

AFFTC-TR-53-33



PHASE II PERFORMANCE AND STABILITY TESTS OF THE YF-100A AIRPLANE



ALFRED D. PHILLIPS
Project Engineer

FRANK K. EVEREST
Lieutenant Colonel, USAF
Project Pilot

December 1953

DISTRIBUTION A. Approved for public release; distribution is unlimited.
412TW-PA-15650

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE MATERIEL COMMAND
UNITED STATES AIR FORCE

A
F
F
T
C

15,116
15-116



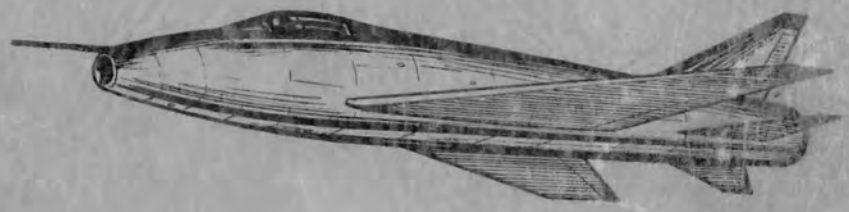
~~SECRET~~
AUTH: ~~Gen. Anderson~~ AFFTC
INITIALS: Wm
DATE: 20 Jan 1954
COPY: 82 OF 109

AD 24334

AIR FORCE FLIGHT TEST CENTER

AIR RESEARCH & DEVELOPMENT COMMAND

AFTC TECH LIB EAFB, CA
5 0605 01151265 9



AF TECHNICAL REPORT
NO. AFFTC 53-33

CLASSIFICATION CANCELLED

PHASE II PERFORMANCE AND
STABILITY TESTS OF THE
YF-100A AIRPLANE

(OR CHANGE TO)

TR RDS 4 MP-6-17-E, 4 pages
AUTHORITY: AMC REPORT

DATE: 15 Sep 58

CLASSIFICATION CANCELLED

APPROVED: AMC JAROC Sec Class
Grade 1 Feb 58
DATE: 20 Oct 58

ALFRED D. PHILLIPS
PROJECT ENGINEER

BY: By Stoliker FRANK K. EVEREST, LT. COL., USAF
PROJECT PILOT

DECEMBER 1953

EDWARDS AIR FORCE BASE
CALIFORNIA

US AIR FORCE
TECHNICAL LIBRARY
AIR FORCE FLIGHT TEST CENTER
EDWARDS AFB CALIFORNIA

53-1959



Cy # 79-53-1959

~~SECRET~~

ADDITIONAL COPIES OF
THIS REPORT
MAY BE OBTAINED FROM THE
ARMED SERVICