the stability and control section of this report. It should be noted that atmospheric conditions may necessitate lowering either the maximum allowable pullup Mach number or pitch angle to remain within the established limits at the apex. (R 2, R 7)

Two other limitations were placed on the aircraft at pullup. A low fuel condition in the aircraft during the zoom climb could cause sporadic fuel flow to the rocket engine and subsequent burning of the fuel injector nozzle; therefore, a minimum fuel load of 1200 pounds at pullup was established. Pullup was planned to be initiated at approximately 40 nautical miles from the AFFTC Radar Guidance and Control Center to insure radar coverage throughout the zoom climb maneuver. It should be noted that the aircraft covered about 50 nautical miles ground distance during all zoom maneuvers except those conducted for stability data.

A typical zoom climb profile is shown in the following figure.

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760



Typical Zoom Climb Mission

■ Cruise Performance

Level flight cruise data for the NF-104A were obtained at a pressure altitude of 35 000 feet at a light gross weight to simulate the post zoom climb condition. These data were obtained to define optimum cruise conditions for use when returning to the base in the event of an "overshoot" in a zoom climb maneuver, and to establish a comparison with F-104A Flight Manual values.

Five percent conservatism was removed from the F-104A Flight Manual range factor values and a comparison was made with the test data. The limited results obtained from the test program showed that the maximum range factor (nautical

air miles per pound x weight) was 3530 and the optimum cruise Mach number was 0.87 at the optimum weight-pressure ratio of 71 000 pounds. This range factor was approximately 2.5 percent less than the F-104A Flight Manual value with the 5 percent conservatism removed. Cruise data are presented in figures 14 and 15, appendix I.

■ Pitot-Static System Calibration

The NF-104A all metal nose cone incorporated a test boom and yaw and pitch sensing (YAPS) head as a permanent installation. A sketch of the nose boom and YAPS head installation is shown in figure 16, appendix I. Airspeed system calibrations were obtained

both in and out of ground effect (figures 17 and 18, appendix I). A new nose boom was locally manufactured shortly after the evaluation began and a separate calibration was required. An RCS hydrogen peroxide leak damaged the original pitot-static lines, and they were replaced with stainless steel mesh covered teflon lines which were compatible with hydrogen peroxide. A lag calibration of the static system was obtained after the new lines were installed and is shown in figure 19, appendix I.

■ Static Thrust Calibration

An installed static thrust calibration was accomplished on J79 engine S/N 147-387 at the beginning of the test program. The results of this calibration are shown in figures 20 and 21, appendix I.

An installed static thrust calibration was accomplished on LR121-NA-1 rocket engines S/N 404 and 405. The average, sea level, static thrust of these two engines was approximately 5250 pounds. The results of this calibration can be seen in figure 22, appendix I.

■ Stability and Control

Emphasis was placed on determination of the NF-104A stability and control characteristics in three critical areas pertinent to the airplane's design mission. The first was to define the lateral-directional stability characteristics of the aircraft at high Mach numbers and high angles of attack with the speed brakes extended. The second was to investigate the two flat spin accidents, involving an NF-104A and an F-104A, which occurred at the AFFTC. The third was to determine the minimum q at which the aircraft could be controlled aerodynamically at the apex of a zoom maneuver.

● lateral-directional stability

At the beginning of the flight test program, the data available for the speed brakes retracted configuration showed that the level of directional stability of the aircraft decreased with increasing Mach number. It was also indicated by the limited amount of data available that the level of directional stability decreased when the speed brakes were extended. On several flights, lateral-directional oscillations were encountered during reentry following a zoom climb maneuver. In each case these oscillations were encountered under the following conditions:

1. High Mach number.
2. Angle of attack greater than 5 degrees.
3. Jet engine windmilling.
4. Rocket engine off.
5. Aerodynamic stability augmentation on.
6. Speed brakes extended.
7. RCS dampers off.

The pilot was able to damp these oscillations by decreasing the aircraft angle of attack. On two flights the oscillations built up almost immediately after the RCS dampers were switched off. On one flight the oscillations occurred during a period of no RCS inputs, and were damped out when the RCS dampers came on.

Since aileron and rudder damper inputs were not desirable during the lateral-directional stability study, yaw and roll gyro cutout switches were installed in both aircraft. Activating these switches prevented damper response to aircraft oscillations.

A 3-degree of freedom analog simulation was used to develop a mission profile that could be flown to obtain lateral-directional stability data. Flight test data below 2.0 Mach number was obtained by flying this profile. The lateral-directional stability parameters extracted from these data were then extrapolated to higher Mach numbers and programmed into a 5-degree of freedom analog simulation. This simulation was then employed to predict handling qualities at Mach numbers of 2.0 and above.

On one stability flight the jet engine was allowed to run during a rudder pulse at 2.12 Mach number. After about two cycles of a damped oscillation the jet engine was shut down. Immediately the oscillations diverged to approximately ± 5 degrees of sideslip and then damped out gradually as angle of attack and Mach number were decreased.

It was concluded that the lateral-directional oscillations which were encountered during reentry were caused by an oscillatory air duct spillage condition that occurred at high angles of attack and Mach numbers when the J79 engine was windmilling. An approximate boundary for these oscillations in terms of angle of attack and Mach number is shown in figure 23, appendix I.

Flight test results were used to determine values for the yawing moment due to sideslip ($C_{n\beta}$), and the rolling moment due to sideslip ($C_{l\beta}$), between 1.3 and 2.1 Mach number with the speed brakes retracted and extended. The effect of having the speed brakes extended was to decrease the values of $C_{n\beta}$ and $C_{l\beta}$ at all Mach numbers as

shown in figure 24, appendix I. Mach number had only a slight effect on $C_{l\beta}$ but $C_{n\beta}$ decreased with increasing Mach number with speed brakes retracted and extended. $C_{n\beta}$ decreased as Mach number increased above Mach 2.0 although not as rapidly as at the lower Mach numbers. Insufficient data were obtained and scatter was too great to quantitatively define the effect of angle of attack on $C_{n\beta}$ and $C_{l\beta}$.

The static lateral-directional stability analysis showed that the levels of $C_{n\beta}$ and $C_{l\beta}$ were sufficient to provide satisfactory handling qualities up to at least Mach 2.12 under the conditions which existed during reentry after a zoom maneuver. Total static lateral-directional stability appeared to be positive to at least Mach 2.3. The dynamic instability, apparently caused by duct spillage at high Mach numbers and high angles of attack, showed that use of the aircraft should be restricted to Mach 2.0 with the jet engine windmilling (reference 3).

● investigation of flat spin accidents

During the early stages of the test program the Aerospace Research Pilot School was given permission and assumed responsibility for a world altitude record attempt. On 10 December 1963, prior to the establishment of an official altitude record, NF-104A S/N 56-0762 was destroyed following a zoom climb to a tapeline altitude of 101 600 feet. After reaching a low dynamic pressure (q) of 10 psf, at the apex of a zoom maneuver the pitchup angle of attack was exceeded and the aircraft entered a fully developed flat spin. The F-104A Flight

Manual spin recovery procedures were followed but failed to recover the aircraft. The drag chute was deployed at a tapeline altitude of about 17 000 feet, momentarily recovering the aircraft from the spin. The aircraft accelerated almost straight down to about 180 KIAS with the chute fully deployed. When the chute was released the resulting noseup trim change caused the aircraft to pitch up immediately and enter another flat spin. The pilot ejected successfully at a tapeline altitude of about 7000 feet above the terrain. The accident investigation board attributed the primary cause of the accident to the fact that the aircraft could not be recovered from a spin which resulted because the angle of attack exceeded the pitchup limit. A contributing factor was that full control response was not available because the aileron and rudder travel limiters were not deactivated (reference 4).

Investigation revealed that the AARS angle of attack indication was nonlinear near the pitchup angle of attack and the cockpit presentation gave an inadequate indication to the pilot.

This accident and a subsequent F-104A accident revealed that both of these aircraft had a flat spin mode from which a recovery could not be effected without a spin recovery device.

The NF-104A had electrical solenoid operated aileron and rudder limiters that were automatically activated when the main gear doors were up and locked. The rudder was limited to one-third of its unlimited travel which was approximately 20 degrees in each direction. The ailerons were limited to two-thirds of their unlimited travel which was 15 degrees in each direction at zero trim. It was possible to deactivate both

limiters by actuating a switch in the cockpit. It was determined that deactivation of the aileron and rudder limiters would have provided full control response and a better opportunity of spin recovery prior to the fully developed flat spin.

This accident pointed out the necessity of spin avoidance, and the need for development of a spin recovery device and improvement in the spin recovery techniques. To avoid a spin it was necessary to maintain positive angle of attack control especially at the apex of a zoom maneuver where q was low. To assist in spin avoidance a separate angle of attack indicator was installed which indicated from +40 to -10 degrees. The AARS angle of attack command bar was mechanized to provide variable null settings so that preset angles of attack could be presented to the pilot. The purpose of this modification was to facilitate the performance of zoom climb maneuvers at angles of attack other than the original 11-degree null setting. It was subsequently determined that 8 degrees angle of attack was the most favorable and the null switch was fixed at this setting. As an additional precaution, to avoid an inadvertent pitchup, the RCS was modified to provide an automatic pitch control "kicker" mechanism. In the event the angle of attack reached 15 degrees, this mechanism actuated both nose down pitch motors. The motors continued to fire until the angle of attack decreased to 13 degrees.

A multi-purpose switch was installed to facilitate preparation for spin recovery in the event that a spin could not be avoided. Activating this switch retracted the speed brakes and deactivated the RCS dampers and the aileron and rudder limiters. The speed brakes were retracted because the lateral-directional stability of

the aircraft was more favorable in that configuration. The RCS dampers were deactivated because they were designed to maintain a constant attitude which would be detrimental to spin recovery. The aileron and rudder limiters were deactivated so that full control authority would be available.

It was determined that the standard F-104A drag chute could create the nosedown pitching moment necessary to recover the aircraft and that there was no upper altitude limit for chute deployment when the aircraft was in a flat spin. A procedure for using the drag chute to recover from a flat spin was developed with the 3-degree of freedom simulation (reference 5). The results of the simulation indicated that the chute should be deployed above 25 000 feet to allow sufficient altitude for recovery. Following chute deployment full nosedown trim should be applied and sufficient back stick should be used to fly straight down until flying speed is attained. Just prior to chute release, the stick should be neutralized. This procedure would counteract the trim change due to drag chute jettison and the aircraft could be recovered by a normal pullout. It should be noted that this procedure was not verified by flight test. (R 3)

On 15 June 1964, a student of the Aerospace Research Pilot School zoomed an F-104A aircraft to a tapeline altitude of approximately 83 000 feet and a q of 8 psf. Near the apex of the zoom the pilot felt the automatic pitch control (APC) actuate. He immediately applied full forward stick, generating a large nosedown pitch rate to avoid pitchup. The nose of the aircraft pitched downward and yawed to the left approximately 135 degrees. The aircraft subsequently entered a fully developed flat spin. The F-104A Flight Manual spin recovery procedures were followed and

although full control authority of the rudder was available the aircraft failed to recover. The drag chute was ejected but failed to deploy and the pilot ejected successfully at a tapeline altitude of about 4000 feet above the terrain. The primary cause of the accident was that the aircraft entered a flat spin from which it could not be recovered. A contributing factor was that the aircraft was being flown in a hazardous regime of low q , approached its aerodynamic limits and the inertial forces beyond the control of the pilot caused the pitchup angle of attack to be exceeded (reference 6).

The 5-degree of freedom analog simulation was used in support of the accident investigation to define:

1. The gyroscopic effects of the J79 engine.
2. The low q handling qualities of the aircraft.
3. A reasonably safe zoom profile.

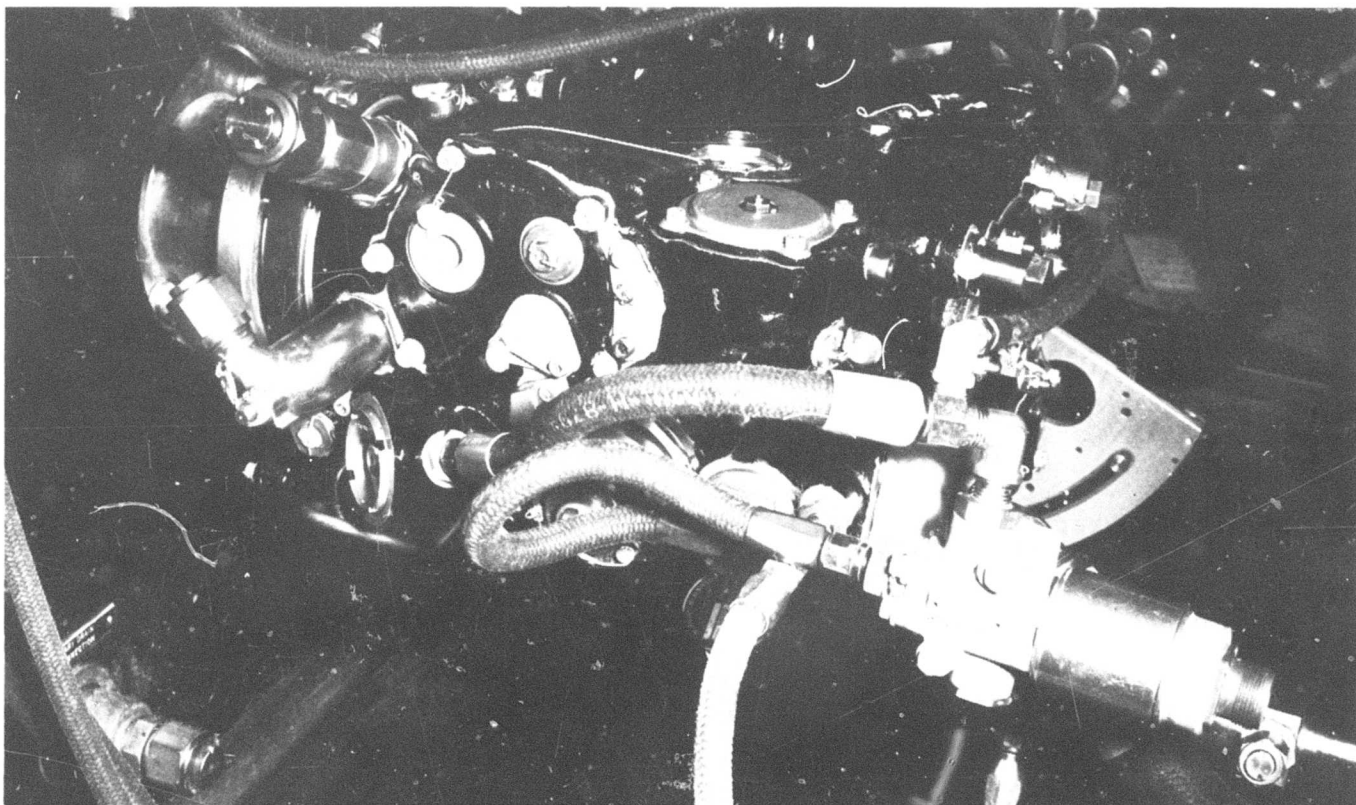
Two significant results were obtained from the simulation. First, it was determined that the lateral-directional damping became negative as q became very low and dynamic lateral-directional instability existed. Secondly, and more important, it was found that the gyroscopic effects of the J79 engine, which couple pitch rate into yawing moment and yaw rate into pitching moment, become significant at low q where aerodynamic restoring forces were low.

As is well known, any movement of a gyro causes precession 90 degrees to the direction of motion. In this case, a negative pitch rate induced by a rapid forward stick action caused the aircraft to yaw left. The yaw in turn

created a left roll caused by dihedral effects. The inertial forces and the aerodynamic forces coupled in the form of a yaw-roll maneuver to place the aircraft at an excessively high angle of attack. The gyroscopic effects of the windmilling J79 engine led to the initial oscillatory spin, but as the engine rpm decayed to a low value this spin evolved into a stabilized pitched-up condition and a flat spin mode.

The normal F-104 oscillatory spin depended on high J79 engine rpm and the consequent gyroscopic

effects for its oscillatory motion. This motion of the nose oscillating above and below the horizon created the favorable environment necessary for recovery using the techniques prescribed in the F-104A Flight Manual. The inertial forces that caused the entry into the spin would have aided in recovery if these forces could have been preserved by maintaining high engine rpm (reference 7). This information led to a modification which decreased the minimum fuel flow and allowed the J79 engine to run to a higher altitude thus producing a higher engine rpm at the apex of a zoom maneuver.



J79 ENGINE MINIMUM FUEL FLOW MODIFICATION

It became apparent from the F-104A accident that the gyroscopic effects of the J79 engine were a major contributing factor in the NF-104A accident, which occurred under similar conditions.

- determination of the minimum allowable dynamic pressure

Investigation of the two flat spin accidents revealed the necessity to determine the minimum q that could be tolerated under the most adverse conditions at the apex of a zoom climb maneuver. The criteria established was that the aircraft should be aerodynamically controllable after sustaining APC and RCS "kicker" actuation.

A 5-degree of freedom analog simulation study was conducted to evaluate the handling qualities at low q and to develop a method of flight testing. The handling qualities were determined by increasing the angle of attack, at various pitch rates, from stable conditions at 11 degrees until the APC actuated at approximately 14 degrees and the RCS "kicker" fired at 15 degrees, then an attempt was made to control the aircraft aerodynamically. The simulator tests were conducted at a q of 10, 20, and 40 psf. The results of the simulation showed that the aircraft became uncontrollable at a q of 10 psf at an angle of attack of 19.5 degrees with a relatively low nose-up pitch rate of 4 degrees per second. At a q of 40 psf with the RCS "kicker" activated a pitch rate could not be generated which rendered the aircraft uncontrollable. The RCS "kicker" provided a pitch-up margin of approximately 3 degrees angle of attack and 4 degrees per second pitch rate at all dynamic pressures. It was determined that a minimum q of 20 psf would be a realistic limit. The results of this study are shown in figure 25, appendix I. The points where pitchup and un-

controllable flight occurred are shown by a series of vertical crosshatched lines.

A sequence of flight tests were then conducted to verify the minimum q limit established by the simulation. These flights started at a minimum q of 40 psf and decreased in 5 psf increments on subsequent flights. The test method was the same as the simulation study except that the pitch rates never exceeded approximately 2 degrees per second.

The moments created by the RCS "kicker" and by the stabilizer when the APC actuated caused a rapid decrease in angle of attack and pitch angle. The negative pitch rate caused the aircraft to yaw left, induced by the gyroscopic action of the windmilling J79 engine. As in the spin accidents the inertial and aerodynamic forces coupled in the form of a yaw-roll maneuver which tended to place the aircraft above the pitchup angle of attack. The insidious nature of this maneuver required prior knowledge of the corrective action to prevent the aircraft from becoming uncontrollable. The corrective action was to push forward on the stick as the angle of attack decreased through zero to decrease the magnitude of the impending noseup pitch rate and angle of attack. The time history of one flight, figure 26, appendix I, shows the characteristics of the oscillations encountered during this maneuver.

During one of the test flights the aircraft was flown to a minimum q of 13.5 psf. Since at this q the entry noseup pitch rates were very low and the aircraft was approaching the transonic region where the aerodynamic restoring moments were less favorable, it was verified that the minimum q of 20 psf as determined by the simu-

lation was a realistic value, and the aircraft were limited accordingly. The RCS dampers were not activated on these flights and it was determined that flight within the limited profile established in the Zoom Climb Performance section of this report was possible without RCS dampers but it was more difficult to control the attitude of the aircraft. After the minimum allowable q at the peak was established, several zoom maneuvers were made to obtain data for Flight Manual information. The first few flights were conducted at an aim angle of attack of 11 degrees at the peak, while the pilot maintained aircraft attitude with the RCS. When the RCS controls were manipulated in the vicinity of the peak of the zoom, the APC actuated on almost all occasions, because the APC setting of 14 degrees was close to the aim angle of attack. To alleviate this problem and allow RCS manipulation at the peak, without APC actuation, the aim angle of attack at the peak was limited to a maximum of 8 degrees. This angle of attack was also determined to be most favorable for reentry. Higher angles of attack decreased the margin to the pitchup angle of attack and lower angles allowed speed buildup on reentry.

■ Analog Simulation

An analog simulation was developed to study performance characteristics and handling qualities of the NF-104A as a supplement to the flight test program. Simulation was developed with 3, 5, and 6 degrees of freedom to study particular aspects of the Aerospace Trainer flight characteristics. Appropriate parameters were computed and displayed to the pilot in the stationary cockpit so that he could experience the same instrument readings as he would during an actual flight. In addition, altitude, Mach number,

angle of attack, normal acceleration, dynamic pressure, flight path angle, pitch attitude, pitch rate, and horizontal stabilizer position were recorded on strip recorders for engineering evaluation. A detailed description of this simulation and the studies conducted is contained in reference 5.

The 3-degree of freedom simulation was used to study aircraft performance and to develop a safe flight profile to study the lateral-directional stability characteristics of the aircraft. Zoom climb performance was also determined in terms of altitude, Mach number, and dynamic pressure at the apex for various conditions of pullup Mach number, pitch angle, Mach number for rocket engine light, initial weight, pullup g , and profile angle of attack. Plots were then obtained of altitude, Mach number, and dynamic pressure at the apex for various pitch angles to show overall performance capability of the aircraft. These performance plots were used for mission planning.

The primary use for the 5-degree of freedom simulation was in conjunction with the NF-104A lateral-directional stability study conducted during the test program. It was programmed with pitch, yaw, and roll dynamics as well as lift and side-force characteristics. For this simulation, control surface positions, body attitudes, body rates, angle of attack, sideslip angle, and normal acceleration were recorded. The primary lateral-directional stability parameters in the simulation were updated with flight test results to insure that the simulator response closely matched the aircraft characteristics. Then, the simulator was used to evaluate handling qualities at various flight conditions of constant Mach number and altitude. This simulation was also used to determine the parameters affecting pitchup

and controllability at low q and to determine a q which would be safe for normal operation. Maximum pitch rate versus maximum angle of attack were plotted in figure 25, appendix I, to show the effect of the APC and RCS "kicker" under low q conditions.

Before the flying phase of the test program was completed, a 6-degree of freedom simulation was programed for the Aerospace Research Pilot School for use as a procedural trainer. The availability of the 6-degree of freedom procedural simulation, which had been updated with flight test data, permitted checkout of the Aerospace Research Pilot School instructor pilots after the test program was completed. The pilots were able to practice typical zoom climb maneuvers, gain experience with the RCS and become familiar with aircraft handling characteristics at low dynamic pressures before they actually flew the airplane.

■ Time Histories of Typical Zoom Climb Maneuvers

Normal procedures and flight characteristics of a typical zoom climb maneuver are given in Section II and IV of the Aerospace Trainer Partial Flight Manual. Figure 26 is a time history of a mission flown to obtain lateral-directional stability data. Figures 27 through 30 are time histories of zoom maneuvers flown at pitch angles of 30, 40, and 50 degrees. All altitudes discussed in this section are pressure altitudes.

● lateral-directional stability flight

Flight No. 24 (figure 27, appendix I) is representative of the flights used to study lateral-directional stability characteristics of the aircraft at 80 000 feet and Mach 1.8 to 2.1.

A 3 g pullup was started at just over Mach 2.0. During pullup the Mach number reached 2.1 and remained there for 15 seconds. The pitch angle reached 25 degrees very rapidly after pullup and was maintained at approximately 25 degrees throughout the climb. Pushover was started at approximately 70 000 feet and 81 000 feet was attained before the aircraft pitch angle decreased to level flight attitude. The rocket engine burned out at approximately 70 000 feet or about the time of pushover. The EGT inadvertently rose to 665 degrees C, which is above the J79 engine limits, before the afterburner was shutdown at 70 000 feet. The J79 engine was shutdown at 79 000 feet.

When level flight attitude was attained, the angle of attack was increased to approximately 11 degrees in preparation for the rudder pulse. After the rudder pulse the angle of attack varied from the planned 11 degrees due to lateral-directional oscillations. Note that altitude and Mach number decreased throughout the time that 11 degrees angle of attack was maintained but dynamic pressure during that time was nearly constant at 120 psf. After the stability test was completed, normal reentry was accomplished. The distance traveled from pullup to reentry on a typical mission of this type was approximately 60 nautical miles.

● zoom climb at a pitch angle of 30 degrees

Flight No. 47 (figure 28, appendix I) was a Mach 1.9 pullup, 30-degree pitch angle mission. The pullup Mach number was 1.85 and increased to just over 1.9 during pullup. It took approximately 14 seconds after pullup to reach the planned 30-degree pitch angle. The pitch angle was then

maintained at 30 degrees until the angle of attack reached 8 degrees. At that point the pilot switched from pitch angle to an angle of attack reference allowing pitch angle to slowly decrease. The afterburner was shutdown at 70 000 feet and the main engine at 82 000 feet, keeping the EGT within operating limitations. The J79 engine windmilled down to 80 percent rpm at peak altitude and to a minimum of 70 percent rpm during reentry. The rocket engine burned out at 80 000 feet. The maximum altitude attained was 88 500 feet.

On this test flight the pilot intentionally increased angle of attack until the APC actuated and the RCS "kicker" fired, shortly after peak altitude, while at a dynamic pressure of 60 psf. This was accomplished to determine the minimum dynamic pressure at which aerodynamic control of the aircraft could be maintained after actuation of both kickers. Upon kicker actuation, the angle of attack decreased from 12 degrees to 2 degrees in about 2 seconds with resultant sudden changes in pitch angle and normal acceleration. The resultant nosedown pitch rates also produced yaw and roll oscillations caused by J79 engine gyroscopic effects. The pilot recovered using only aerodynamic controls.

Reentry was at 6 to 8 degrees angle of attack with Mach number reaching a maximum of 1.6 at 65 000 feet. J79 engine restart was accomplished at 47 000 feet after pullout. The distance traveled from pullup to reentry on a typical mission of this type was approximately 50 nautical miles.

- zoom climb at a pitch angle of 40 degrees

Flight No. 61 (figure 29, appendix I) was a 40-degree pitch angle flight with a 2.15 Mach number

pullup. Mach number decreased more rapidly than for Flight No. 47 due to the higher pitch angle attained on this flight. The 38-degree sustained pitch angle was slightly lower than the planned 40 degrees. As the angle of attack reached 8 degrees, pitch angle began to decrease. On this flight the afterburner was shutdown at 64 000 feet and the J79 engine was shutdown at 80 000 feet, keeping the EGT within limits, but the rocket engine continued burning until 90 000 feet. The maximum altitude attained was approximately 100 000 feet. Just after peak altitude the dynamic pressure decreased to 30 psf and Mach number decreased to 1.35. The J79 engine rpm decreased to a minimum of 68 percent near the apex on this flight as compared to 74 percent for Flight No. 47.

Maximum reentry Mach number reached 1.83 for this flight which shows the general trend toward higher reentry Mach numbers as pullup Mach number, pitch angle and resultant peak altitude were increased. The J79 engine was restarted at approximately 43 000 feet. The distance traveled from pullup to reentry on a typical mission of this type was approximately 49 nautical miles.

- zoom climb at a pitch angle of 50 degrees

Flight No. 62 (figure 30, appendix I) was planned to result in the maximum altitudes obtainable, while remaining within the minimum allowable conditions of 20 psf dynamic pressure and 1.0 Mach number at the apex.

Pullup actually occurred at 2.18 Mach number. The pitch angle increased rapidly to 51 degrees and remained there until the angle of attack increased to 8 degrees. Mach number began to decrease

shortly after pullup and reached a minimum of 1.2 approximately 10 seconds after peak altitude was reached. The afterburner was shutdown at 65 000 feet, and the J79 engine was shutdown at 78 000 feet, again keeping the EGT within limits. The rocket engine burned out at 93 000 feet. The maximum altitude attained was approximately 105 500 feet.

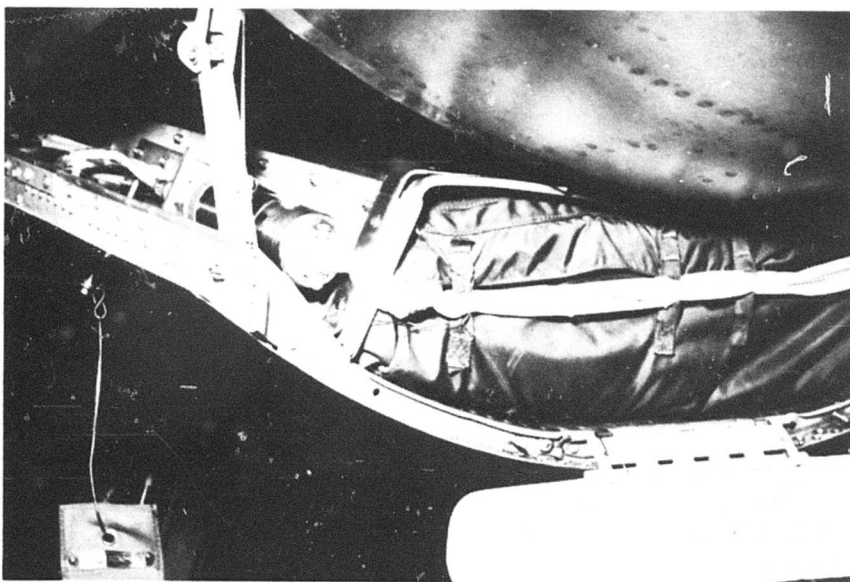
Aerodynamic controllability of the aircraft was evaluated on this flight at a dynamic pressure of 20 psf. APC actuation to decrease angle of attack caused a rapid rate of change of pitch angle inducing the usual lateral-directional oscillations. The aircraft was recovered aerodynamically but control was more difficult at 20 psf than for higher dynamic pressures. J79 engine rpm decreased to a minimum of 63 percent which showed the trend of the engine to windmill to a lower rpm as altitude approached the maximum and q approached the minimum. Maximum Mach number reached on reentry was 1.9 for this flight. The J79 engine was restarted at 42 000 feet after the reentry dive recovery was accomplished.

■ Aircraft Systems

Many systems problems were encountered during the evaluation, and it was necessary to incorporate numerous development items into the aircraft. Some of these developments were made to correct design deficiencies and others were made to improve flying qualities and to increase safety of flight. The more significant problem areas and developments are discussed below.

● drag chute

Investigation of the NF-104A flat-spin accident of 10 December 1963 revealed that at low engine rpm the F-104 aircraft had a flat spin mode from which recovery could not be made without the use of a spin recovery device. Calculations revealed that the standard F-104A drag chute could create the nose-down pitching moment necessary to recover the aircraft from a flat spin. In order to use the drag chute as a spin recovery device several modifications were required to insure positive deployment under the low airspeed, high angle of attack condition encoun-



DROGUE GUN INSTALLATION

tered during a flat-spin. The drag chutes on the test aircraft were modified as follows:

1. The standard 16 foot drag chute was packed in a modified, F-104G chute bag so that the canopy could be deployed first.
2. The pilot chute was removed and a drogue gun was installed for positive deployment.
3. The spring tension was increased in the "D" ring lock of the attachment mechanism to prevent the chute from detaching from the aircraft on deployment.
4. A nichrome burn wire was installed on the parachute riser as a jettison redundancy.
5. The chute riser was extended from 22 feet to 35 feet to provide adequate clearance from the horizontal stabilizer when the chute was deployed.

● J79-GE-3B turbojet engine

Three J79-GE-3B engines were used in aircraft S/N 56-0760 during the evaluation. The first one was removed due to excessive surging during an acceleration and the second was removed because the EGT limits were inadvertently exceeded during a flight investigating lateral-directional stability.

During the contractor's flight test program and at the beginning of the evaluation at the AFFTC, J79 engine compressor stalls were encountered at and above 2.0 Mach number. The design of the F-104 was such that, as speed approached 2.0 Mach number, the engine airflow requirements and normal bypass of

air were approximately equal to the air provided by the inlet ducts. If more air were supplied than could be used by the engine and bypass system, a mismatch existed which could result in duct spillage and compressor stall. Secondary airflow around the engine was supplied through bypass flaps located at the mating point of the engine intake and the inlet duct. The bypass flap assembly contained 10 movable flaps located around the perimeter of the engine inlet. Originally, the only bypass flaps that were movable were the two at the bottom of the inlet, one on each side of the centerline. These two flaps opened when the landing gear was retracted to provide 44 square inches of bypass area and were termed the "A" group flaps. Four additional flaps, two on each side, were activated on the Aerospace Trainers and made operable from the cockpit. These flaps, termed the "B" group flaps, provided an additional 38 square inches of bypass area. By increasing the bypass area, airflow capability was increased. This modification alleviated compressor stalls and duct spillage at high Mach numbers.

After the F-104A spin accident on 15 June 1964, it was determined that high J79 engine rpm was desirable at the apex of a zoom maneuver. A modification to decrease the minimum fuel flow to the J79 engine from 550 pounds per hour to 325 pounds per hour was installed on the fuel control of both aircraft. This modification allowed the engine to run at a higher altitude without exceeding the EGT limit in order to increase the rpm at the apex, and consequently the high altitude restart capability was also improved.

Airstarts with the J79 engine were obtained both at high altitude during reentry and at low altitude after pullout. The high altitude

starts were accomplished below a tapeline altitude of 75 000 feet during reentry whenever sufficient attention could be given to the engine instruments. Both start switches were actuated and the throttle placed at idle. After pullout was completed the throttle was advanced momentarily to determine if an airstart had been obtained. After engine relight had been obtained, the minimum fuel flow switch was turned off and the bypass flaps were closed. The low altitude restart was accomplished before the engine decreased below 65 percent rpm, if possible, so that the number 1 and 3 fuel boost pumps would still be operating. Prior to engine start, the speed brakes and "B" bypass flaps were closed and the minimum fuel flow switch was turned off.

On several occasions the pilot was unable to obtain a relight until very low altitudes were reached, and in one instance a "flamed-out" landing was made. Investigation revealed that because of the low fuel quantity condition the two aft fuel boost pump inlets were uncovered; and that, when the J79 engine rpm was below 65 percent, the two forward fuel boost pumps were inoperative. Under these conditions a relight could be obtained in only two ways: (1) by leveling the aircraft which would cover the inlets of the 3 electrically driven boost pumps and allow the J79 engine driven boost pump to provide fuel flow, or (2) deploy the Ram Air Turbine (RAT) to activate one of the forward boost pumps which had an inlet immersed in fuel. Examination of the fuel and electrical system revealed that a boost pump was not connected to the RAT for emergency operation; and also that, because of faulty preflight procedures, the number 2 boost pump had been inoperative for an indefinite period.

The preflight procedures for checking for boost pump and generator failures were changed, and the No. 2 JP-4 fuel boost pump underfrequency relay was changed to the F-104D configuration to insure boost pump operation down to at least 40 percent engine rpm. The No. 1 JP-4 fuel boost pump was wired to operate from the RAT to allow J79 engine restarts in a gliding attitude, with low fuel conditions to at least 40 percent engine rpm. It was necessary to remove some less essential equipment from the RAT circuit so that the No. 1 fuel boost pump could be accommodated. A No. 1 JP-4 fuel boost pump relay was also installed to eliminate boost pump operation during transient flap operation when on RAT electrical power.

● LR121-NA-1 rocket engine and hydrogen peroxide system

During the test program several problems were encountered with the rocket engine and associated systems, which delayed the programs considerably. Twelve rocket engines were removed from the test aircraft for repair because of major component malfunction. Seven of these removals were because of turbopump JP-4 seal leakage in excess of the limit of 100 cc per minute. The rocket engine service life was increased substantially during the program by increasing the spring tension behind the JP-4 seal and by improving the seal polishing techniques.

During the contractor's flight test program, trapped gases in the rocket engine hydrogen peroxide line had prevented continuous hydrogen peroxide flow causing delayed rocket engine starts and malfunction shutdowns; consequently, a solenoid valve, controllable from the cockpit, was installed at the high point in the line to purge the system of gases before rocket en-

gine light-off. During the test program conducted at the AFFTC several rocket engine malfunctions occurred even though the recommended 1-minute bleed had been accomplished. It was determined that on a hot day, when takeoff was delayed, large amounts of gases were generated in the main hydrogen peroxide feed line, and under these conditions it was necessary to bleed the hydrogen peroxide system for 2 one-minute periods.

During a zoom climb maneuver the rocket engine usually shutdown prior to reaching the peak. This shutdown was indicated by 2 or 3 thrust surges as the mouth of the hydrogen peroxide feed line was uncovered. Because of the design of the rocket engine hydrogen peroxide system, approximately 50 gallons of residual hydrogen peroxide remained after rocket engine shutdown. This residual hydrogen peroxide was a hazard to the aircraft, and the system should be redesigned to permit hydrogen peroxide depletion or jettison and purging with gaseous nitrogen after rocket engine shutdown. (R 6)

On 5 July 1964 the teflon bellows between the rocket engine hydrogen peroxide tanks on aircraft S/N 56-0760 ruptured because of high temperature and pressure created by contamination in the tanks. This incident happened over a weekend when the hydrogen peroxide temperature and pressure indications were not being monitored. It was determined that the contamination was caused by iron oxide (rust) which had accumulated in the tanks over a long period of time. The rust probably came from the 303 stainless steel sumps, which had a high iron content and from an iron plug in the hydrogen peroxide filler line filter. It was necessary to heat treat the forward tank to make it passive

with hydrogen peroxide. Two new sumps were locally manufactured out of 347 stainless steel, which was more compatible with hydrogen peroxide. It was determined, that after dumping, the hydrogen peroxide tanks should be vented into de-ionized water at all times when the aircraft was not flying.

● RCS and associated systems

An X-15 type RCS was incorporated into the Aerospace Trainers to provide attitude stabilization when q was too low for conventional aerodynamic control effectiveness. Early in the test program a hydrogen peroxide leak in the nose of the aircraft destroyed a section of the teflon tape covered pitot-static lines. These lines were replaced with flexible stainless steel covered teflon lines which were compatible with hydrogen peroxide.

On several occasions the O-rings failed in the RCS roll motor hydrogen peroxide line slip-joints, which allowed hydrogen peroxide to leak into the wing. It was found that the overlap in the slip-joint was not adequate and the O-rings were exposed during pressure application. As a result of these leaks a bracket was installed at the wing roots to prevent movement in the lines during pressure surges. The slip-joint overlap was increased to prevent exposure of the O-rings, and wing fillet inspection panels were installed to allow quick access to the wing root areas. The location of the slip-joints made maintenance and inspection difficult, and it was impossible to determine if the O-rings were damaged during installation. Although slip-joints were used in other systems in the aircraft a more positive connection is required in this application because a hydrogen peroxide leak while airborne could result in loss of an aircraft. (R 4)

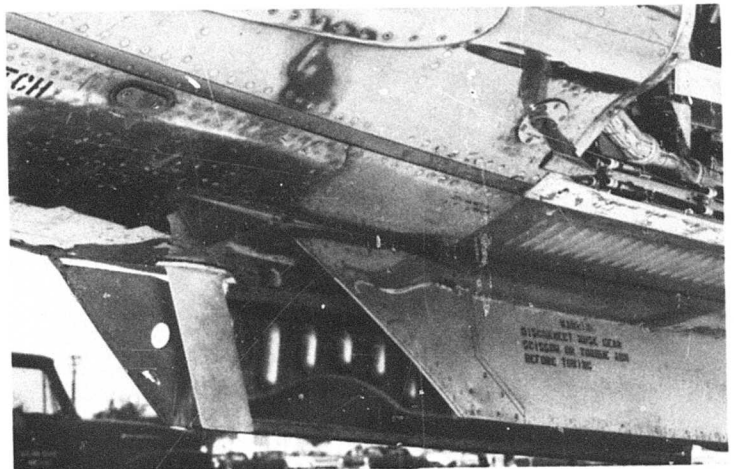
At the completion of a zoom maneuver several gallons of hydrogen peroxide remained in the RCS system. This residual hydrogen peroxide was considered a hazard. In the event of a leak this residual hydrogen peroxide could start a fire or decompose and create an explosive atmosphere. Inadvertent firing of the RCS motors on the ground could also cause serious injury to ground personnel. A redesign of the RCS should be accomplished so that the hydrogen peroxide tank could be depleted and purged with gaseous nitrogen prior to landing. (R 5)

When the aircraft were received from the Lockheed-California Company it was necessary to have the J79 engine running to functionally check the RCS dampers and to operate the instrumentation. An external power relay was installed so that these functions could be performed using external power. Two RCS damper computers were damaged during functional checks of the system. One incident occurred while maintenance was being performed on the RCS hydrogen peroxide system of aircraft S/N 56-0762, and the other incident occurred during RCS instrumentation calibration on aircraft S/N 56-0760. It was subsequently determined that, when the RCS motor heater circuit breakers were pulled and the RCS control handle was manipulated, electrical feedback from the motor heaters damaged the damper computer. Diodes were installed in the circuits to prevent electrical feedback and recurrence of these incidents.

● UHF radio transceivers

Because of weight and space considerations the NF-104A design incorporated the small Collins Radio of Canada 718B-8 transceiver. Two transceivers were installed in each aircraft because the 718B-8 transceiver had a bandwidth of only

two megacycles (mc), and operation was required on a test frequency of 319.6 mc and also on the guard frequency of 243.0 mc. The usable range for satisfactory communication during the test program was about 80 nautical miles. Originally, the only antenna was flush mounted on the lower forward access hatch. This antenna, coupled with the low transmitter power output, caused the aircraft transmissions to be highly directional. Transmissions made to the ground station while the aircraft was turning or banking such that the antenna was pointed away from the station were often garbled and always weak. A "knife-blade" antenna was installed externally under the cockpit of one aircraft in an attempt to alleviate the transmission problem. The results were minimum satisfactory and this modification was not included on the other aircraft. The aircraft radio transmitter power output was increased to approximately 2-watts which improved the quality of transmission from the aircraft somewhat, but the radios remained unsatisfactory to the end of the test program. (R 9)



KNIFE BLADE ANTENNA INSTALLATION

● MA-5 battery

A 34-amp-hr MA-5 Sonotone or Gulton nickle-cadmium battery was

incorporated into each aircraft during the original design to assume the electrical load during a zoom maneuver after the J79 engine had windmilled below 65 percent rpm. Early in the test program when the battery assumed the full electrical load its output would rapidly drop below 22 volts which was too low for effective radio operation. This low voltage also degraded the quality of the oscillograph and photopanel data. To alleviate the electrical system voltage drop three additional cells were added, which permitted the batteries to maintain 24 to 25 volts for at least 5 minutes.

Another problem was encountered when the alkaline battery solution overflowed during zoom maneuvers and damaged the aircraft skin. It was found that the battery vented overboard when the aircraft was climbing and into a sump when descending. This condition was alleviated by altering the vent configuration so that both the forward and aft vents drained into the sump.

● pressure suit

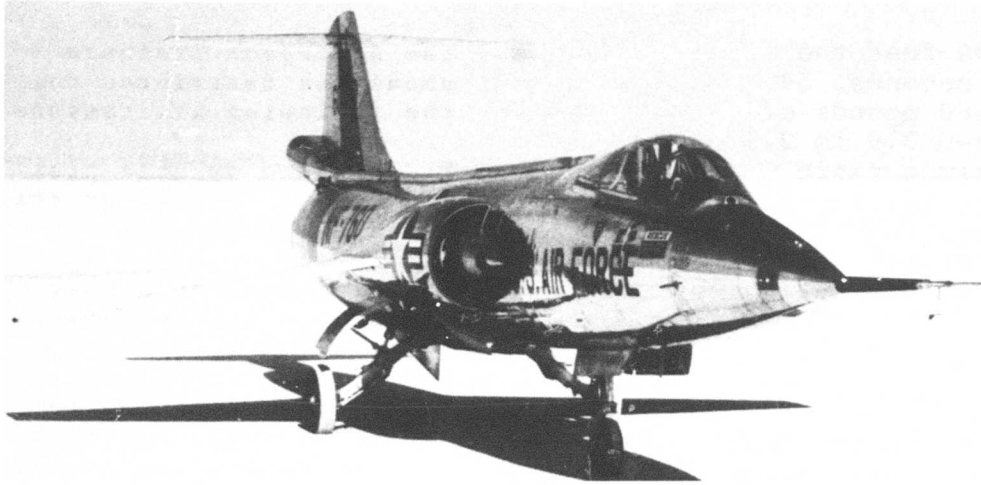
The AP22S-2 full pressure suit was worn on all zoom climb maneuvers. During the subsonic climb portion of one flight, a seal in the helmet oxygen regulator failed and blocked the flow of oxygen to the pilot. This incident would have been disastrous if it had occurred during the zoom maneuver since gaseous nitrogen was

used in the auxiliary cockpit pressurization system. In the gaseous nitrogen atmosphere the face plate could not be opened to obtain oxygen. After this incident all AFPTC AP22S-2 full pressure suit helmets were modified to eliminate this seal in the oxygen regulator.

Helmet visor fogging could be anticipated if cockpit pressurization were lost during a zoom climb maneuver. This would be an intolerable situation since a safe reentry could not be effected to an altitude where the cockpit could be purged of gaseous nitrogen and the visor opened. Visor fogging could be avoided by incorporating a visor heater into the helmet. (R 10)

● oscillograph

The permanent instrumentation system provided in the aircraft included a 50 channel Consolidated Electrodynamics Corporation (CEC) oscillograph mounted in the nose. Early in the test program the electrical components in the oscillograph were arcing when the aircraft was above a pressure altitude of approximately 80 000 feet during the zoom climb maneuver. This arcing caused the oscillograph to malfunction and created a hazard to the aircraft. It was determined that the arcing was caused by the low density atmosphere in the unpressurized nose cone at high altitudes. These arcing components were insulated to alleviate this problem.



CONCLUSIONS AND RECOMMENDATIONS -----

The NF-104A represented a significant advance over the F-104A in the ability to attain higher energy levels and higher altitudes during a zoom maneuver. However, the aircraft had several stability and control characteristics that prevented utilization of its maximum energy capabilities. It was thus necessary to compromise the performance envelope and impose flight restrictions on the aircraft in order to attain a reasonable degree of safety. The results of the test program showed that successful utilization of these aircraft as Aerospace Trainers requires; (1) strict observance of limitations imposed on the aircraft, (2) complete familiarization with the flight characteristics of the aircraft and the mission profile, and (3) meticulous care in maintenance of the aircraft and its systems.

The performance of the NF-104A with the rocket engine inoperative was slightly decreased from that of the F-104A because of increased gross weight. Due to a more forward cg position on the NF-104A takeoff speed was 12 knots higher and ground roll distance was 1000

feet longer than for the F-104A at the same gross weight. Downwind takeoffs made on a hot day at the Air Force Flight Test Center could result in speeds approaching the tire limit of 239 knots (ground speed).

1. The aircraft should be restricted to upwind takeoffs (page 5).

The aircraft traveled 89 nautical air miles and required approximately 10 minutes and 970 pounds of fuel to climb from sea level to 35 000 feet on a standard day using military power. The acceleration portion of the zoom climb mission required large amounts of fuel, especially through the low excess thrust region of 1.0 to 1.5 Mach number. The acceleration technique employed during the test program used maximum afterburner power throughout with the rocket engine operating from 1.1 to 1.4 Mach number and re-lighting at a Mach number one-tenth less than pullup Mach number. This technique gave more fuel for landing without significantly impairing the zoom climb performance of the aircraft. Using this technique on a

standard day at 35 000 feet the NF-104A required 255 seconds, 59 nautical miles and 1660 pounds of JP-4 to accelerate from 0.9 to 2.1 Mach number. The maximum range factor for the NF-104A was approximately 2.5 percent less than the F-104A Flight Manual values for the optimum weight-pressure ratio of 71 000 pounds.

The static lateral-directional stability was sufficient to provide satisfactory handling qualities up to at least 2.12 Mach number under conditions that existed during re-entry after a zoom climb maneuver. Dynamic stability analyses showed that practical use of the aircraft should be restricted to 2.0 Mach number with the jet engine windmilling. It was suspected that a dynamic instability was caused by an oscillatory air duct spillage condition at high Mach numbers and high angles of attack. At very low q , near the apex of a zoom climb maneuver, dynamic lateral-directional instability existed and J79 engine gyroscopic effects could cause loss of control. Investigation of an F-104A spin accident revealed that high pitch rates caused by rapid stick inputs under low q conditions caused the inertial and aerodynamic forces to couple in the form of a yaw-roll maneuver. These gyroscopic effects of the windmilling J79 engine led to the initial oscillatory spin, and as the engine rpm decreased to a low value the spin evolved into a stabilized high angle of attack condition in a flat spin mode. Simulation studies and subsequent flight test showed that to allow aerodynamic controllability near the peak of a zoom maneuver the aircraft should be restricted to a minimum dynamic pressure of 20 psf, and a minimum Mach number of 1.0. The aircraft should be restricted to a maximum pitch angle of 50 degrees during the climb.

2. The Aerospace Trainers should be restricted to the following limitations:

- a.** Minimum dynamic pressure of 20 psf at the apex (page 8).
- b.** Minimum Mach number of 1.0 at the apex (page 8).
- c.** Maximum pitch angle of 50 degrees during the climbs (page 8).

The standard F-104A drag chute could provide the nosedown pitching moment necessary to recover the NF-104A from a flat spin. A noseup trim change caused by chute release, however, could cause secondary pitchup.

- 3.** When the drag chute is used as a spin recovery device the following technique should be employed; apply full nose-down trim and use sufficient back stick to fly straight down to accelerate the aircraft to flying speed. The stick should be neutralized prior to releasing the chute. This technique has been incorporated into the F-104A Flight Manual (page 13).

Many materials in the near vicinity of the hydrogen peroxide system are catalytic in nature and a hydrogen peroxide leak could result in the loss of an aircraft. During the test program several leaks developed in the reaction control system slip-joint connections. After reaching the peak of a zoom maneuver approximately 50 gallons of hydrogen peroxide remained in the rocket engine hydrogen peroxide system and about 5 gallons

of hydrogen peroxide remained in the reaction control system (RCS). This residual hydrogen peroxide presents a hazard in the event a leak should develop in either system.

- *4. All slip-joints in the RCS hydrogen peroxide system should be replaced with a more positive connection to preclude any possibility of a hydrogen peroxide leak (page 22).²
- *5. The RCS should be modified so that the hydrogen peroxide tank can be depleted and purged with gaseous nitrogen before landing (page 23).
- *6. The rocket engine hydrogen peroxide system should be redesigned to permit hydrogen peroxide depletion or jettison and purging with gaseous nitrogen after rocket engine shutdown (page 22).

The maximum allowable speed of the J79 engine was limited by a compressor inlet temperature of 155 degrees C. This temperature occurred at 2.18 Mach number on a standard day at 35 000 feet. Since Mach number had a tendency to increase after pullup, the aircraft was limited to a maximum Mach number of 2.15 at pullup.

² Corrective action has been taken on the recommendations marked with an asterisk.

The scale of the cockpit compressor inlet temperature instrument was limited to 150 degrees C.

- 7. The Aerospace Trainers should be restricted to a maximum pullup Mach number of 2.15 (page 8).
- 8. To determine maximum allowable compressor inlet temperature, additional scale graduations are required to include 160 degrees C (page 6).

On many flights radio contact with the aircraft was lost during cruise away from the base, at the apex of a zoom, or on reentry.

- 9. Adequate radios should be installed to permit radio contact throughout the zoom mission (page 23).

Helmet visor fogging could be anticipated if cockpit pressurization were lost during a zoom climb maneuver. The pilot would not be able to open his face plate to effect reentry because of the gaseous nitrogen atmosphere in the cockpit.

- 10. The AP22S-2 full pressure suit helmet should be modified to incorporate a visor heater (page 24).

APPENDIX I -----

TEST TECHNIQUES AND DATA ANALYSIS METHODS

■ Standard Atmosphere

The data in this report have been corrected from test values to the 1959 ARDC Model Atmosphere defined in reference 8.

■ Takeoff Performance

Takeoff data were recorded by Askania phototheodolite cameras from brake release through a height of at least 50 feet. Ambient conditions of wind, temperature, and pressure were simultaneously noted. The data were corrected to sea level standard day conditions by the methods outlined in reference 9. The mean effective net thrust was calculated using ambient air temperature and pressure and curves from the J79 Performance Manual (reference 10). The standardized takeoff performance is shown in figure 1.

■ Subsonic Climb Performance

Continuous climbs at military power were accomplished from approximately 5000 feet to 35 000 feet. The climb schedule used was 450 KIAS to 0.86 IMN and then, 0.86 IMN to altitude. Test rate of climb was determined by taking slopes of the E/W versus time curve at each test point.

$$(R/C)_t = \frac{d(E/W)_t}{dt} = \frac{F_{ex_t} V_t}{W_t}$$

where

$(R/C)_t$ is the test rate of climb (fpm)

$\frac{d(E/W)_t}{dt}$ is the slope of the specific energy versus time curve at a test point (fpm)

The test rate of climb was corrected for temperature, weight, flight path angle, rate of change of flight path angle, and the component of air mass acceleration normal to the flight path using the standardization techniques outlined in equation (89), reference 11. The last two terms of equation (89) were replaced by one term which included a correction for rate of change of flight path angle as well as flight path angle and the component of air mass acceleration normal to the flight path. The additional term follows:

$$F_{ex_s} = F_{ex_t} + \dots + \frac{\partial F_{ex}}{\partial n_{fp}} (n_{fp_s} - n_{fp_t})$$

where

$$n_{fp_s} = \cos \gamma_s + \frac{V_{t_s}}{g} \frac{d\gamma_s}{dt}$$

$$n_{fp_t} = \cos \gamma_t + \frac{V_{t_t}}{g} \frac{d\gamma_t}{dt} - \frac{a_{z_w}}{g}$$

$$\frac{\partial F_{ex_t}}{\partial n_{fp}} = - \frac{\frac{F}{C_{L_\alpha}} \sin(\alpha + \lambda) + q S \frac{\partial C_D}{\partial C_L}}{\frac{F}{C_{L_\alpha}} \cos(\alpha + \lambda) + q S} W_t$$

It should be noted that the climb data shown in figure 2 were not corrected for power changes or Mach number being off schedule.

■ Level Acceleration Performance

Supersonic acceleration data were obtained at 35 000 feet pressure altitude. Accelerations were performed using maximum rocket engine thrust and/or maximum J79 engine thrust.

Net excess thrust was determined from the time rate of change of specific energy at the test condition expressed as rate of climb in the following equation:

$$F_{ex_t} = \frac{(R/C)_t W_t}{101.333(V_{t_t})} = \frac{d(E/W)_t W_t}{101.333(V_{t_t})dt}$$

where

F_{ex_t} is the test excess thrust (lb)

W_t is the test gross weight (lb)

V_{t_t} is the test true airspeed (kt)

The test excess thrust was corrected for ambient temperature, ambient pressure, aircraft gross weight, flight path angle, and rate of change of flight path angle, using the standardization techniques as outlined in equation (89) of reference 11. The last two terms of equation (89) were changed as shown in the climb section of this appendix. A correction for the component of air mass acceleration normal to the flight path was not made because this correction to level acceleration data was very small.

Excess thrust and total JP-4 fuel flow are presented in figures 3 through 8 for the J79 engine plus rocket engine and J79 engine only, as functions of flight Mach number and ambient air temperatures.

■ Zoom Climb Performance

To analyze the zoom maneuver, the potential and kinetic energy relationship at the apex of a zoom maneuver was developed from ballistic trajectory theory. In ballistic trajectory theory (assuming no drag) the motion of

a projectile is subject to a constant acceleration, g , directed downward. There is no horizontal component of acceleration, therefore the horizontal velocity is constant. (reference 12)

$$V_p = V_o \cos \gamma \quad (1)$$

where

V_p is the horizontal speed at the peak referenced to a fixed point on the surface of the earth (fps)

V_o is the initial speed along the flight path referenced to a fixed point on the surface of the earth (fps)

γ is the initial flight path angle from horizontal (deg)

Ballistically, the specific energy of the projectile at the peak of a trajectory is equal to the initial specific energy along the flight path.

$$\frac{V_o^2}{2g} = \frac{V_p^2}{2g} + \Delta h \quad (2)$$

where

g is the acceleration of gravity (ft per sec²)

Δh is the tapeline altitude gained from pullup to peak (ft)

Equation (1) was substituted into equation (2):

$$\frac{V_p^2}{2g \cos^2 \gamma} = \frac{V_p^2}{2g} + \Delta h \quad (3)$$

$$\Delta h = \frac{V_p^2 - V_p^2 \cos^2 \gamma}{2g \cos^2 \gamma} \quad (4)$$

$$\Delta h = \frac{V_p^2 \tan^2 \gamma}{2g} \quad (5)$$

therefore

$$\gamma = \tan^{-1} \left[\frac{2g\Delta h}{V_p^2} \right]^{1/2} \quad (6)$$

The initial effective flight path angle of the aircraft, γ_{eff} , was calculated using equation (6) above and plotted vs the actual maximum pitch angle of the aircraft as obtained from the oscillograph for 8 and 11 degrees angle of attack (figure 9). Theoretically the specific energy at the peak is equal to the specific energy at pullup plus the average rocket engine specific energy input from pullup to burnout.

$$\frac{V_o^2}{2g} + (E/W)_{R_s} = \frac{V_p^2}{2g} + \Delta h \quad (7)$$

where

$(E/W)_{R_s}$ is the average specific energy input of rocket engine from pullup to burnout (ft)

In actual practice the specific energy at the peak deviated from this theory because of energy changes resulting from aircraft drag and J79 engine thrust.

Test day "k" factors (k_t) were determined by the following equation to account for variations in these parameters from one flight to the next:

$$k_t = \frac{\frac{V_o^2}{2g} + (E/W)_{R_t}}{\frac{V_p^2}{2g} + \Delta h_t} \quad (8)$$

where

$(E/W)_{R_t}$ denotes test day specific energy input of the rocket engine from pullup to burnout (ft).

The test day "k" factors calculated using equation (8) were plotted vs the initial effective flight path angle in figure 10. An average factor (k_s) of 1.14 was used to standardize the data.

The following equation was used to calculate the test specific energy of the rocket engine:

$$(E/W)_{R_t} = \frac{(F_R)(t)(V)}{W} \quad (9)$$

where

F_R is the rocket engine thrust (lb)

V is the average speed along the flight path, referenced to a fixed point on the surface of the earth during the time the rocket engine was burning (fps)

t is the time the rocket engine burned from pullup to burnout (sec)

W is the average weight of the aircraft from pullup to rocket engine burnout (lb)

The thrust of the rocket engine at altitude was calculated using the techniques in reference 13 and the following equation:

$$F_{R_{alt}} = (P_{C_{d/s}}) A_T C_{F_{SL}} - A_e (P_{a_{alt}} - P_{a_{SL}}) \quad (10)$$

where

- $F_{R_{alt}}$ is the rocket engine thrust at altitude (lb)
- $P_{C_{d/s}}$ is the rocket engine combustion chamber pressure (psia)
- A_T is the rocket engine throat area (6.5 in.²)
- $C_{F_{SL}}$ is the sea level thrust coefficient (dimensionless)
- A_e is the nozzle exit area (79.2 in.²)
- $P_{a_{alt}}$ is the ambient pressure at altitude (psi)

Equation (8) can now be expressed in terms of standard (average) conditions as

$$\frac{V_{O_s}^2}{2g} + (E/W)_{R_s} = \frac{V_{P_s}^2}{2g} + \Delta h_s \quad (11)$$

Equations (13) and (16) were developed to determine how the addition of the rocket engine energy was distributed between velocity and altitude at the peak. To determine the velocity at the peak, equation (5) was substituted into equation (11) to give

$$\frac{V_{P_s}^2}{2g} \left(1 + \tan^2 \gamma \right) =$$

$$\frac{1}{2gk_s} \left[V_{O_s}^2 + 2g (E/W)_{R_s} \right] \quad (12)$$

$$V_{P_s} = \left(\frac{\cos^2 \gamma}{k_s} \left[V_{O_s}^2 + 2g (E/W)_{R_s} \right] \right)^{1/2} \quad (13)$$

To determine the altitude gained from pullup to peak, equation (5) was rewritten in terms of V_p :

$$V_p^2 = \frac{2g \Delta h}{\tan^2 \gamma} \quad (14)$$

Equation (14) was substituted into equation (11) and rewritten:

$$\Delta h_s \left(1 + \frac{1}{\tan^2 \gamma} \right) = \frac{1}{k_s} \left[\frac{V_{O_s}^2}{2g} + (E/W)_{R_s} \right] \quad (15)$$

$$\Delta h_s = \frac{\sin^2 \gamma}{k_s} \left[\frac{V_{O_s}^2}{2g} + (E/W)_{R_s} \right] \quad (16)$$

Equations (18) and (20) were developed to standardize the velocity at the peak, V_p , and the altitude gained from pullup to peak, Δh . The subscript (s) denotes standardized parameters and the subscript (t) denotes test parameters.

The equation for the standardized velocity at the peak was developed from equation (13):

$$\frac{V_{P_s}}{V_{P_t}} = \frac{\left(\frac{\cos^2 \gamma}{k_s} \left[V_{O_s}^2 + 2g (E/W)_{R_s} \right] \right)^{1/2}}{\left(\frac{\cos^2 \gamma}{k_t} \left[V_{O_t}^2 + 2g (E/W)_{R_t} \right] \right)^{1/2}} \quad (17)$$

$$V_{P_s} = V_{P_t} \left(\frac{k_t}{k_s} \right) \left[\frac{V_{O_s}^2 + 2g (E/W)_{R_s}}{V_{O_t}^2 + 2g (E/W)_{R_t}} \right]^{1/2} \quad (18)$$

The test day ground speeds at the peak were standardized to three pullup ground speeds and plotted vs the initial effective flight path angle in figure 11.

The equation for standardizing the increment of altitude gained from pullup to peak was developed from equation (16):

$$\frac{\Delta h_s}{\Delta h_t} = \frac{\frac{\sin^2 \gamma}{k_s} \left[\frac{V_{O_s}^2}{2g} + (E/W)_{R_s} \right]}{\frac{\sin^2 \gamma}{k_t} \left[\frac{V_{O_t}^2}{2g} + (E/W)_{R_t} \right]} \quad (19)$$

$$\Delta h_s = \Delta h_t \frac{k_t}{k_s} \left[\frac{V_{O_s}^2 + 2g (E/W)_{R_s}}{V_{O_t}^2 + 2g (E/W)_{R_t}} \right] \quad (20)$$

The test day increments of altitude gained from pullup to peak were standardized to three pullup ground speeds and plotted vs initial effective flight path angle in figure 12.

A Zoom Climb Performance chart, figure 13, was constructed to allow the Aerospace Research Pilot School students to completely preplan a zoom climb maneuver. This chart is applicable if the following conditions are met:

1. Maximum afterburner plus rocket engine thrust at pullup.
2. A 3.5 g pullup at 35 000 feet pressure altitude to an aim pitch attitude.

3. Pitch attitude was held until the angle of attack increased to a maximum of 8 degrees.
4. Eight degrees angle of attack was held throughout the remainder of the maneuver.
5. Afterburner shutdown at 63 000 feet tapeline altitude.
6. Main J79 engine shutdown at 80 000 feet tapeline altitude.
7. Rocket engine specific energy input from pullup to burnout was 43 400 feet.
8. Average weight during zoom maneuver was 17 000 pounds.
9. Average "k" factor was 1.14.

The Zoom Climb Performance chart can be used to determine the altitude, Mach number, and dynamic pressure at the peak if the ambient temperature and wind component at the acceleration altitude, and the ambient temperature, pressure, and wind component at the peak altitude are known. To plan for a high pullup Mach number, 50-degree pitch attitude mission, it may be necessary to iterate through the chart several times, adjusting the pullup Mach number and accounting for conditions at the peak each time, so that the minimum dynamic pressure limitation at the peak will not be exceeded.

Plot number one of the chart, Pullup Mach Number vs True Airspeed at Pullup, as a function of ambient temperature, was constructed using the following equation:

$$V_t = 65.81 M \sqrt{T_a} \quad (21)$$

where

V_t is the true airspeed (fps)
 M is the pullup Mach number
 T_a is the ambient temperature (deg K)

Plot number two, Ground Speed at Pullup vs True Airspeed at Pullup as a function of the wind component, was constructed using the following equation:

$$V_g = V_t \pm \Delta V_w \quad (22)$$

where

V_g is the speed over the ground at pullup (fps)
 ΔV_w is the wind component along the flight path at pullup (fps)
(+ for tailwind and
- for headwind)

Plot number three, Ground Speed at Pullup vs Altitude gained from pullup to peak as a function of pitch attitude, was constructed using figure 9, to determine the initial effective flight path angle and using figure 12 to determine the altitude gained.

Plot number four, Ground Speed at Pullup vs Ground Speed at Peak as a function of pitch attitude, was constructed using figure 9 to determine the initial effective flight path angle and using figure 11 to determine the ground speed at the peak.

Plot number five, Ground Speed at Peak vs True Airspeed at Peak as a function of the wind component at the peak, was constructed in a manner similar to the second figure of the chart.

Plot number six, True Airspeed at Peak vs Mach Number at Peak as a function of the ambient temperature at the peak, was constructed in a manner similar to the first figure of the chart.

Plot number seven, Mach Number at Peak vs Dynamic Pressure at Peak as a function of ambient pressure at the peak, was constructed using the following equation:

$$q = 0.7 P_a M^2 \quad (23)$$

where

q is the dynamic pressure (psf)
 P_a is the ambient pressure (psf)
 M is the Mach number at the peak

Sample Problem:

For a planned 2.0 Mach number pullup, 40-degree pitch attitude mission, assume that the following ambient conditions at the 35 000 feet pressure altitude level have been obtained from base weather: (1) Ambient air temperature -53 degrees C; (2) Wind 66 knots from 305 degrees relative to true north; (3) Tapeline altitude 36 000 feet (geopotential altitude referenced to mean sea level). Also assume an aircraft heading of 260 degrees relative to true north (245 degrees magnetic at the AFFTC).

- a. At a reported wind velocity of 66 knots and an angle between wind and aircraft headings of 45 degrees (305 deg - 260 deg), the headwind component is 47 knots.
- b. Enter the Zoom Climb Performance chart, figure 13,

at a pullup Mach number of 2.0 and traverse right horizontally to an ambient temperature of -53 degrees C at pullup. Traverse vertically upward to a headwind component of 47 knots at pullup. At this point a pullup ground speed of 1870 fps and a true airspeed of 1950 fps are obtained.

- c. Traverse right horizontally to the pitch attitude line of 40 degrees, and then vertically to obtain a tapeline altitude increment of 55 750 feet. A tapeline altitude of 91 750 feet at the peak of the zoom is obtained by adding this increment to the tapeline altitude at pullup.

Assume that the following ambient conditions for the 91 750 feet tapeline altitude level have been obtained from base weather:

- (1) Ambient air temperature -47 degrees C; (2) Wind 28 knots from 35 degrees relative to true north; (3) Ambient pressure 35 psf.

- d. Traverse right horizontally at 1870 fps pullup ground speed to the next 40 degree pitch attitude line where a ground speed of 1380 fps at the peak of the zoom is obtained.
- e. At a reported wind velocity of 28 knots and an angle of 45 degrees between wind and aircraft heading, the tailwind component is calculated to be 20 knots.
- f. Traverse vertically downward to the 20 knot tailwind component line where a true airspeed of 1346 fps at the peak of the zoom is obtained.

- g. Traverse left horizontally to the ambient temperature of -47 degrees C at the peak and then vertically downward to obtain a Mach number of 1.35 at the peak.

- h. From this same vertical line at the given ambient pressure of 35 psf on plot seven, traverse right horizontally to a dynamic pressure of 45 psf at the peak of the zoom.

■ Cruise Performance

Cruise performance was obtained from stabilized conditions of speed and fuel flow at constant weight-pressure ratios maintained by flying successively higher altitudes to compensate for decreasing aircraft weight due to fuel consumption. Basic relations to determine Mach number, altitude, etc., are from reference 14.

Test fuel flow was corrected to a standard day altitude of 35 000 ft and standard temperature at that altitude by the following relationships:

$$d\left(\frac{W_f}{\delta t_2}\right) = \frac{\partial\left(\frac{W_f}{\delta t_2}\right)}{\partial N} dN + \frac{\partial\left(\frac{W_f}{\delta t_2}\right)}{\partial T_{t_2}} dT_{t_2}$$

where

$$dN = \frac{\frac{\partial\left(\frac{F_n}{\delta t_2}\right)}{\partial T_{t_2}}}{\frac{\partial\left(\frac{F_n}{\delta t_2}\right)}{\partial N}} dT_{t_2}$$

Slopes for these relationships were obtained from J79 engine curves supplied by the Lockheed-California

Company. Test fuel flow was further corrected for any existing excess thrust using the slope of fuel flow versus net thrust curve. The specific range of the NF-104A is shown in figure 14, and the range factor as compared to the F-104A can be seen in figure 15.

■ Pitot-Static System Calibration

A sketch of the nose boom and YAPS head installation on the NF-104A aircraft is shown in figure 16.

The airspeed system calibration in ground effect was obtained by plotting the change in altimeter reading during the takeoff roll vs indicated airspeed and converting this to a change in indicated velocity as per reference 15. The correction to be added can be seen in figure 17.

The airspeed system calibration out of ground effect was accomplished by the tower fly-by, stabilized pace, and smoke trail acceleration methods. Data reduction was accomplished in accordance with reference 14 and 16. Figure 18 presents the results of the airspeed system calibration.

A lag calibration of the static system was accomplished and data were reduced in accordance with reference 17. The results of this calibration can be seen in figure 19.

■ Static Thrust Calibration

The J79-GE-3B engine thrust, fuel flow, exhaust gas temperature, and nozzle area were plotted versus engine rotor speed in figures 20 and 21.

The LR121-N¹-1 rocket engine static thrust and thrust coefficients, obtained from the static thrust calibration, were corrected

to sea level standard day conditions using the techniques outlined in reference 13. The results can be seen in figure 22.

■ Stability and Control

The technique used to obtain lateral-directional stability data was to ignite the rocket engine at 2.0 Mach number and make a 3 g pullup at approximately constant Mach number to a pitch angle of 25 degrees. At a tapeline altitude of 63 000 feet the J79 engine power was reduced to military power. A zero g pushover was made at a tapeline altitude of 65 000 to 70 000 feet, and the aircraft attained level flight at a tapeline altitude of about 80 000 feet and 2.0 Mach number. The rocket and jet engines were then shutdown, the speed brakes extended, and the aircraft stabilized at the desired angle of attack. The aerodynamic yaw and roll gyro cutout switches were turned off and rudder pulses were performed at constant angle of attack while minimizing aileron inputs. The aircraft decelerated rapidly; therefore, to obtain high Mach number points, it became necessary to keep the engines running until after the angle of attack was stabilized, the speed brakes extended, and the dampers turned off. The primary advantage of this test technique was that recovery could be effected by decreasing the angle of attack in the event uncontrollable lateral-directional oscillations were encountered. The boundary for the lateral-directional oscillations which occurred during reentry because of an air duct spillage conditions as shown in figure 23. The effect on $C_{n\delta}$ and $C_{l\delta}$ of having the speed brakes extended is shown in figure 24.

The technique used to determine the minimum allowable q at the apex of a zoom climb maneuver

was to stabilize at 11 degrees angle of attack with the aerodynamic dampers on and the RCS dampers off. The angle of attack would then be slowly increased until the APC and the RCS "kicker" actuated. After APC and RCS "kicker" actuation occurred, an attempt was made to recover to stable flight at 11 degrees angle of attack using only aerodynamic controls. The time history shown in figure 26 is characteristic of proper pilot technique for controlling the aircraft.



PERFORMANCE PLOTS

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-3B Engine
 Takeoff Flaps, Maximum Power
 Sea Level, no Wind, Standard Day
 Gross Weight 21 900 lb

Test System
 Airspeed Calibration
 in Ground Effect
 (See figure 17)

Symbol	cg	Condition
○	-4.4	Fully Serviced (21 900 lb)
●	-2.7	Fully Serviced Less RCS Hydrogen Peroxide (21 750 lb)

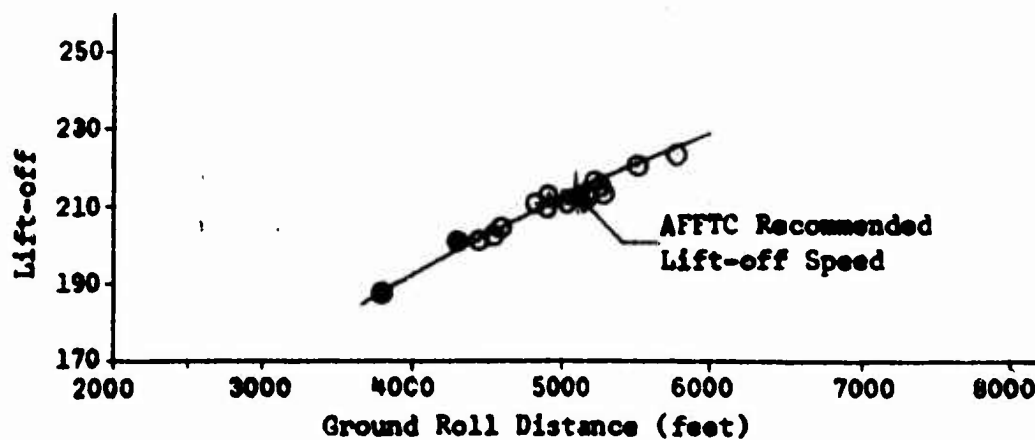
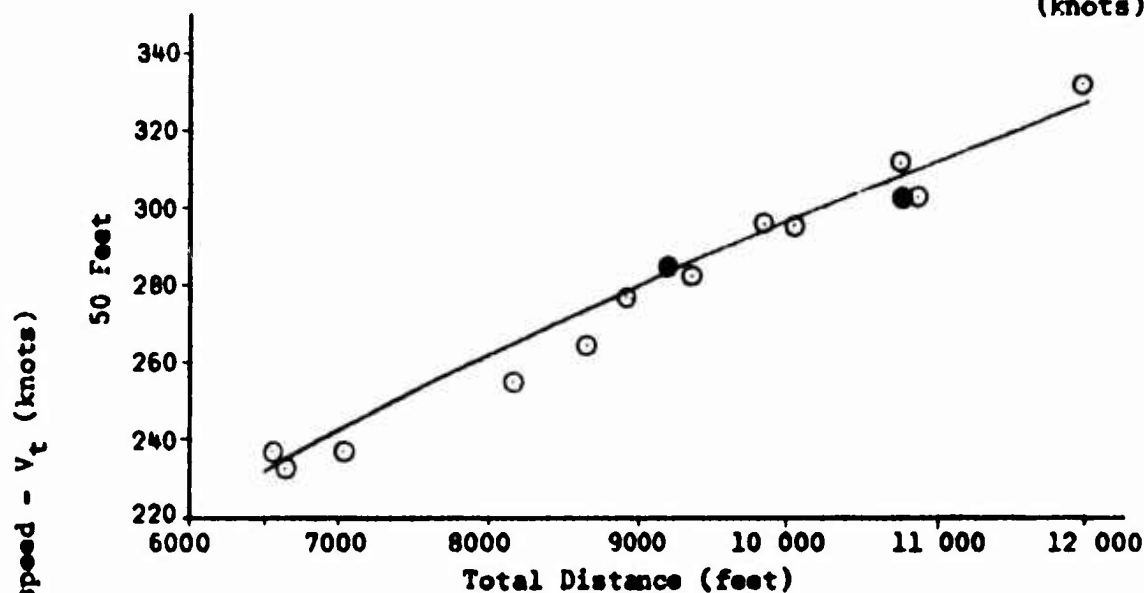
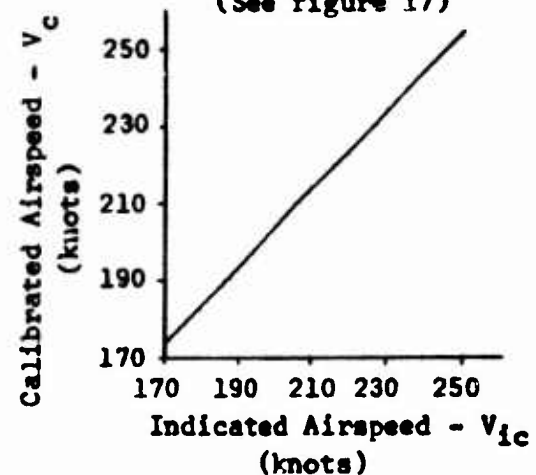


Figure 1. Takeoff Performance

NF-104A Aerospace Trainer
USAF S/N 56-0760
Military Power
Clean Configuration
Gross Weight at Sea Level = 21 400 lb

Symbol	Flight
◐	19
◑	24
○	26
◻	62
◕	63

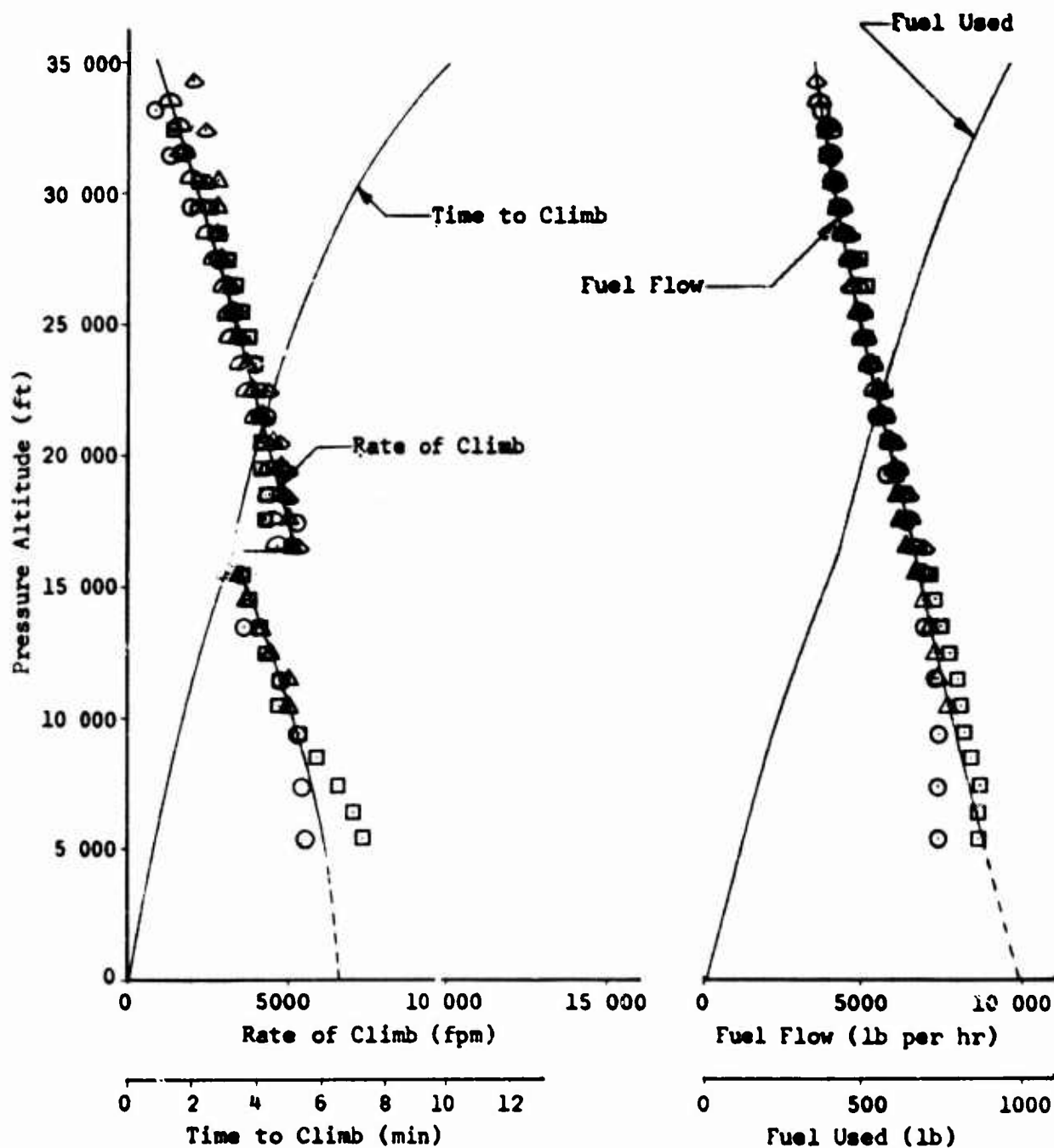


Figure 2. Subsonic Climb Performance

MF-104A Aerospace Trainer
USAF S/N 56-0760

Symbol	Flight
△	19
◊	24
○	26
□	62
△	63

Note: Allow 730 lb of fuel for taxi, maximum power takeoff, and acceleration to climb speed.

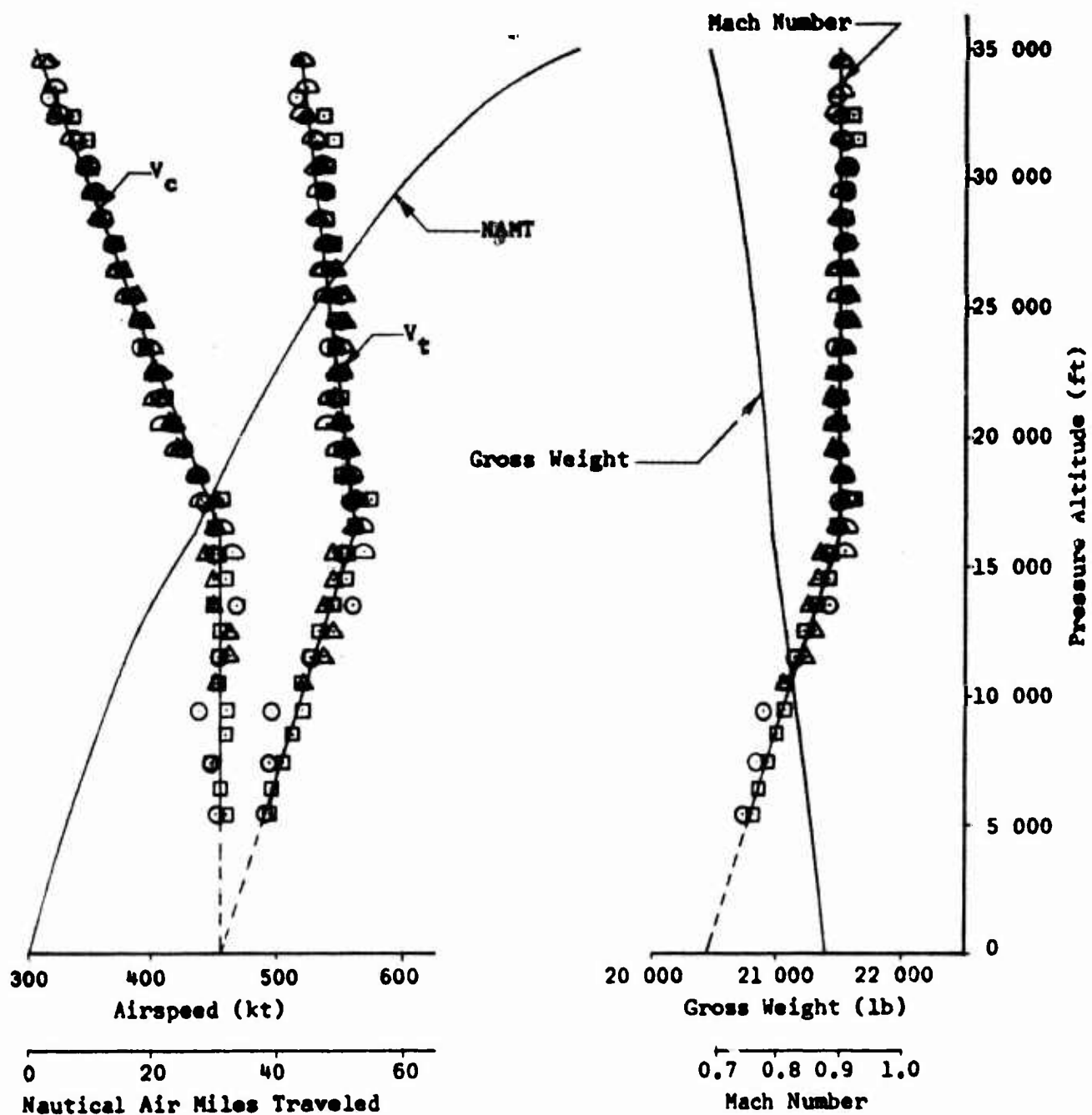


Figure 2. Subsonic Climb Performance (concluded)

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-38 Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet
 Standard Day

Weight at 0.9 Mach Number = 20 350 lb

Symbol	Flight
○	10
□	12
◇	17
◊	18
▽	19
∇	21
○	22
☆	24
△	25
Δ	26
▷	29
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

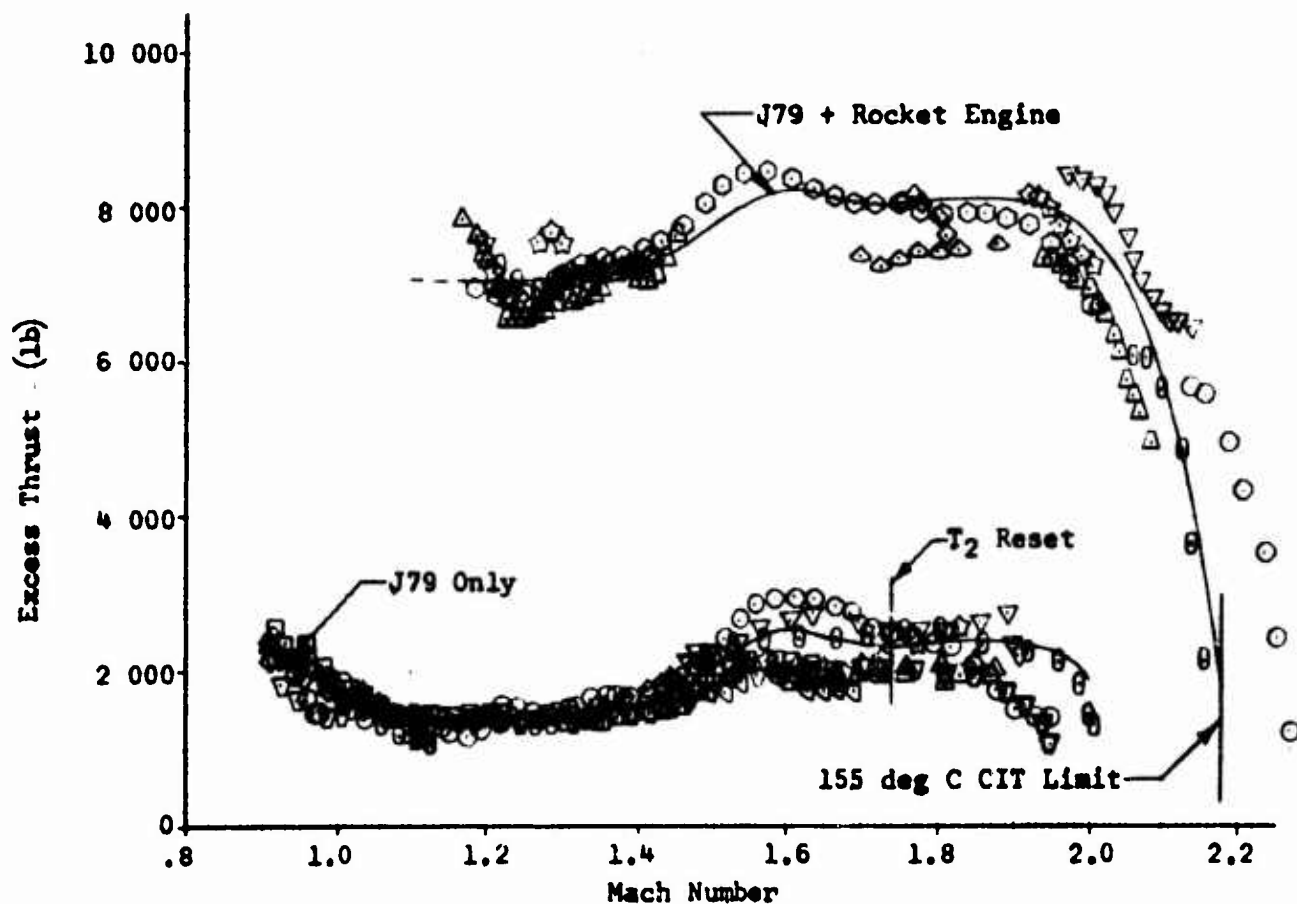


Figure 3. Maximum Power Acceleration

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-3B Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet Pressure Altitude
 10 deg C Hot Day (-44.3 deg C)
 Weight at 0.9 Mach Number = 20 350 lb

Symbol	Flight
○	10
□	12
◇	17
△	18
▽	19
▽	21
○	22
☆	24
△	25
△	26
▷	29
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

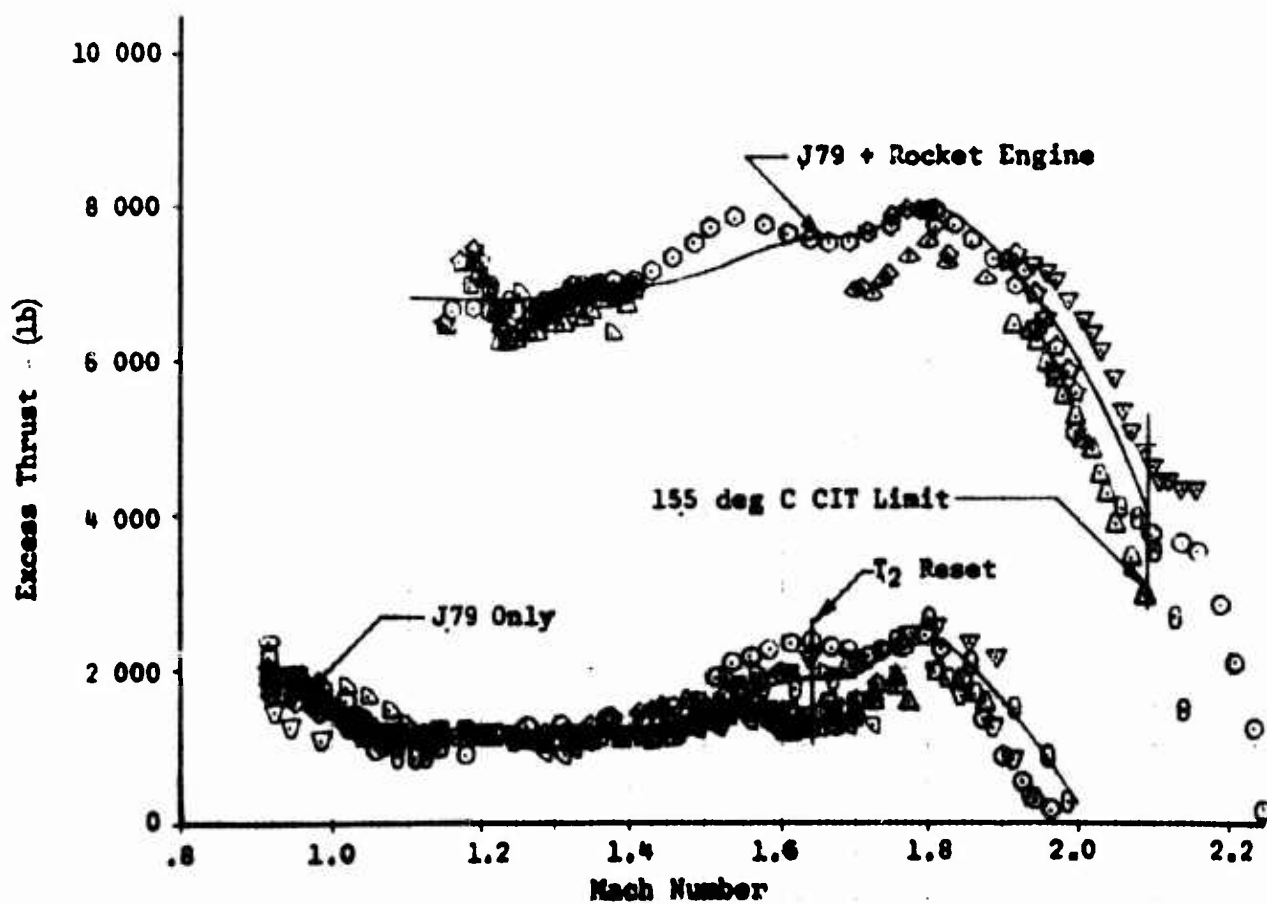


Figure 4. Maximum Power Acceleration

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-3B Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet Pressure Altitude
 10 deg C Cold Day (-64.3 deg C)
 Weight at 0.9 Mach Number = 20 350 lb

Symbol	Flight
○	10
□	12
◇	17
△	18
▽	19
▽	21
○	22
☆	24
△	25
△	26
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach Number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

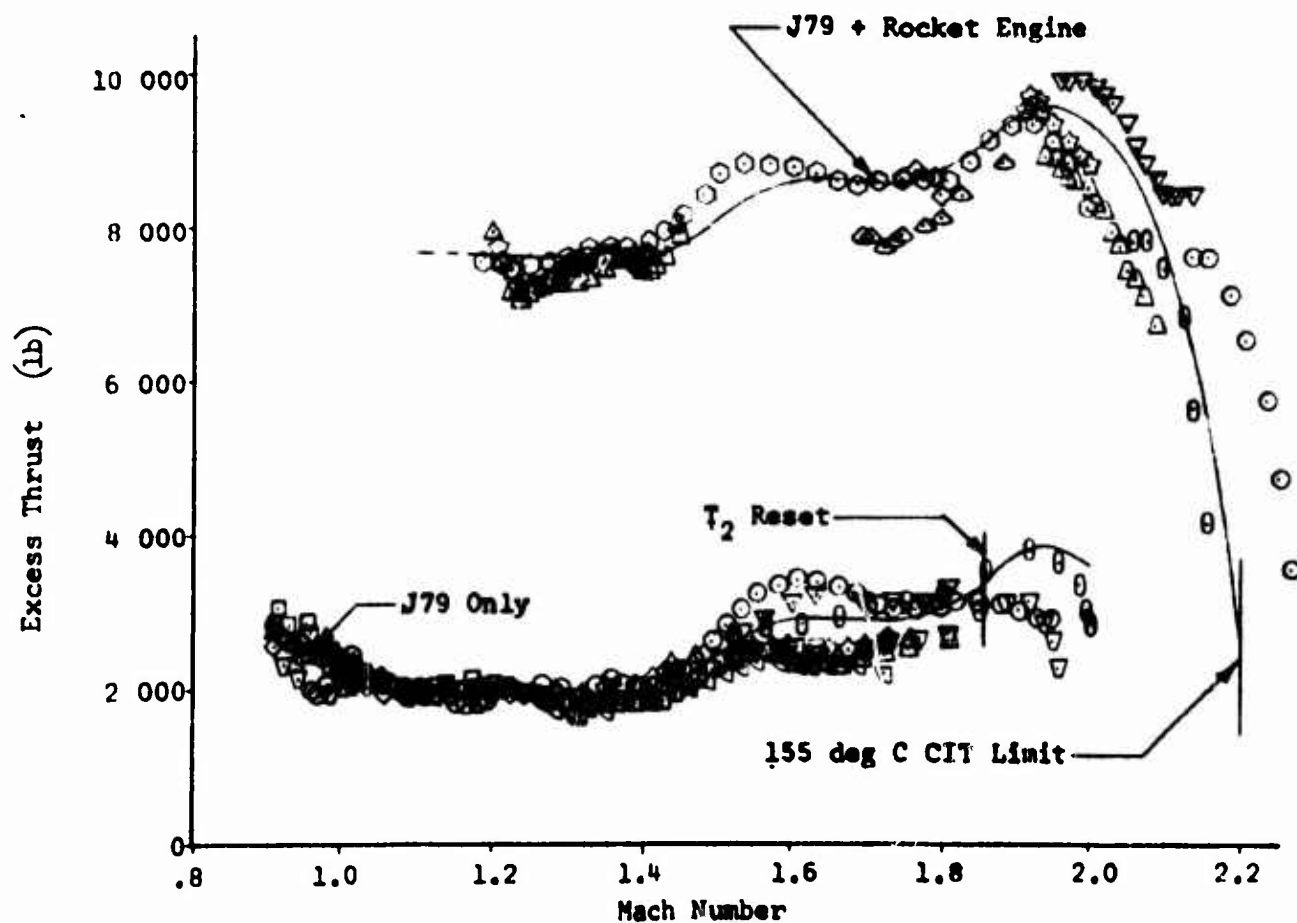


Figure 5. Maximum Power Acceleration

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-3B Turbojet Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet
 Standard Day

Symbol	Flight
○	10
□	12
◇	17
△	18
▽	19
▽	21
○	22
☆	24
△	25
△	26
▷	29
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

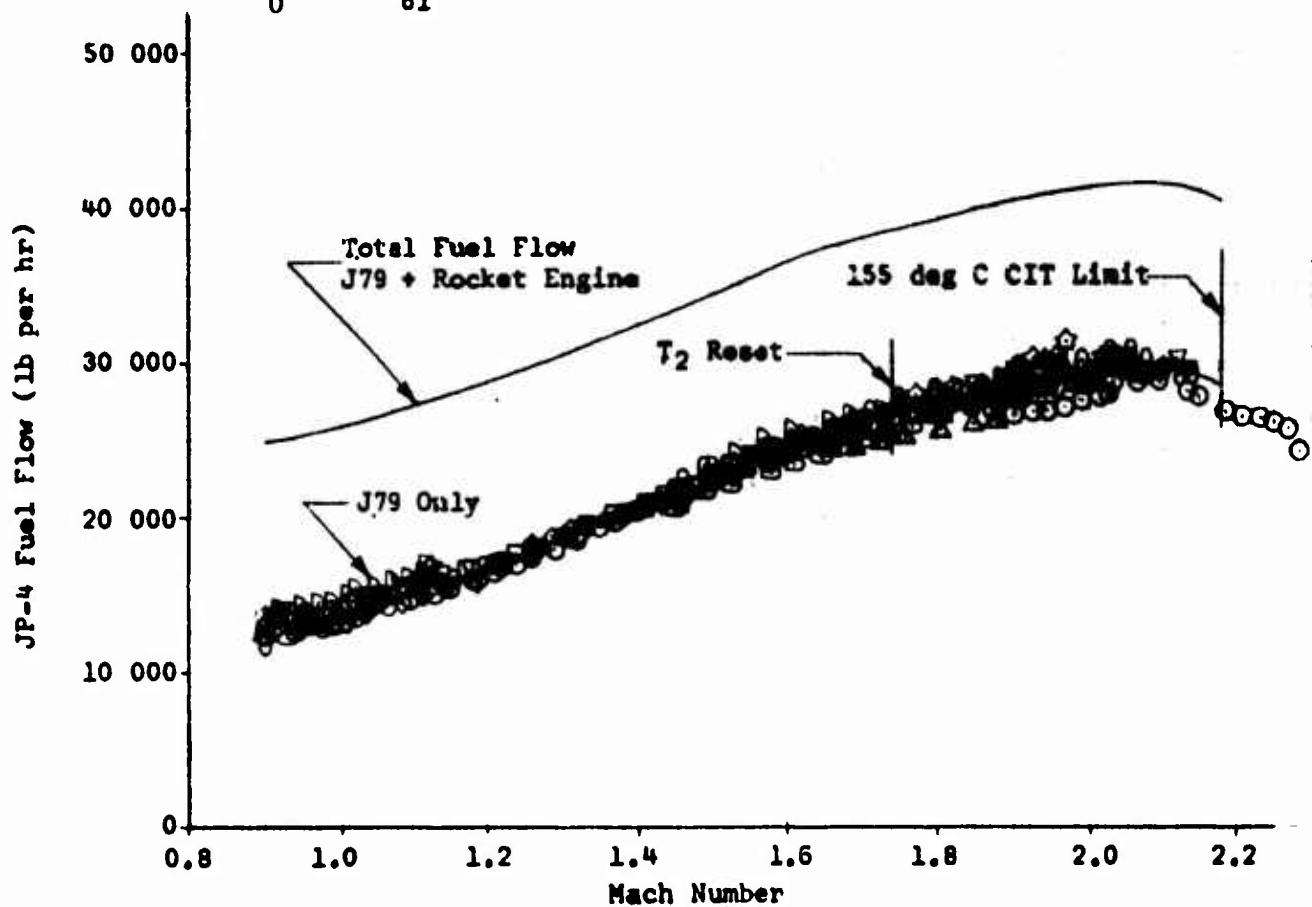


Figure 6. Maximum Power Acceleration

NF-104A Aerospace Trainer
 USAF S/N 54-0756 and 56-0760
 J79-GE-3B Turbojet Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet Pressure Altitude
 10 deg C Hot Day (-44.3 deg C)

Symbol	Flight
○	10
□	12
◇	17
△	18
▽	19
▽	21
○	22
☆	24
△	25
△	26
▷	29
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

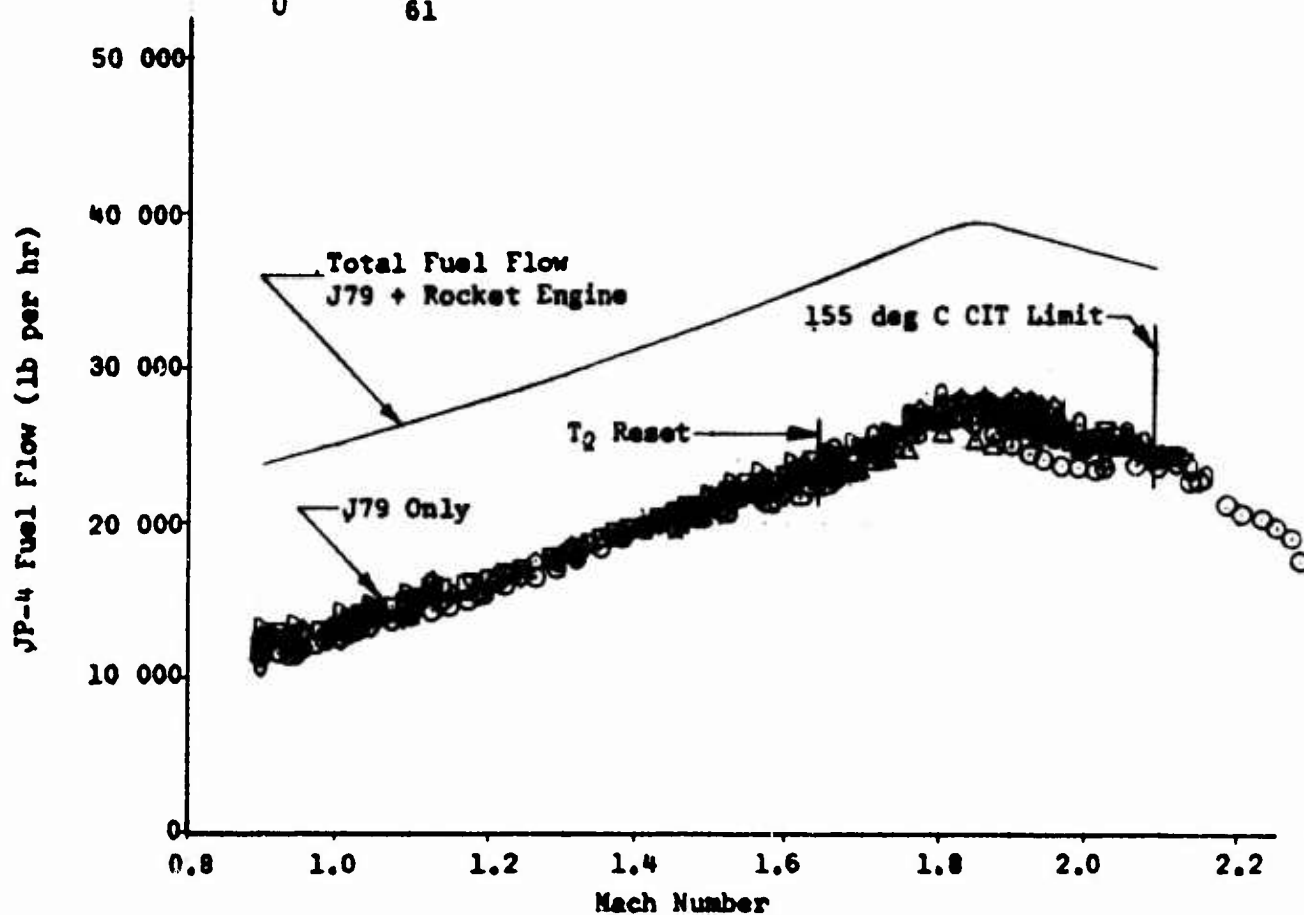


Figure 7. Maximum Power Acceleration

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 J79-GE-3B Turbojet Engine
 LR121-NA-1 Rocket Engine
 35 000 Feet Pressure Altitude
 10 deg C Cold Day (-64.3 deg C)

Symbol	Flight
○	10
□	12
◇	17
△	18
▽	19
▽	21
○	22
☆	24
△	25
△	26
▷	29
▽	30
○	36
○	61

Notes:

1. "B" bypass flaps opened at 1.6 Mach number.
2. T_2 reset occurred at 79 deg C CIT.
3. EGT was monitored at 615 deg C on test day.
4. J79 engine limited to 155 deg C CIT.

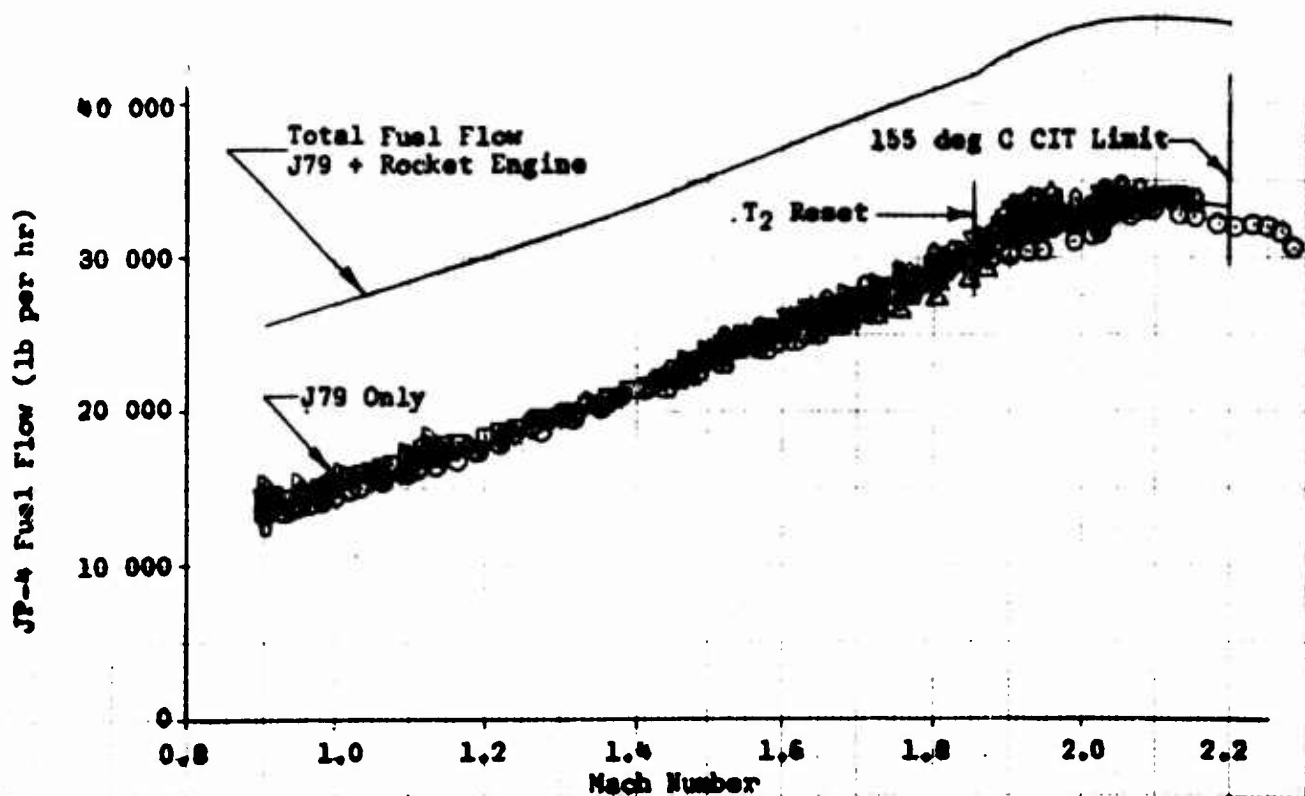


Figure 2. Maximum Power Acceleration

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

<u>Symbol</u>	<u>Angle of Attack (deg)</u>
○	8
□	11

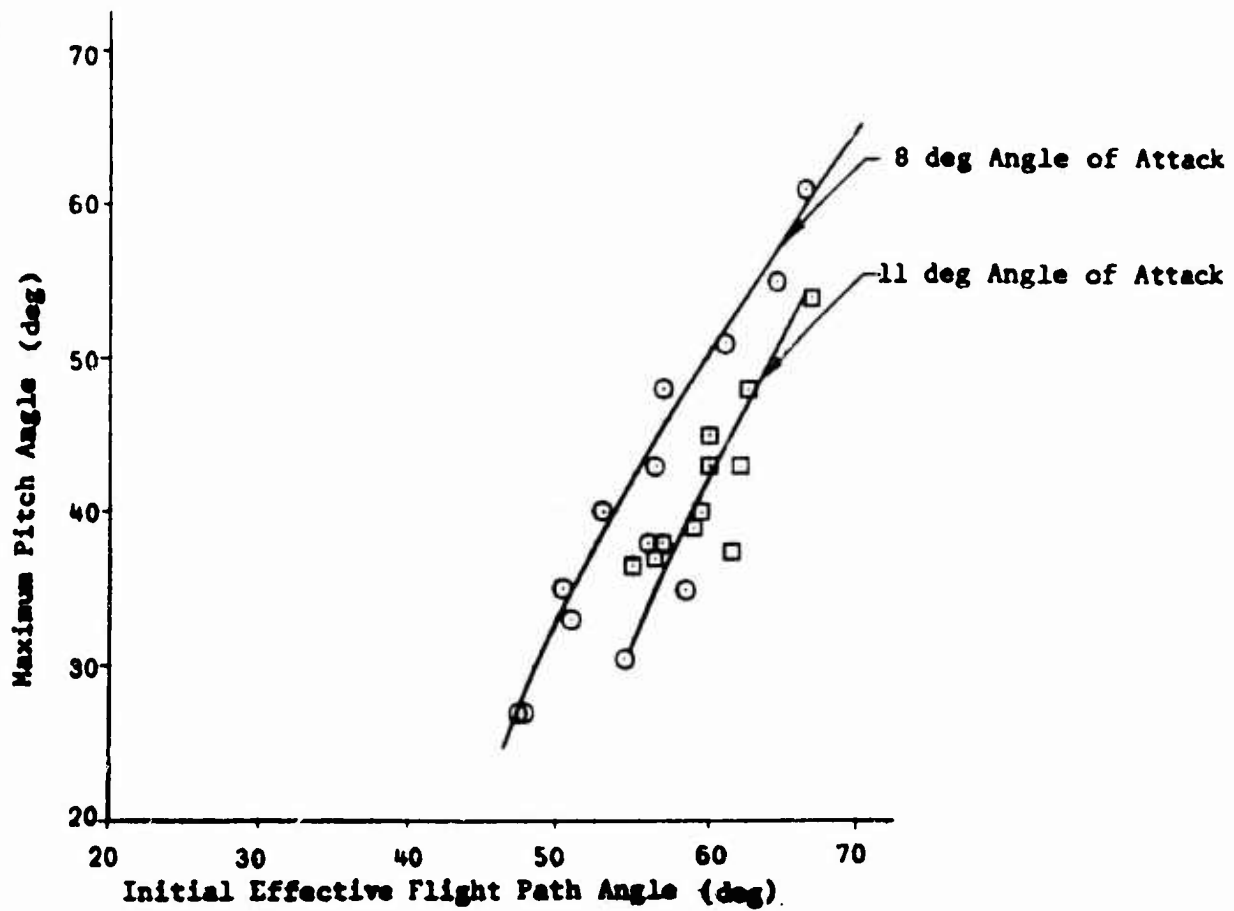


Figure 9. Zoom Climb Performance

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

Symbol	Angle of Attack (deg)
○	8
□	11

Note: "k" factor was used to account for variations in aircraft drag and J79 engine thrust during maneuver.

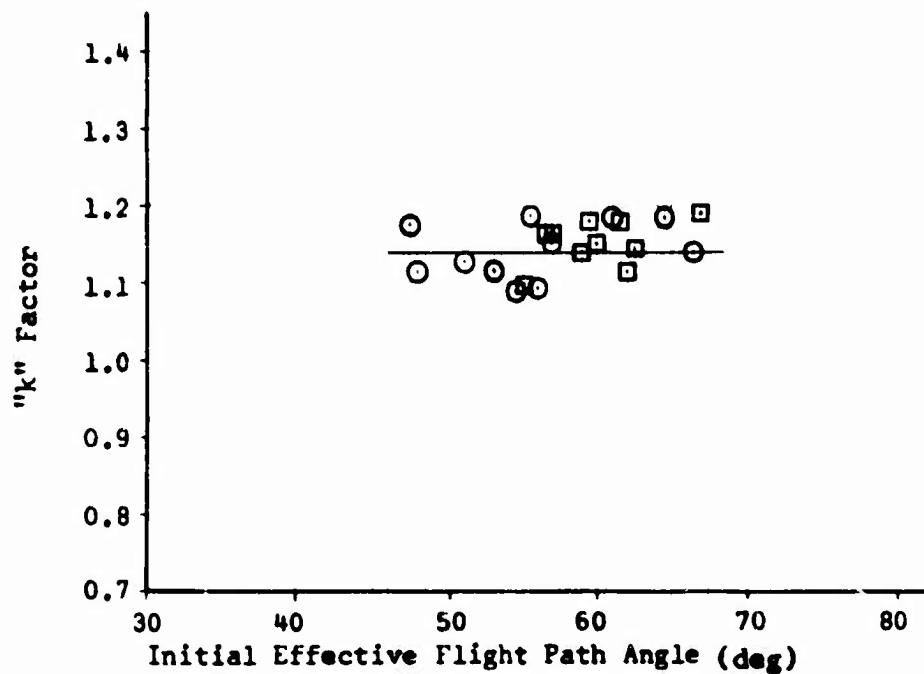


Figure 10. Zoom Climb Performance

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

<u>Symbol</u>	<u>Ground Speed at Pullup (fps)</u>	<u>*Mach Number at Pullup</u>
○	1850	1.9
□	1946	2.0
△	2092	2.15

NOTE: This plot is valid for the flight techniques and conditions listed on page 32
 *Mach number shown is for standard day no wind conditions

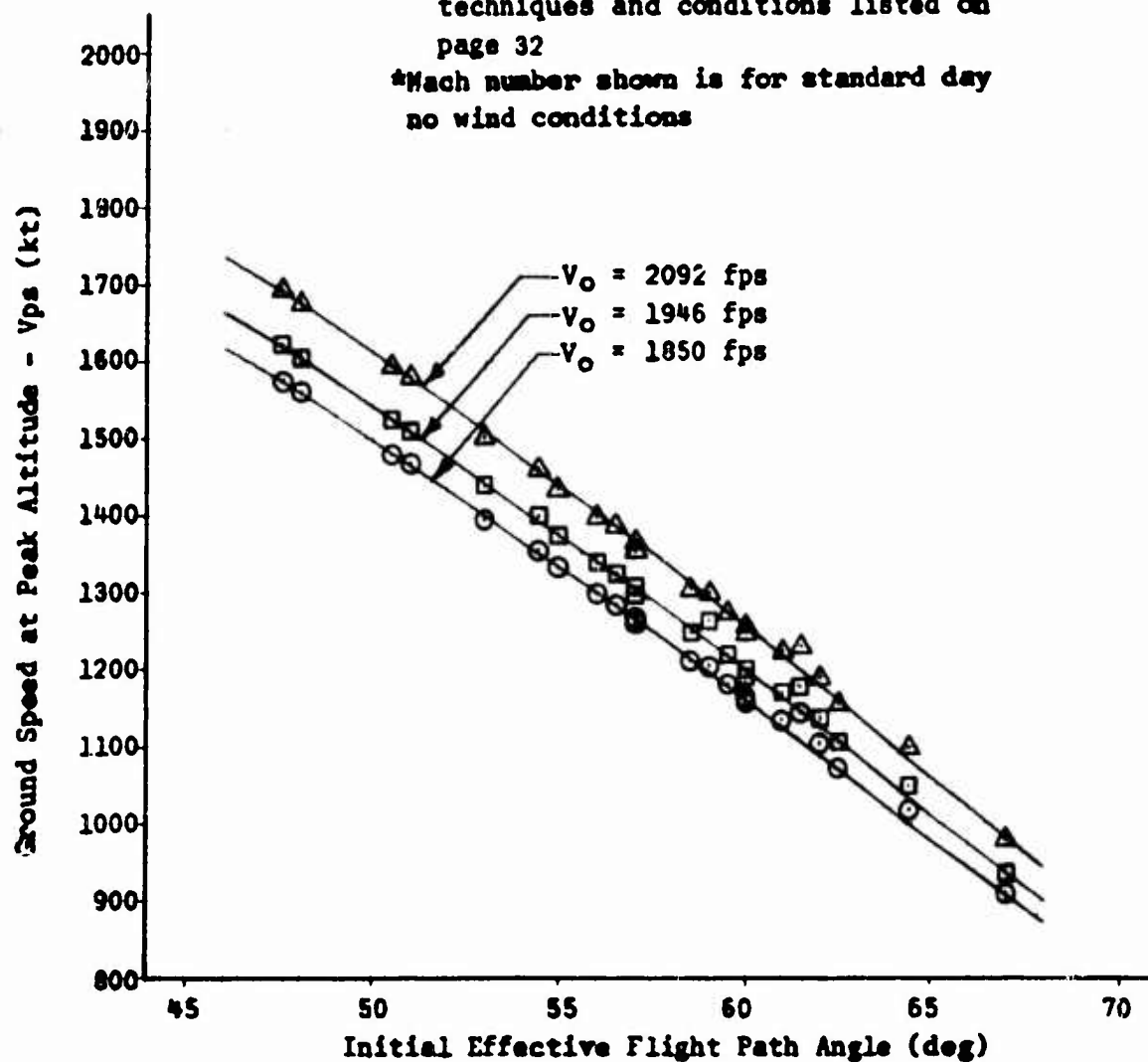


Figure 11. Zoom Climb Performance

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760

Symbol	Ground Speed at Pullup (fps)	*Mach Number at Pullup
○	1850	1.9
□	1946	2.0
△	2092	2.15

NOTE: This plot is valid for the flight techniques and conditions listed on page 32

*Mach number shown is for standard day no wind conditions

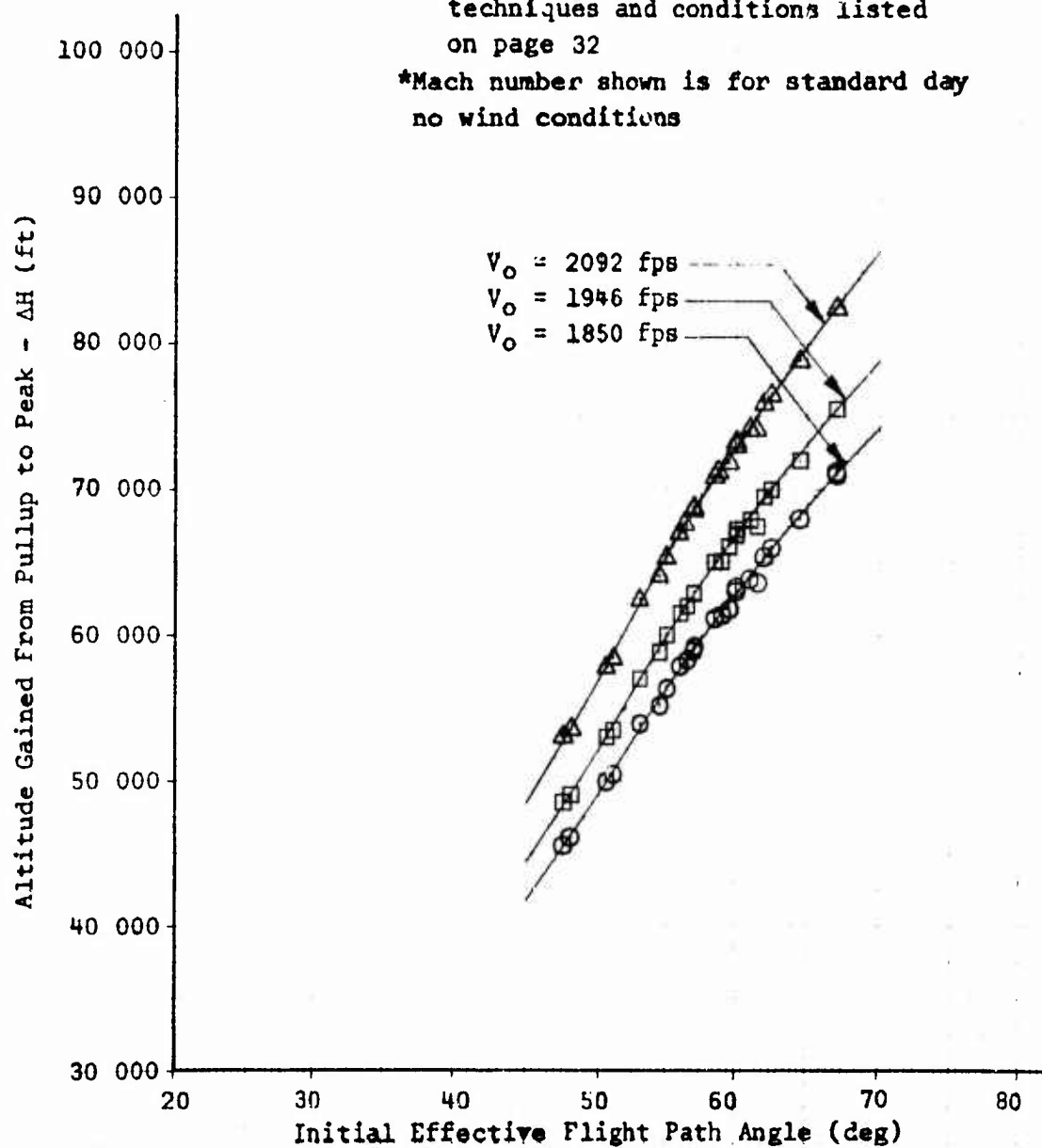


Figure 12 . Zoom Climb Performance

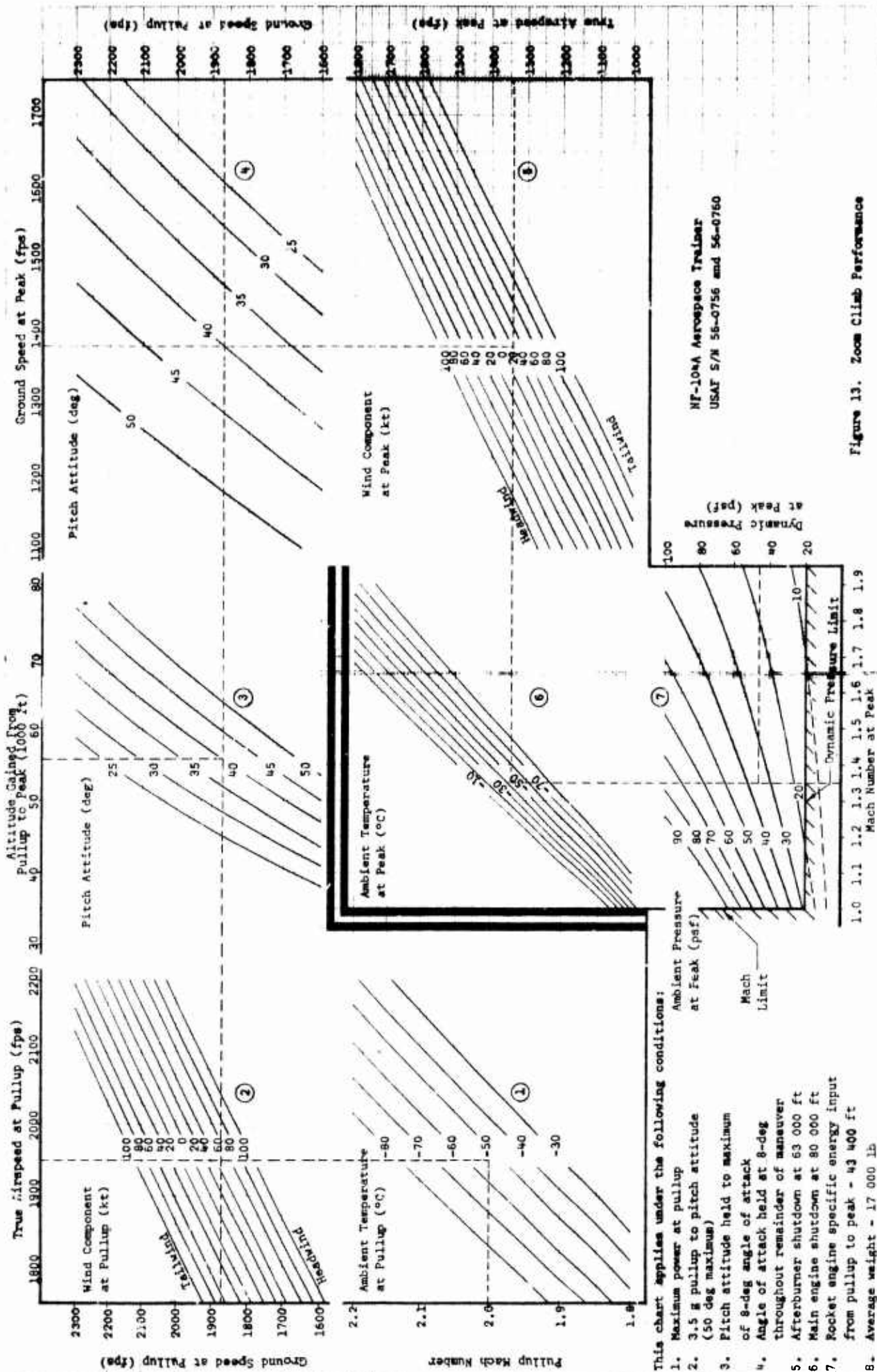


Figure 13. Zoom Climb Performance

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 35 000 Feet Pressure Altitude
 Cruise Configuration

Symbol	Weight (lb)	W/S (lb)
□	16 000	68 000
○	16 700	71 000
△	17 400	74 000

Note: x Denotes recommended cruise Mach number.

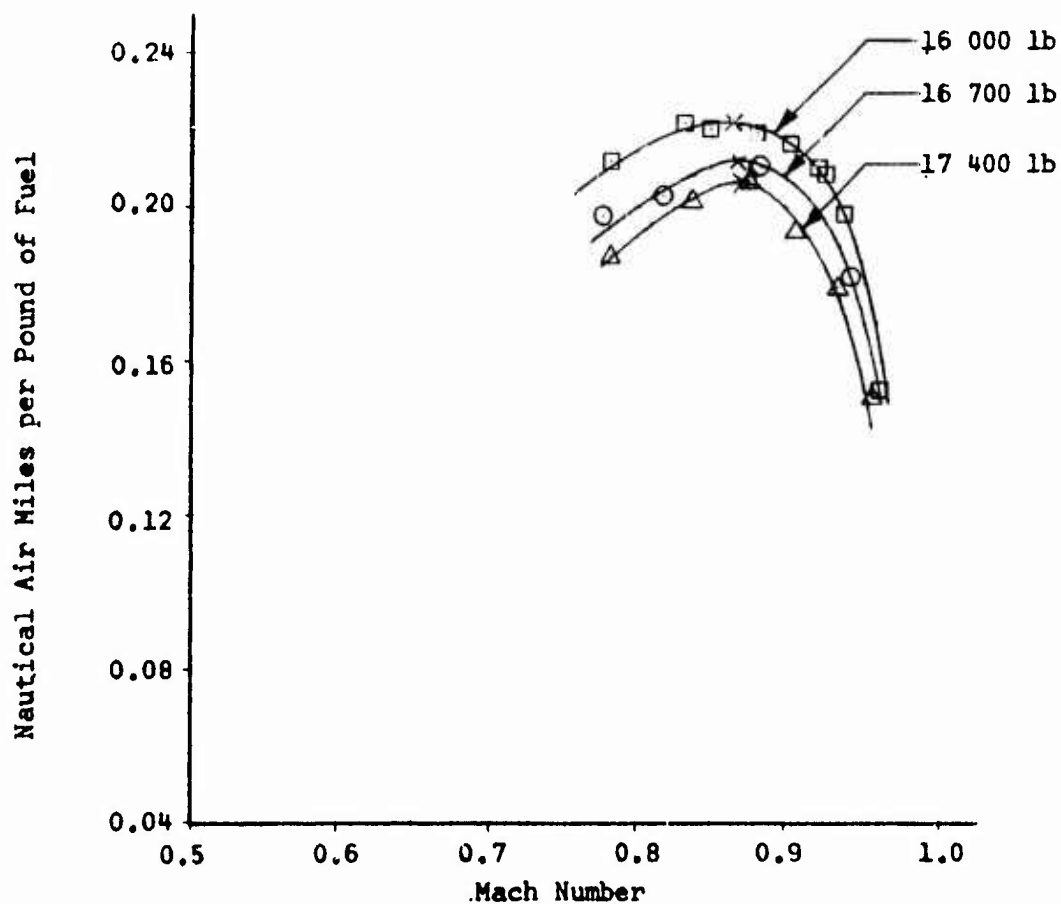


Figure 14. Specific Range

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 35 000 Feet Pressure Altitude
 Cruise Configuration

Symbol	Weight (lb)
□	16 000
○	16 700
△	17 400

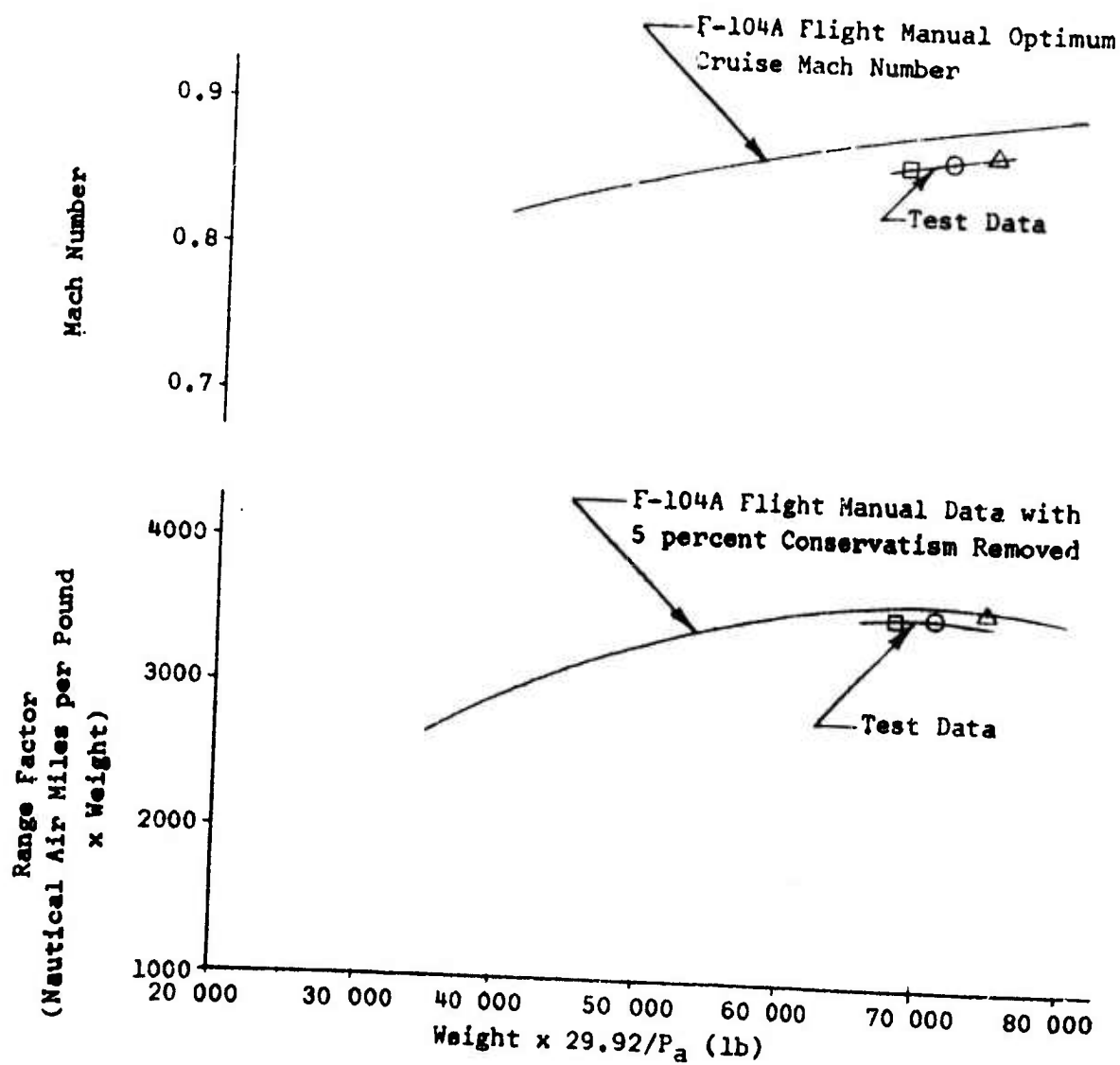


Figure 15. Maximum Range Factor

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

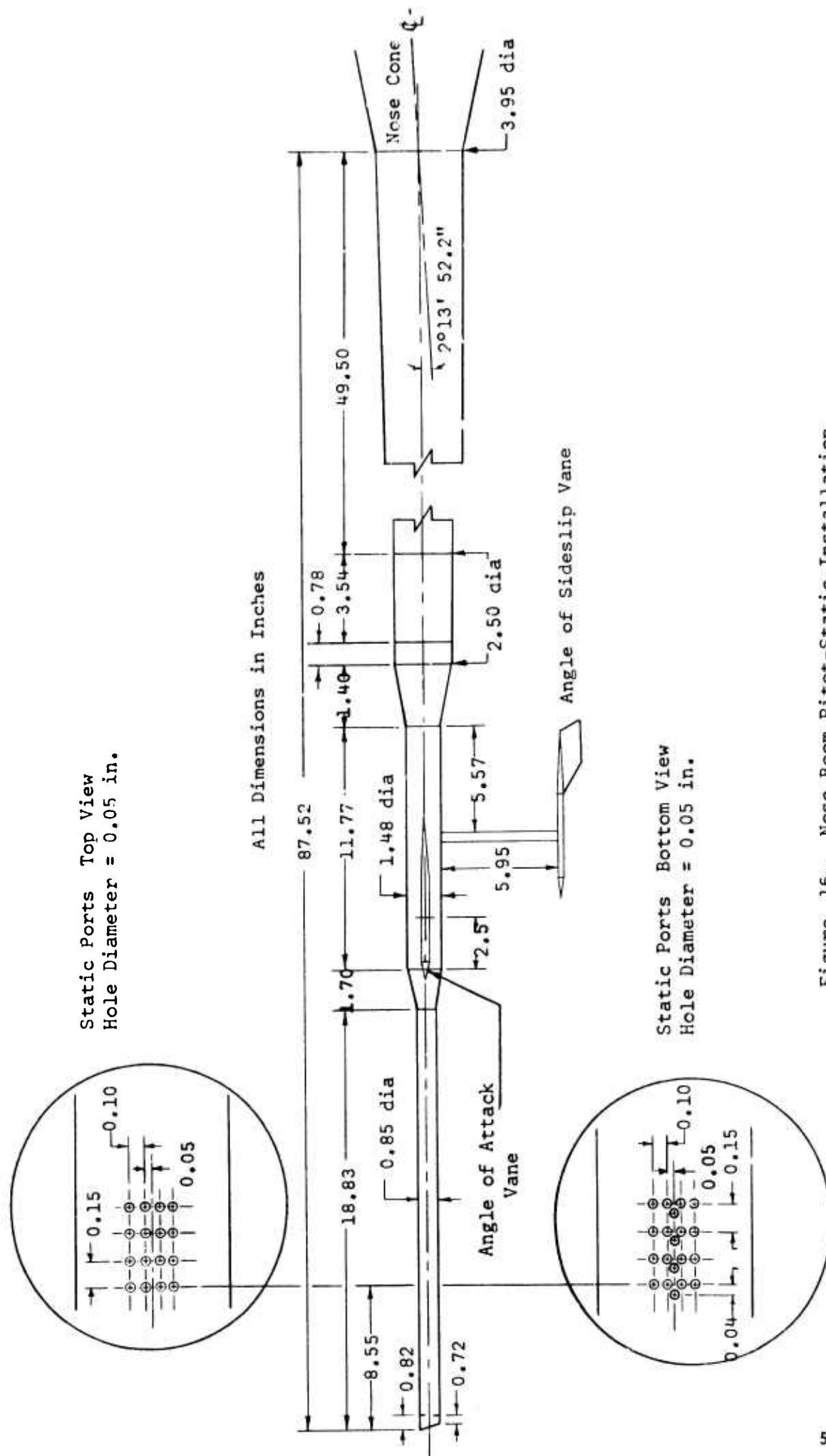


Figure 16. Nose Boom Pitot-Static Installation

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

Pressure Altitude 2300 Feet
Gross Weight = 22 000 lb

Symbol	Flight
△	36
○	47
□	52
◇	56

Data obtained during takeoff ground runs

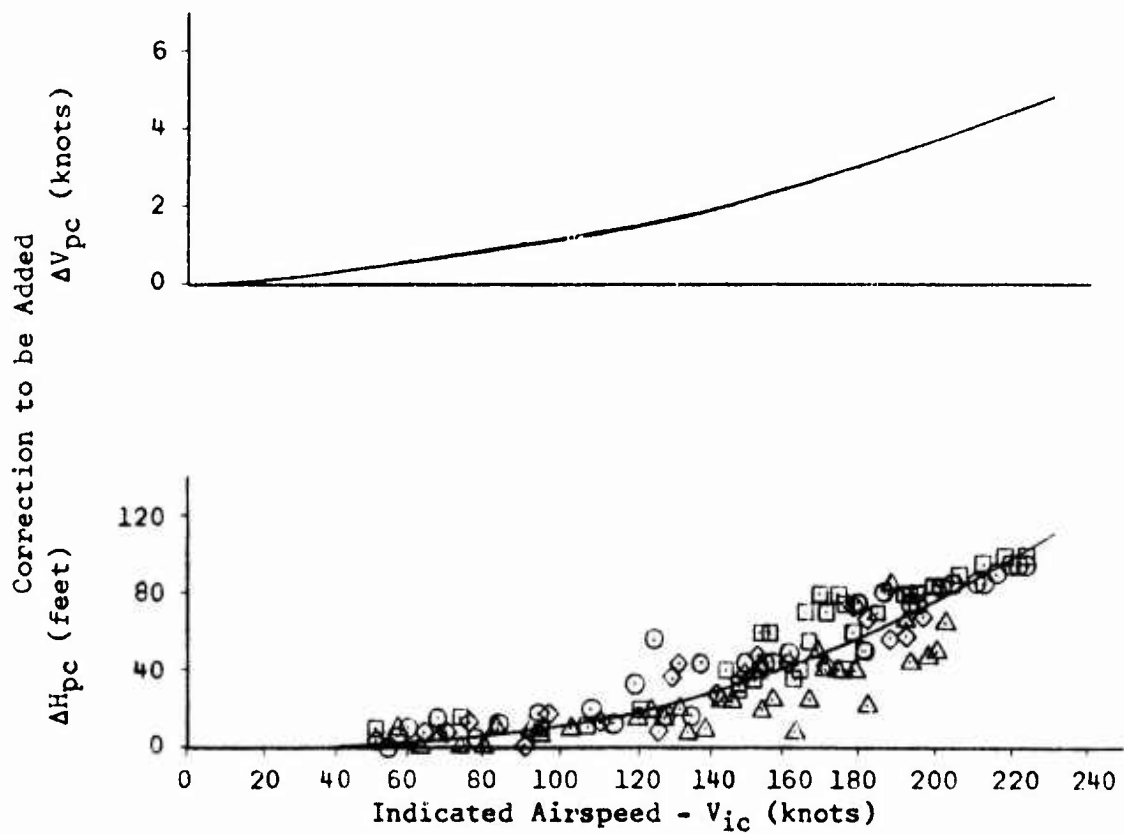


Figure 17. Airspeed Calibration During Acceleration in Ground Effect

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760

Symbol	Pressure Altitude	Method
○	35 000	T-38 Pacer
△	35 000	Smoke Trail Accel
▽	35 000	Smoke Trail Decel
◐	35 000	Smoke Trail Accel
◑	35 000	Smoke Trail Decel
□	2 300	Tower Fly-by

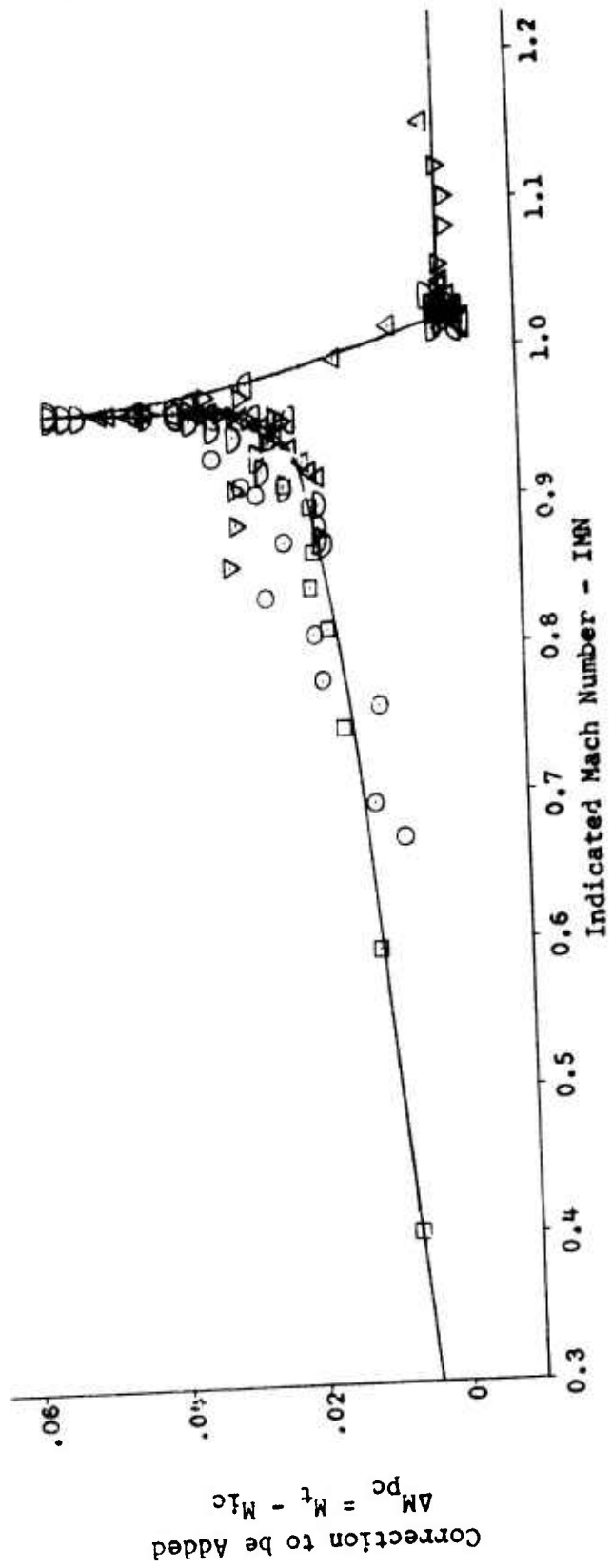


Figure 18. Airspeed Calibration

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760

Symbol	Rate of Climb (fpm)
⊙	-25 000
⊠	+25 000
⊕	-25 000
△	+25 000
⊞	-28 000

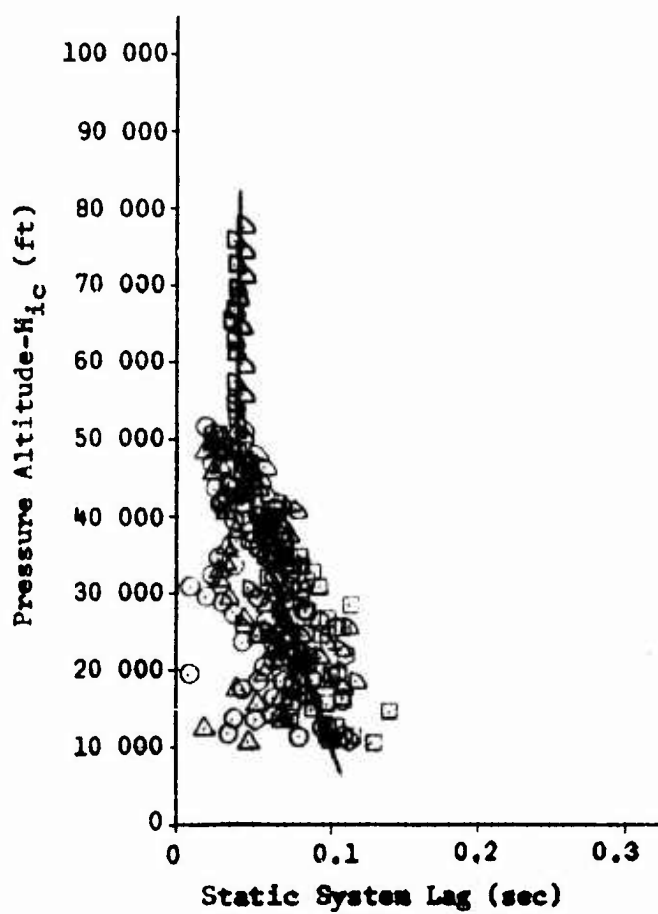


Figure 19. Static System Lag Calibration

NF-104A Aerospace Trainer
 USAF S/N 56-0760
 J79-GE-3B Engine S/N 147-387
 1 Nov 1963
 Nonafterburner Operation

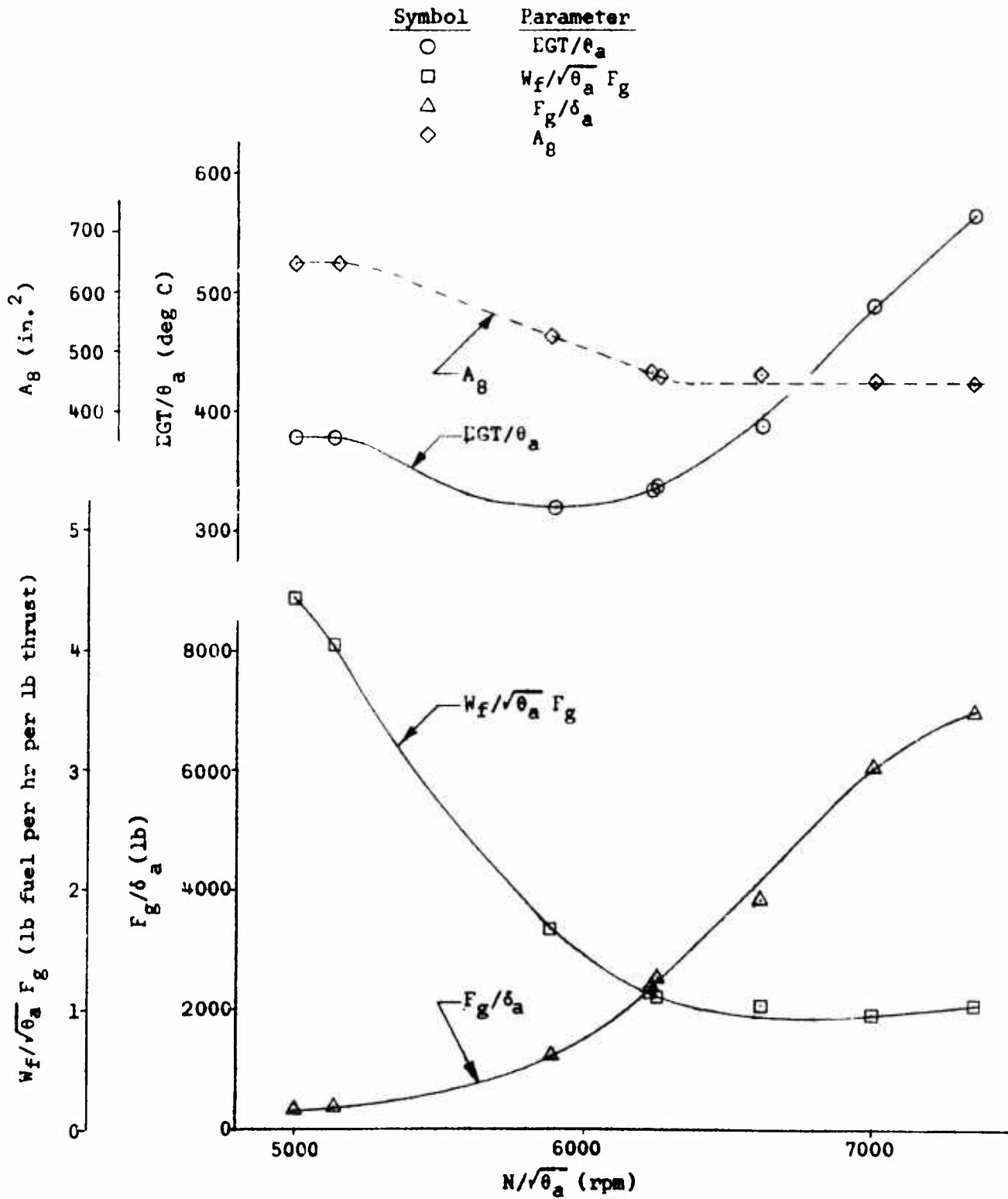


Figure 20. J79-GE-3B Installed Static Thrust Calibration

NF-104A Aerospace Trainer
 USAF S/N 56-0760
 J79-GE-3B Engine S/N 147-387
 1 Nov 1963
 Afterburning

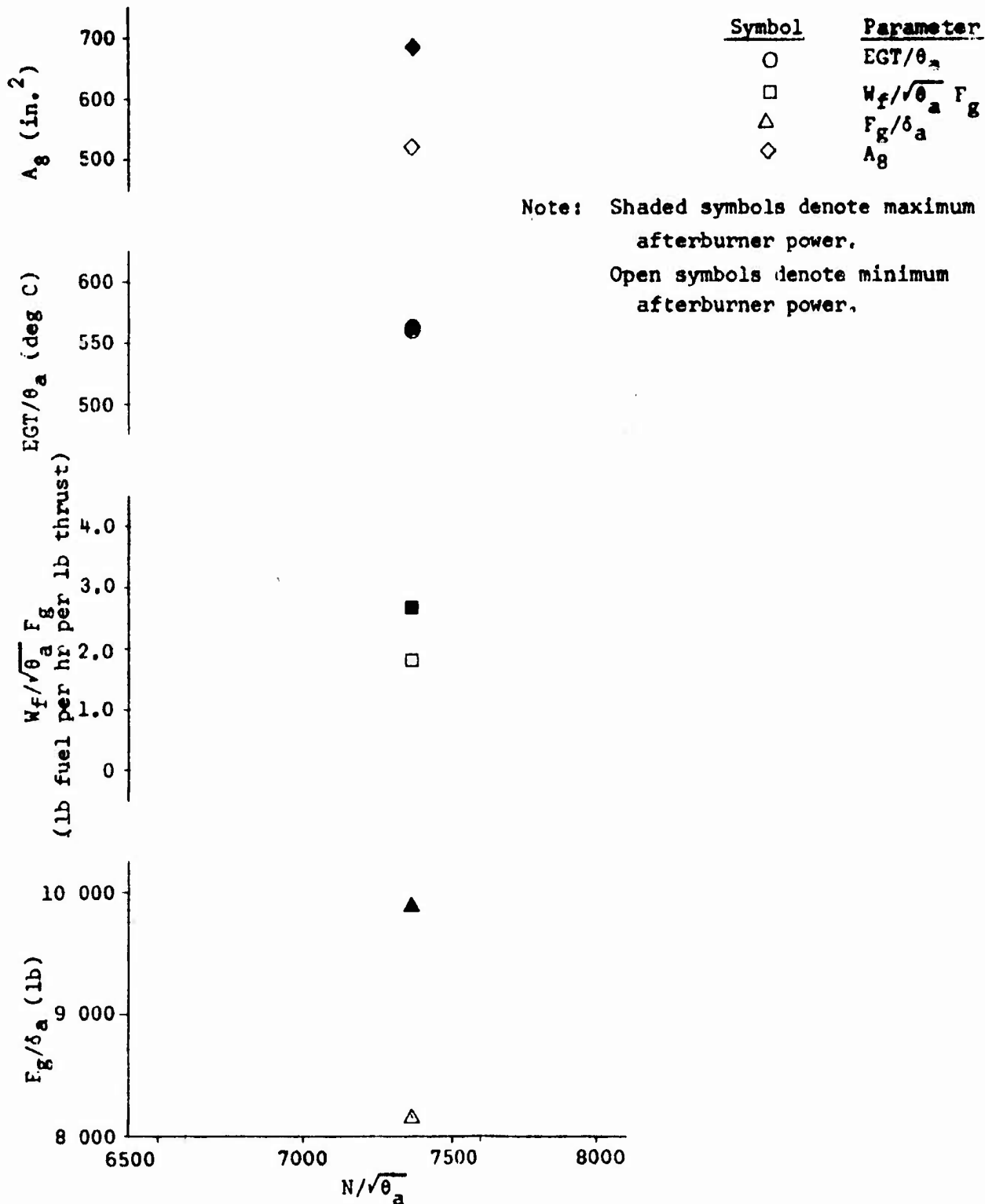


Figure 21. J79-GE-3B Installed Static Thrust Calibration

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

Altitude - Sea Level

<u>Symbol</u>	<u>Engine</u>	<u>Date</u>
○	405	1 Nov 1963
□	404	11 Mar 1964

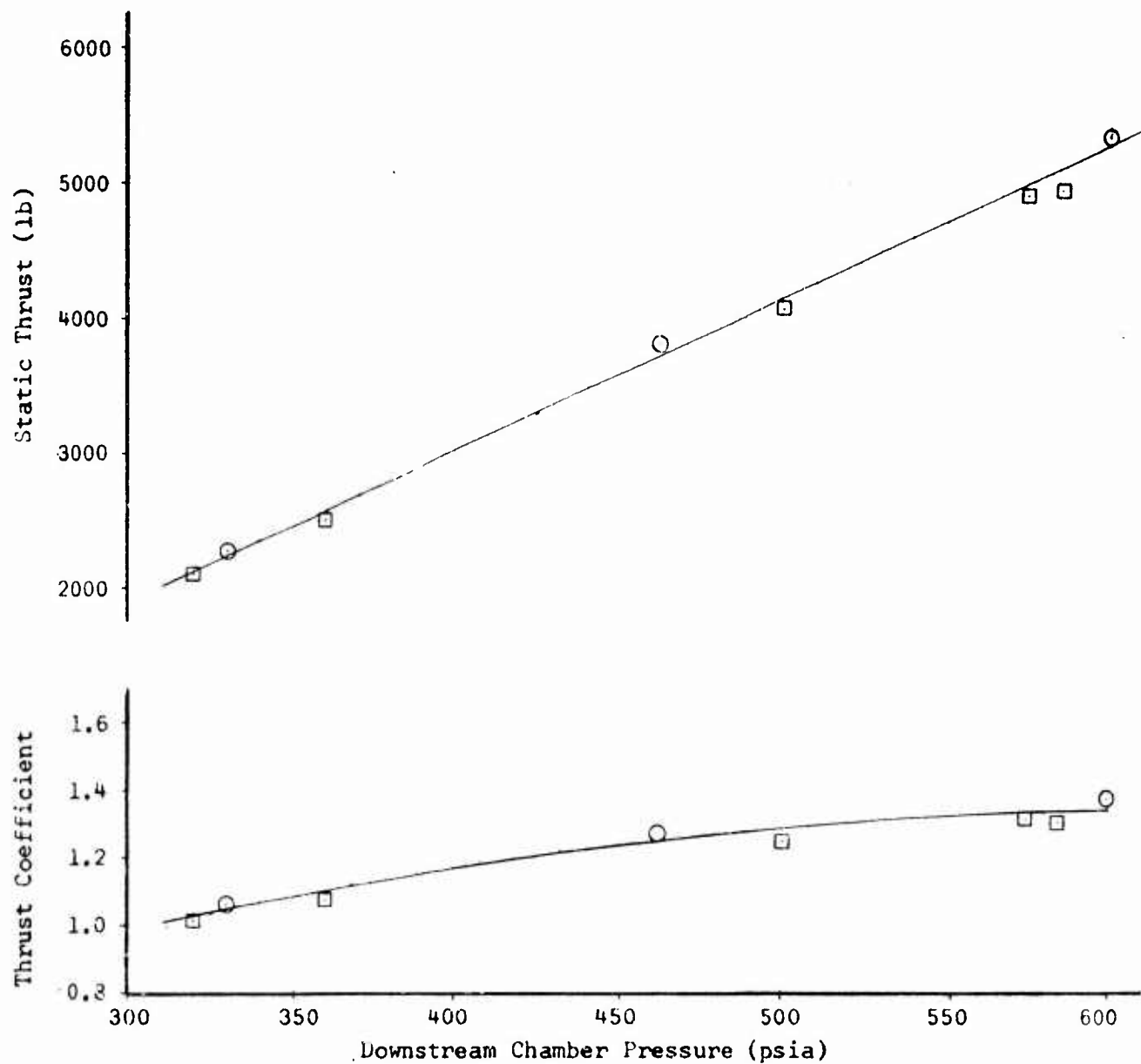


Figure 22. LR121-NA-1 Installed Static Thrust Calibration

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760

Conditions:

1. J79 engine windmilling
2. Rocket engine off
3. Aerodynamic stability augmentation on
4. Speed brakes extended
5. RCS dampers inactive
6. Average gross weight 15 500 lb
7. Average cg 17 pct MAC
8. Approximate pressure altitude 60 000 ft

Note: Darkened symbols indicate sideslip oscillation were between approximately ± 2 and ± 5 degrees. Tail denotes J79 engine on. Clear symbols indicate sideslip oscillations did not occur.

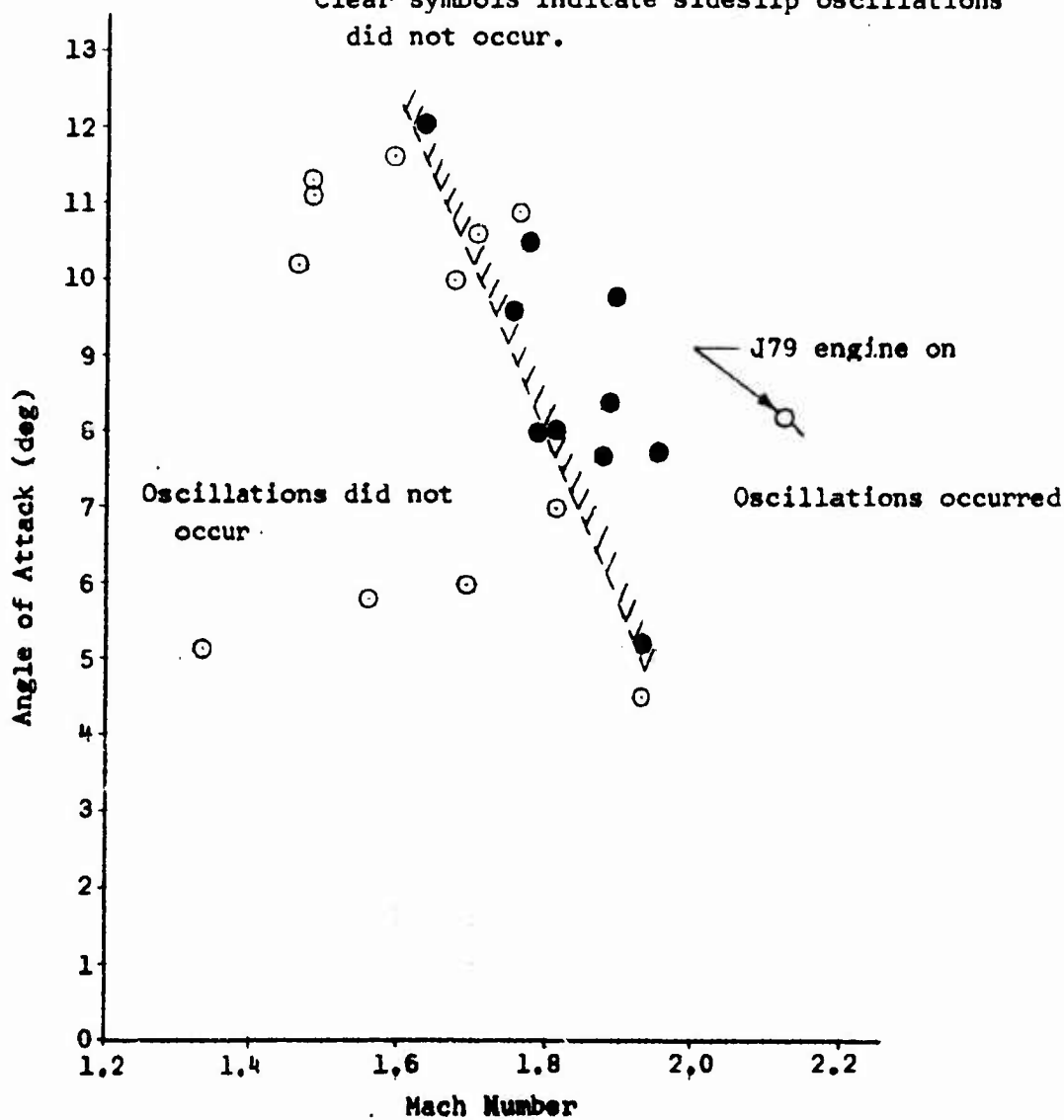


Figure 23. Lateral-Directional Oscillation Boundary

NF-104A Aerospace Trainer
USAF S/N 56-0756 and 56-0760
Stability Axes

Conditions pertaining to AFFTC data:

1. Pressure altitude 80 000 ft
2. J79 engine windmilling
3. Rocket engine off
4. Aerodynamic yaw and roll augmentation off
5. RCS dampers off
6. Average gross weight 15 800 lb
7. Average cg 15.7 pct MAC
8. Average dynamic pressure 200 psf

Symbol	Data Source
△	F-104G (Lockheed)
○	NF-104A (Lockheed)
□	NF-104A (AFFTC)
◇	NF-104A (AFFTC speed brakes extended)

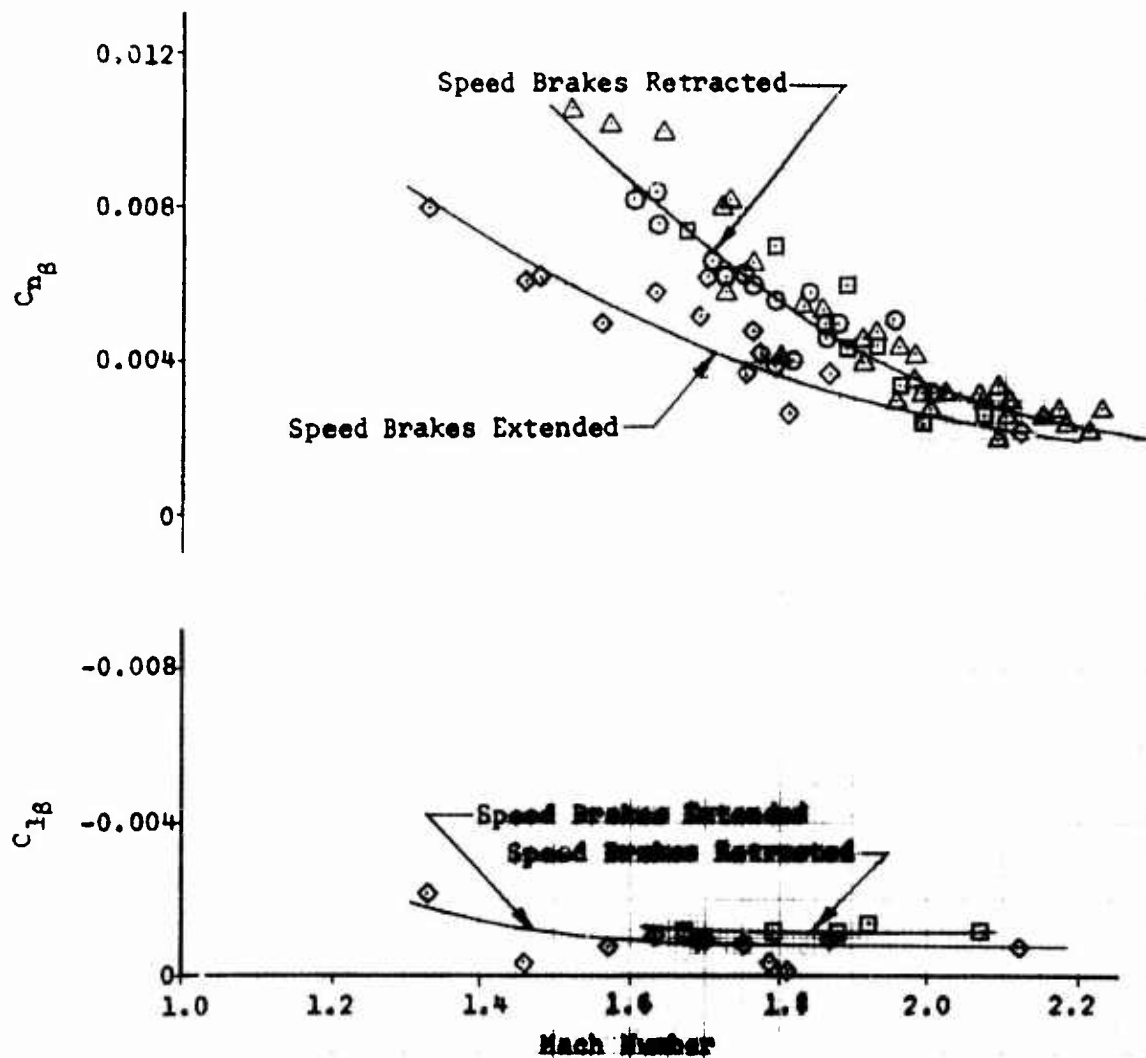


Figure 24. Lateral-Directional Stability

NF-104A Aerospace Trainer Five Degree of Freedom Simulation

Conditions

Weight	15 800 lb
Engine rpm	70 pct
Center of Gravity	18 pct MAC
Automatic Pitch Control (APC) Setting	13.8 deg Angle of Attack and 20 deg per sec Pitch Rate
RCS Kicker Setting	15 deg Angle of Attack
AARS Null Angle of Attack	11 deg
Stabilizer Trim Angle	2 deg Trailing Edge Up
Aerodynamic Dampers	On
RCS Dampers	Off

Symbol	Dynamic Pressure (psf)	Mach Number
○	10	0.9
□	20	1.2
△	40	1.5

- Note: 1. Clear symbols denote APC only.
 2. Shaded symbols denote APC and RCS kicker.
 3. Vertical crosshatched lines denote limit of aerodynamic controllability.

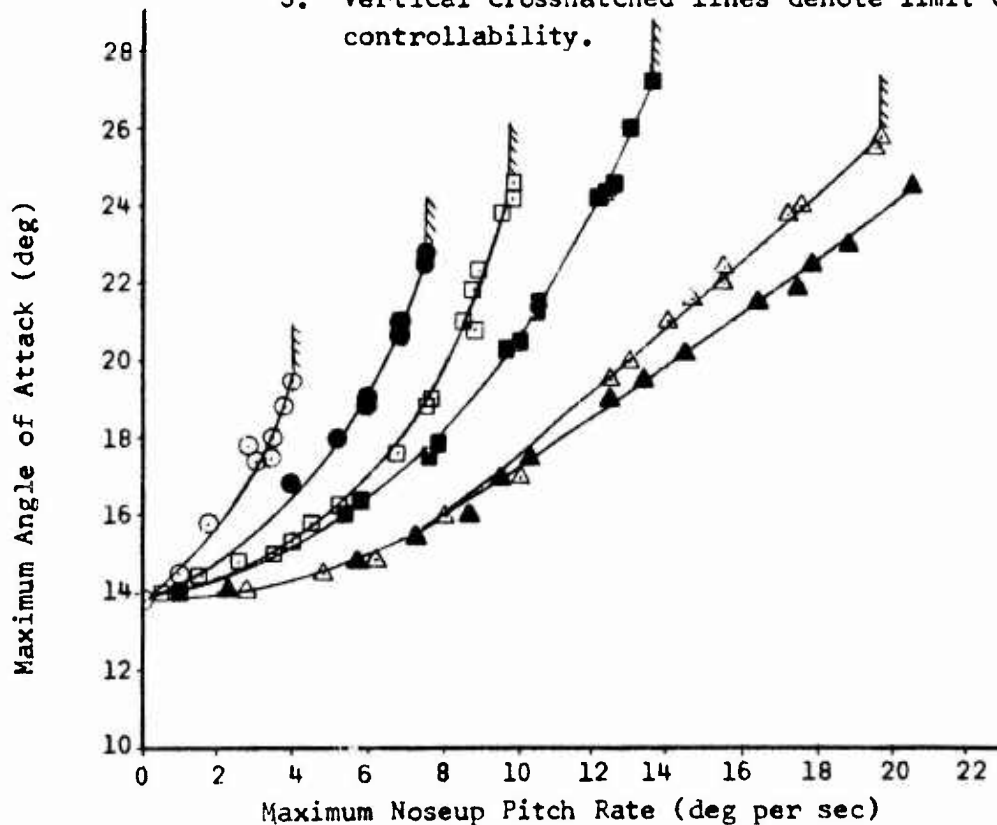


Figure 25. Handling Qualities Evaluation

NF-104A Aerospace Trainer

USAF S/N 56-0760

Flight No. 46 18 Nov 1964

Pressure Altitude = 100 600 ft Mach Number = 1.22

Airspeed = 99 KIAS Dynamic Pressure = 23 psf

Center of Gravity = 14.5 percent MAC

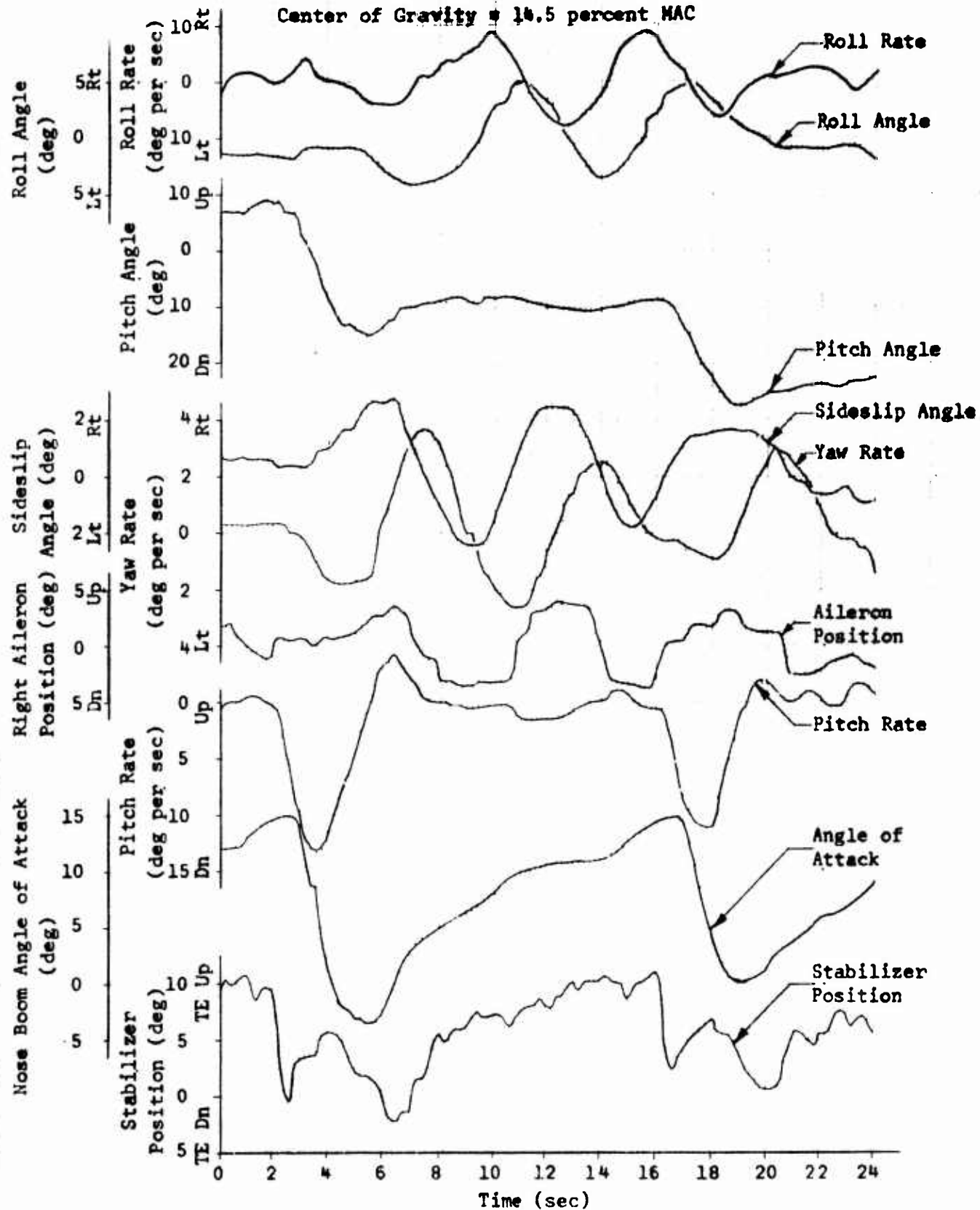


Figure 26. Determination of Minimum Dynamic Pressure

NF-104A Aerospace Trainer
USAF S/N 56-0760

Weight = 17 500 lb at Pullup
Flight No. 24 22 May 1964

Lateral-Directional Stability Flight

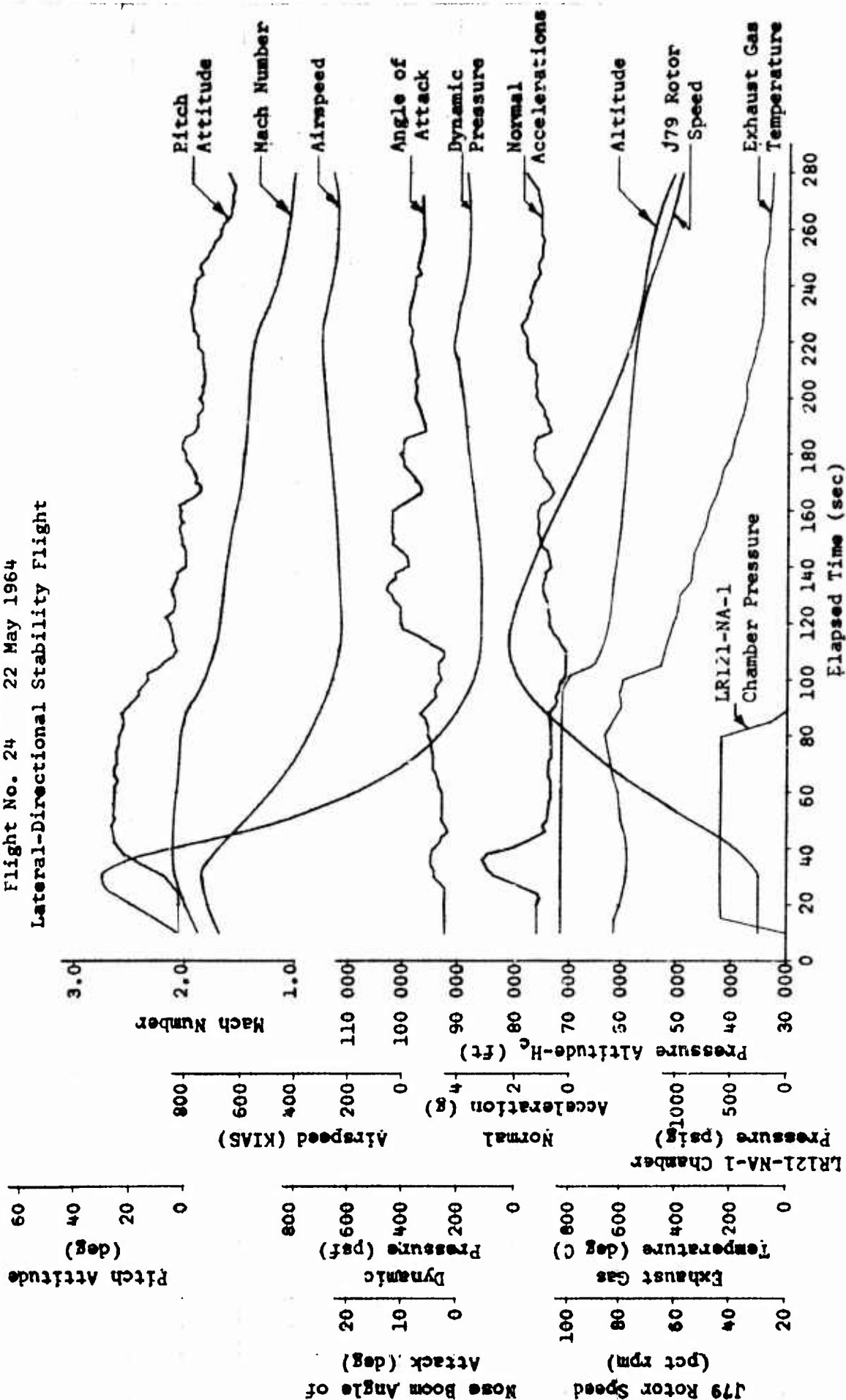


Figure 27. Time History of Zoom Climb Maneuver

NF-104A Aerospace Trainer

USAF S/N 56-0760

Weight = 17 500 lb at Pullup

Flight No. 47 19 Nov 1964

Pitch Attitude 30 Degrees

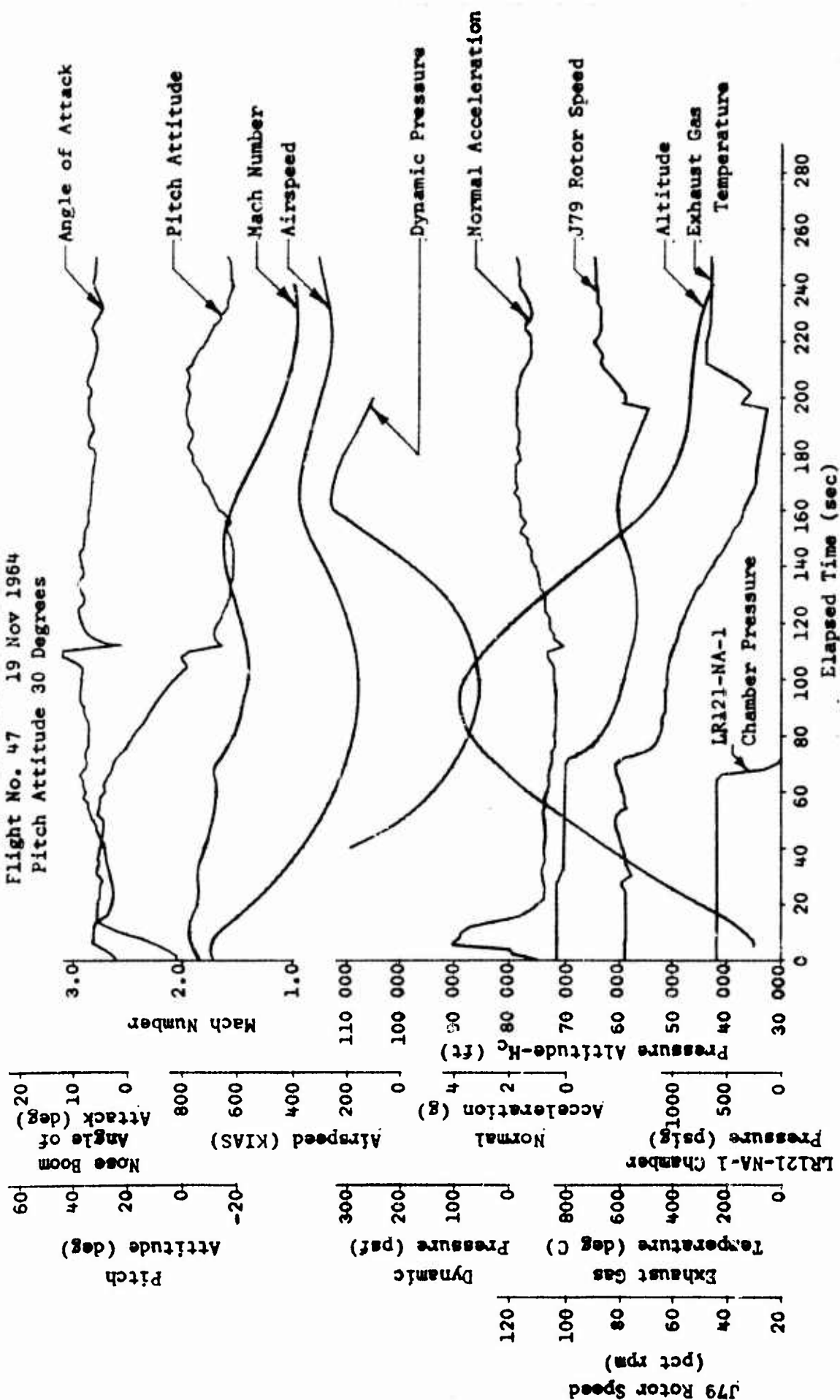


Figure 28. Time History of Zoom Climb Maneuver

NF-104A Aerospace Trainer
USAF S/N 56-0760
Weight = 17 500 lb at Pullup
Flight No. 61 16 Dec 1964
Pitch Attitude 40 Degrees

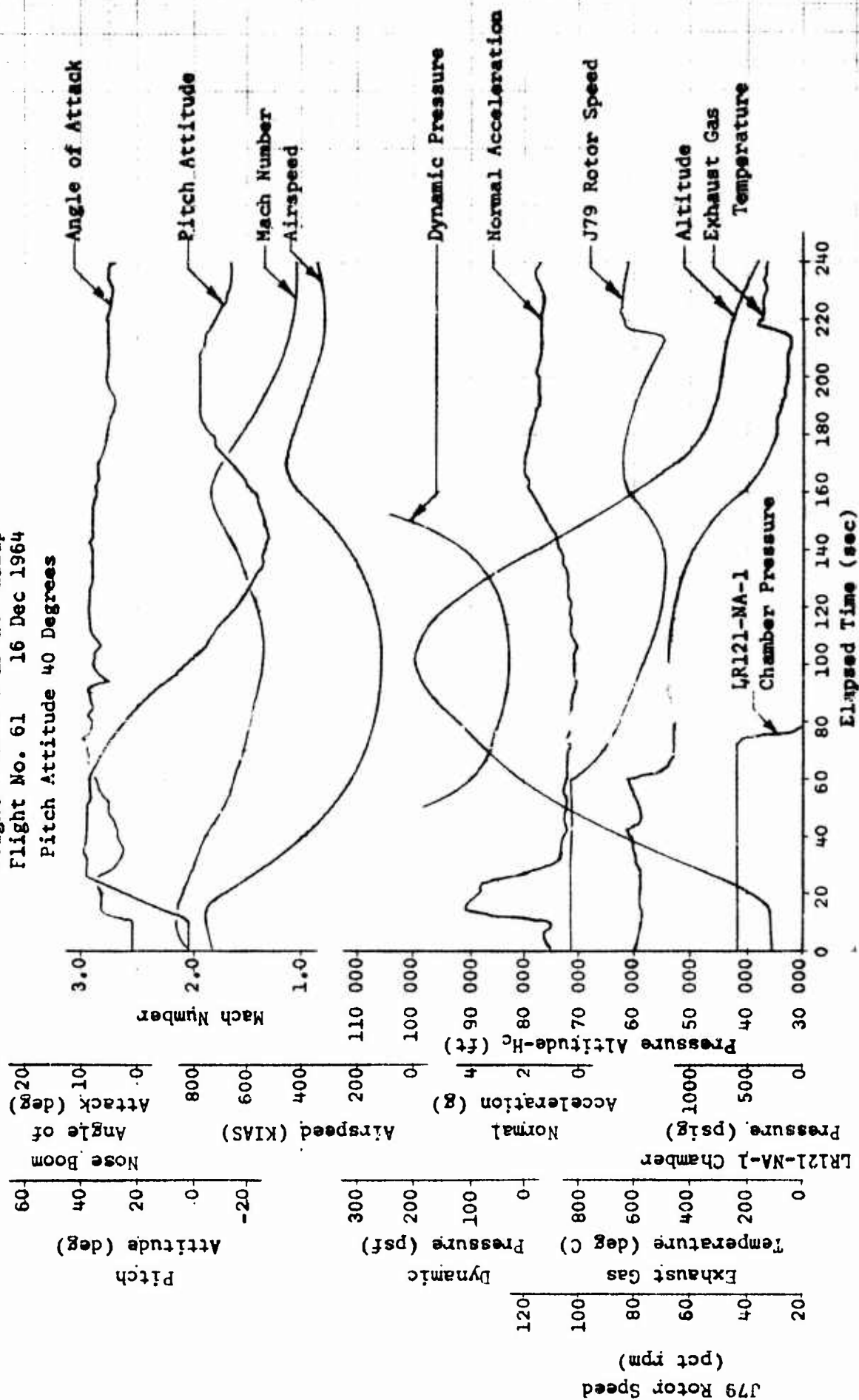


Figure 29. Time History of Zoom Climb Maneuver

NF-104A Aerospace Trainer
USAF S/N 56-0760

Weight = 17 500 lb at Pullup
Flight No. 62 16 Dec 1964

Pitch Attitude 50 Degrees

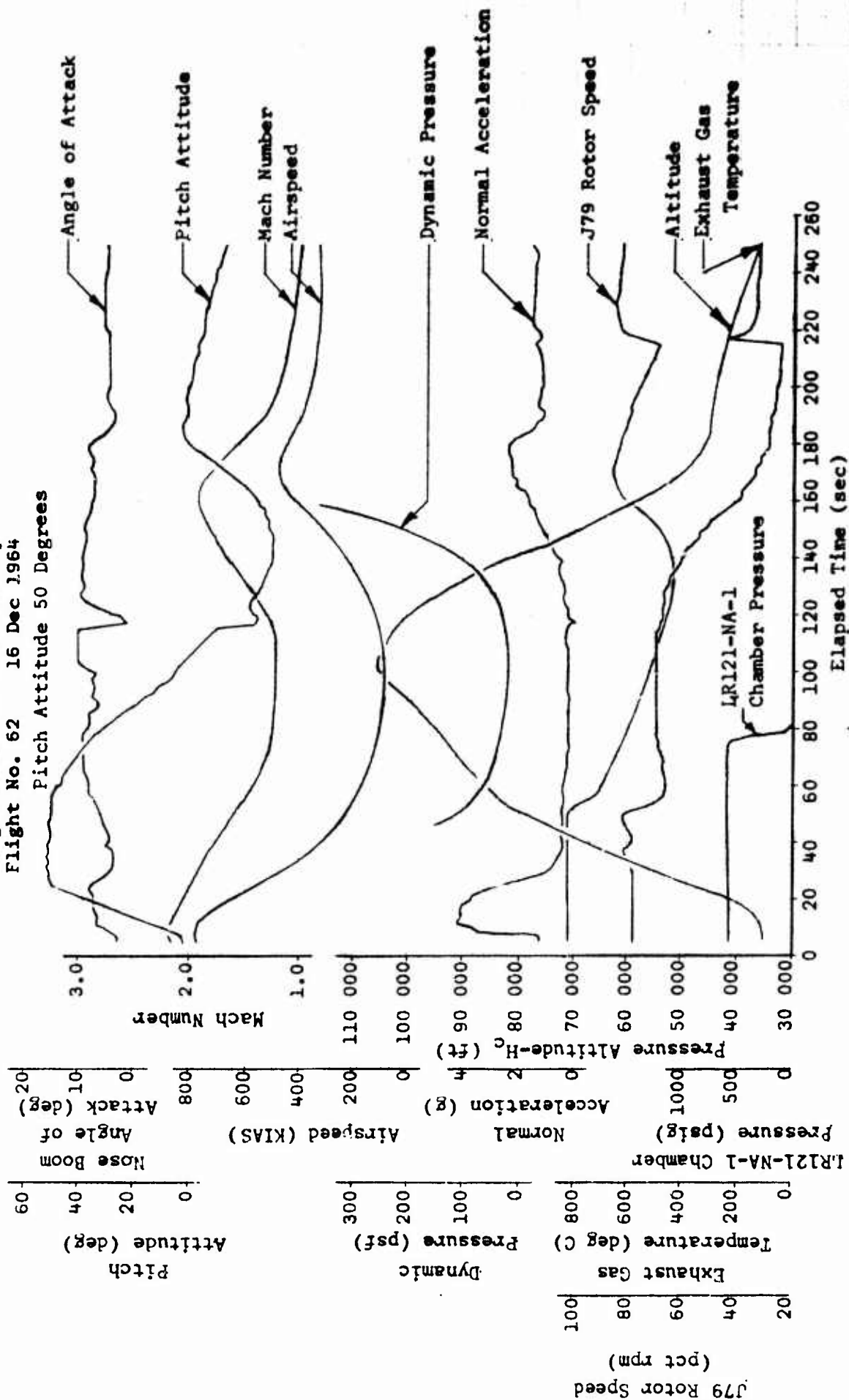


Figure 30. Time History of Zoom Climb Maneuver

**AIRPLANE DIMENSION
AND DESIGN DATA**

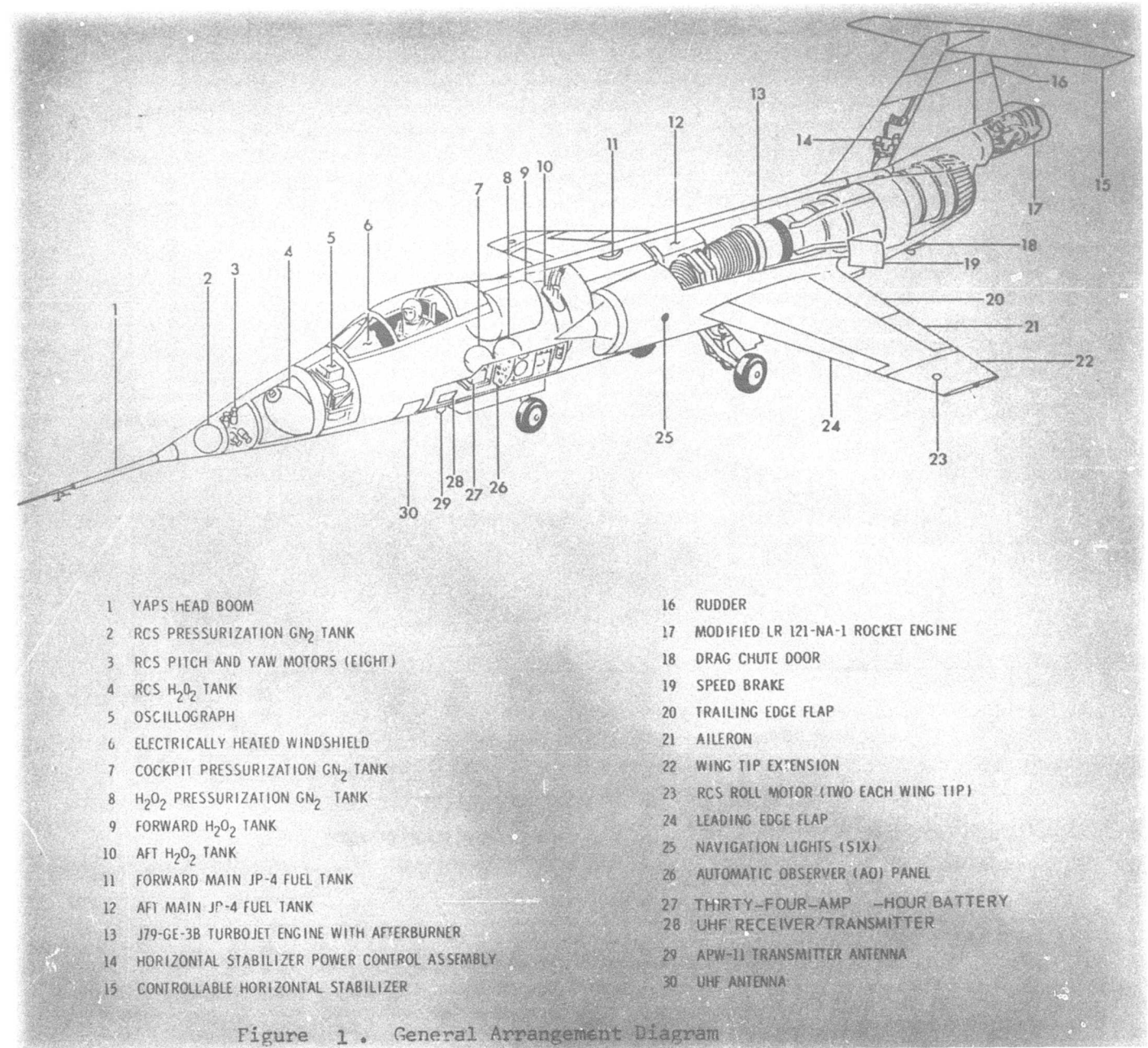


Figure 1 . General Arrangement Diagram

■ General

The general arrangement diagram shown in figure 1 and the overall dimensions shown below were taken from Lockheed-California Company Reports No. 16766-1 and 16766-2.

The overall dimensions and weights of the aircraft were as follows:

Wing area (with wingtip extensions)	212.8 ft ²
Wing span	25.83 ft
Length	62.36 ft
Height (to top of vertical stabilizer)	13.49 ft
Tread	8.77 ft
Operating weight empty	13 900 lb
JP-4 Fuel	4 960 lb
Rocket engine hydrogen peroxide	2 575 lb
RCS hydrogen peroxide	155 lb
Gaseous nitrogen and liquid oxygen	150 lb
Pilot and chute	260 lb
Engine-start gross weight	22 000 lb

■ Modifications

Conversion of the production F-104A aircraft to Aerospace Trainers included the following installations prior to the beginning of testing at the Lockheed-California Company:

1. An LR121-NA-1 rocket engine faired between the fuselage and vertical stabilizer.
2. A TF-104G empennage assembly and canopy.
3. X-15-type reaction controls. (Yaw, pitch and roll)
4. Two-foot wingtip extensions to accommodate the roll reaction control motors.

5. Modified engine air inlets.
6. Hydrogen peroxide tanks located in the armament and electronic bays replacing auxiliary JP-4 fuel tanks.
7. APW-11-S band beacon.
8. Auxiliary gaseous nitrogen cockpit pressurization.
9. Auxiliary 34-amp-hr battery.
10. All metal nose cone and boom with YAPS vanes.
11. Adequate canopy defrosting for all conditions of flight.
12. A J79-GE-3B turbojet engine.
13. Two Collins Radio of Canada 718B-8 transceivers.
14. AARS with angle of attack and sideslip indicators superimposed in front of the ball.
15. Aileron and rudder limiter cutout.
16. Radar beacon transponder (APX-46).

The following equipment and systems were removed from the production airplane to make room for the new installations:

1. M-61 gun.
2. ARC-66 UHF radio.
3. Fire control system.
4. Data link provisions.
5. VOR radio.
6. Instrument Landing System (ILS).
7. Missile auxiliaries.
8. Auxiliary JP-4 fuel tank.
9. Standard radar beacon transponder.

■ Description of the Rocket Engine and Reaction Control Systems

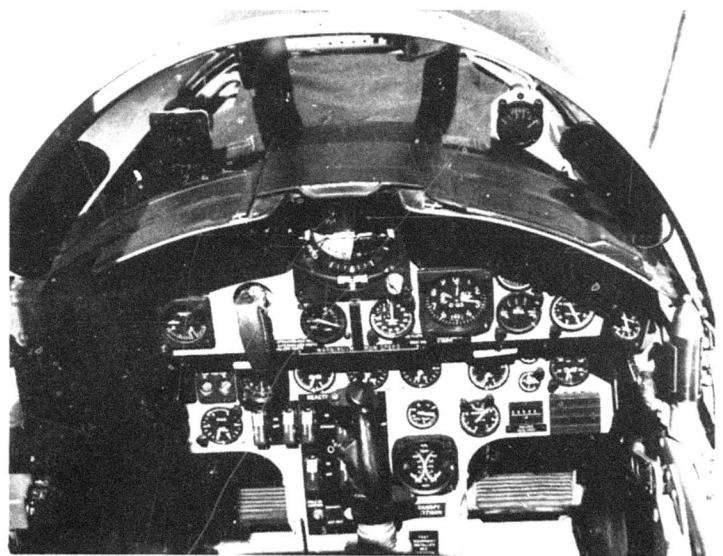
The following descriptions of the rocket engine and reaction control systems was taken from reference 1.

The LR121-NA-1 rocket engine was a variable-thrust, multistart, bipropellant engine rated at 6000 pounds static thrust at a design altitude of 35 000 feet. The variable thrust range was any value between 50 and 100 percent thrust. As long as fuel and oxidizer were available, the engine could be started at any altitude, then shut-down and restarted if additional thrust was required. The propellants consisted of JP-4 fuel and a 90 percent concentration by weight of hydrogen peroxide. These propellants were automatically supplied to the engine in a ratio of approximately 6 pounds of oxidizer to 1 pound of fuel. Oxidizer flow rate at 100 percent thrust was approximately 23 pounds (2 gallons) per second. A turbopump forced the oxidizer and JP-4 fuel into a thrust chamber where the hydrogen peroxide contacted a silver catalyst in the screen pack and rapidly decomposed into free oxygen and superheated steam. This chemical reaction produced a temperature of approximately 740 degrees C which ignited the JP-4 fuel spontaneously causing it to burn intensely in the oxygen-rich atmosphere of the combustion chamber.

The principal parts of the rocket engine were the thrust chamber, the turbopump, and the gas generator system which drove the turbopump. Two pressure-operated valves controlled propellant flow to the thrust chamber. One electrically operated valve and one servo-operated valve controlled gas generator hydrogen peroxide flow. One electrically operated valve controlled rocket engine purging.

The RCS incorporated hydrogen peroxide reaction motors for control in each of the three axes. Eight pitch and yaw motors (113-pound thrust each) four in each axis with two in each direction were installed in the all metal nose section. Four roll motors (43-pound thrust each) were installed in the wingtip extensions. These motors were oppositely directed, one up and one down, in each wingtip. A solenoid valve controlled hydrogen peroxide flow to each reaction motor. The 13.4 gallon RCS hydrogen peroxide tank and its 12-inch diameter gaseous nitrogen pressurizing tank, together with the associated valves and plumbing were located in the nose section. The RCS could be operated manually using the control handle and automatically with the damper system.

The reaction control damper system which consisted of one motor in each direction, in each axes, could be overridden manually and was a self-contained unit mounted in the aft gun bay of the aircraft. This unit contained a 20 degree per second rate gyro in each of the three axes to provide signals through amplifier-demodulation circuits which controlled a Schmidt trigger in each axis. The threshold of the Schmidt triggers could be adjusted from 0.2 degrees per second to 2 degrees per second in pitch and yaw, and 0.5 degrees per second to 5 degrees per second in roll. The settings used during the evaluation were 2.0 degrees per second in pitch, 0.2 degrees per second in yaw, and 0.5 degrees per second in roll.



PILOT'S PANEL

■ Test Instrumentation

The instrumentation provided was designed as a permanent system for the aircraft. The installation consisted of two onboard recorders, an automatic observer (AO) photopanel and a 50 channel CEC oscillograph. Both recorders were controlled from the instrument control panel located on the left canopy sill in the cockpit.

One amber light on the panel indicated camera speed to the pilot by flashing once per camera frame. Another amber light indicated oscillograph malfunction.

The instrumentation master switch actuated the AO camera at 1-frame per second and energized the instrumentation gyros, oscillograph, gross angle of attack instrument, hydrogen peroxide used counters and a portion of the RCS "kicker" circuit. When the record switch was placed in the up position it operated the oscillograph at approximately one-inch per second, uncaged the instrumentation gyros, and increased the camera speed from 1-frame per second to 2 frames per second.

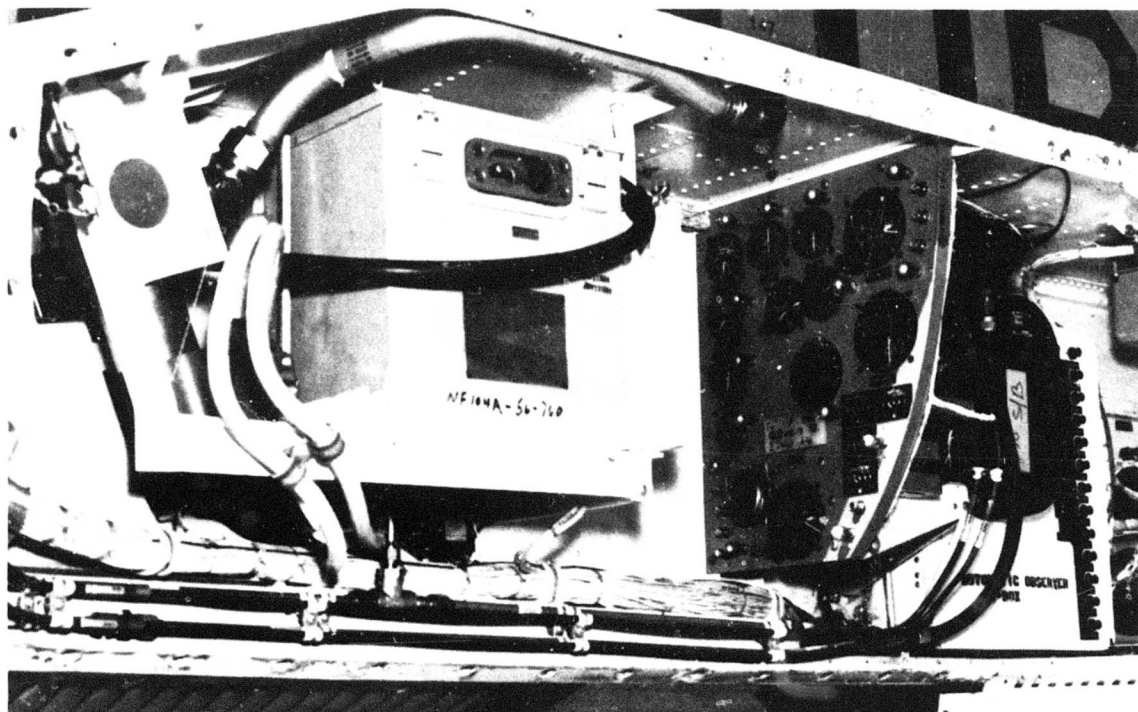
An "event" button on the instrumentation control panel enabled the pilot to identify certain events during the flight. Pushing the button transmitted a UHF side-tone and illuminated an "event" light on the AO panel. The "event" light

marked the film and caused a pilot-signal trace deflection on the oscillograph record. In aircraft S/N 56-0760 the control stick trigger also controlled the UHF side-tone and the stick grip bomb button increased the speed of the oscillograph to 10 inches per second for a better display of fuel flow.

Three 5-digit rotating drum counters, one on the instrument panel, one on the AO panel, and one on the oscillograph, were used for correlation of events.

● automatic observer

The AO instrument photopanel and camera were installed in the original gun-bay area to provide a film record of pertinent instrument readings. The camera had a film capacity of 400 feet, which was sufficient to record an entire zoom flight at 2 frames per second. The AO panel contained the following instruments:



**AUTOMATIC OBSERVER INSTRUMENTATION
AND 34 AMP-HR BATTERY INSTALLATION**

Rocket chamber pressure indicator

Rocket engine hydrogen peroxide used counter

Correlation counter

J79 engine JP-4 used counter

J79 afterburner JP-4 used counter

(JP-4 used by the rocket engine was not recorded)

J79 engine EGT indicator

J79 nozzle position indicator

J79 tachometer

Clock

Altimeter (100 000 ft Kollsman)
(Aircraft S/N 56-0760 only)

Altimeter (150 000 ft Bendix;
31215-9M12-A1)

Airspeed indicator

Free air temperature indicator

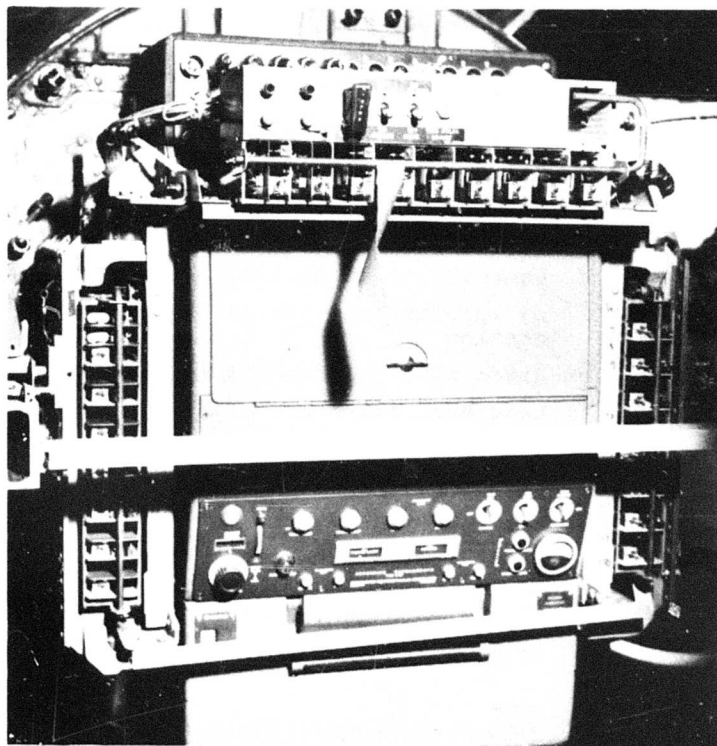
Numbered indicator lights

Voltmeter (dc)

● oscillograph

The oscillograph, located in the nose section, provided a system for recording parameters for which it was desirable to have a continuous record of events. Following is a list of parameters measured on the oscillograph of aircraft S/N 56-0760:

Parameter	Sensitivity (units per deflection)	Range of Instrument
Timing lines	10 per second	- - - -
Oscillograph run number	1 per second	- - - -
Correlation count	1 per second	- - - -
Event mark	- - - -	- - - -
Reference	- - - -	- - - -
J79 engine fuel flow	- - - -	- - - -
J79 afterburner fuel flow	- - - -	- - - -
J79 engine fuel temperature	9.0 degrees C per inch	0 to 50 degrees C
J79 afterburner fuel temperature	17.2 degrees C per inch	0 to 100 degrees C



OSCILLOGRAPH INSTALLATION

<u>Parameter</u>	<u>Sensitivity (units per deflection)</u>	<u>Range of Instrument</u>
Rocket engine turbine wheel hydrogen peroxide pressure	100 psi per inch	0 to 500 psi
Rocket engine turbine pump hydrogen peroxide inlet pressure	24.5 psi per inch	0 to 100 psi
Rocket engine turbine pump hydrogen peroxide discharge pressure	203.5 psi per inch	0 to 1000 psi
Rocket engine turbine pump JP-4 inlet pressure	18.0 psi per inch	0 to 50 psi
Rocket engine turbine pump JP-4 discharge pressure	212.5 psi per inch	0 to 1000 psi
Rocket engine throttle position	- - - -	- - - -
Rocket engine exit ambient pressure	0.58 psi per inch	0 to 5 psi
Pitchup RCS actuation	- - - -	- - - -
Pitch down RCS actuation	- - - -	- - - -
Right yaw RCS actuation	- - - -	- - - -
Left yaw RCS actuation	- - - -	- - - -
Right roll RCS actuation	- - - -	- - - -
Left roll RCS actuation	- - - -	- - - -
Pitch angle	43.5 degrees per inch	+ 86 degrees
Roll angle	18 degrees per inch	+ 177 degrees
Angle of sideslip	3.3 degrees per inch	+ 10 degrees
Angle of attack	5.85 degrees per inch	+ 25 to -5 degrees
Pitch rate	16.55 degrees per second per inch	+ 45 degrees per second
Yaw rate	9.2 degrees per second per inch	+ 20 degrees per second
Roll rate	75.25 degrees per second per inch	+ 300 degrees per second
cg vertical acceleration	1.6 g's per inch	+ 10 g's
cg longitudinal accel- eration	0.53 g per inch	+ 2 g's
Right-hand aileron position	8.8 degrees per inch	+ 22 to -17 degrees
Left-hand aileron position	8.95 degrees per inch	+ 22 to -17 degrees
Stabilizer position	5.2 degrees per inch	+ 15 to -5 degrees
Rudder position	10.7 degrees per inch	+ 20 degrees
Aileron stick force	14.35 pounds per inch	+ 50 pounds
Stabilizer stick force	30 pounds per inch	100 pounds pull to 50 pounds push
Rudder pedal force	76 pounds per inch	+ 200 pounds
Pilot signal	- - - -	- - - -

Note: RCS damper trace deflections were one-half the amplitude of RCS manual trace deflections.

■ Weight and Balance

The gross weight, center of gravity, and center of gravity variations of the NF-104A aircraft, with and without RCS hydrogen peroxide, were determined prior to the contractor's first flight (figures 2 and 3). The horizontal increments, shown by the darkened symbols on these two figures, indicate the cg change for each 85 gallons (550 lb) of JP-4 fuel and the vertical increments, shown by the clear symbols, indicate the cg change for each 55

gallons (650 lb) of hydrogen peroxide. The aircraft were weighed before and after several flights at the AFFTC to check the fuel used instrumentation and to verify the center of gravity travel. At the fully loaded takeoff gross weight of 22 000 pounds the cg was -4.1 percent MAC (gear down) and the load on the nose gear was about 5000 pounds. After return from a zoom climb maneuver at a gross weight of 15 500 pounds, the cg was approximately 17 percent MAC.

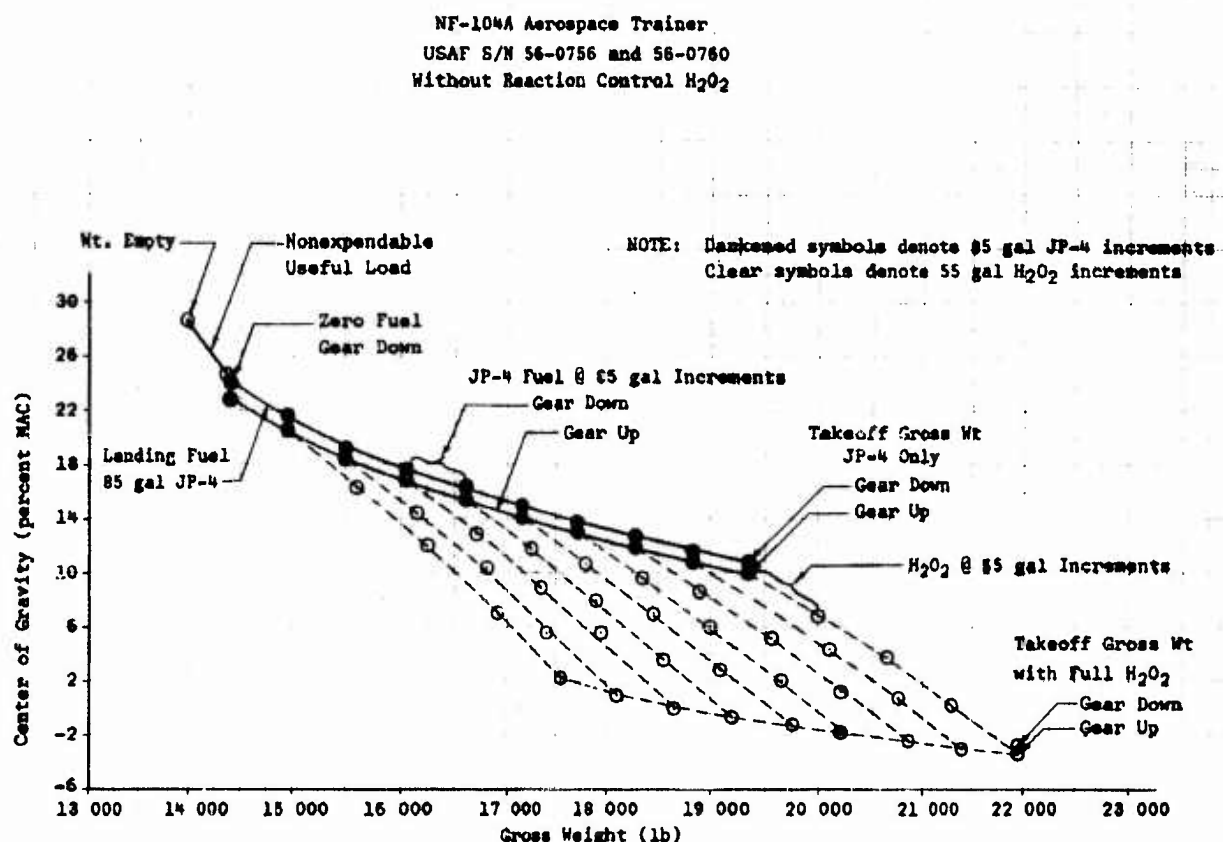


Figure 2 . Center of Gravity Travel

NF-104A Aerospace Trainer
 USAF S/N 56-0756 and 56-0760
 With Reaction Control H_2O_2

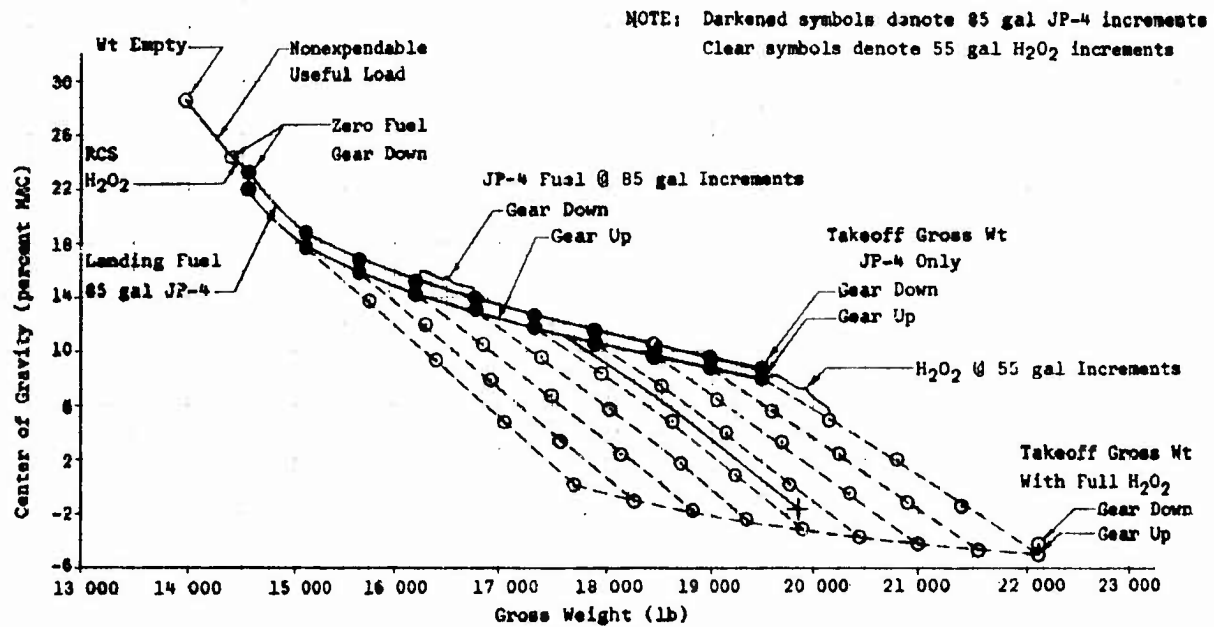


Figure 3 . Center of Gravity Travel

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* References marked with an asterisk were reproduced in the Appendix III.

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13 ABSTRACT		
The NF-104A Aerospace Trainers were designed to provide students of the Aerospace Research Pilot School with experience in rocket engine operation, reaction control operation, and precise zoom climb profile flying. This report presents the results of the flight test evaluation which was conducted to determine whether the aircraft had acceptable flying qualities for this design mission. It was necessary to incorporate many development items into the aircraft and impose flight restrictions in order to attain a reasonable degree of safety. The aircraft were restricted to a maximum pullup Mach number of 2.15, and a maximum pitch angle of 50 degrees. At the apex of a zoom maneuver the aircraft were restricted to a minimum dynamic pressure of 20 pounds per square foot and a minimum Mach number of 1.0. The limitations at the apex were placed on the aircraft after it was determined that the J79 engine gyroscopic effects coupled with rapid rate of change of yaw or pitch angles could cause loss of control at very low dynamic pressures. Observing these limitations the aircraft can attain altitudes in excess of 100 000 feet and provide approximately 90 seconds of zero g flight and reaction control experience. The results of the test program showed that successful utilization of these aircraft as Aerospace Trainers requires: (1) strict observance of limitations imposed on the aircraft; (2) intimate familiarization of flight characteristics of the aircraft, and mission profile; and (3) meticulous maintenance of the aircraft and its systems.		

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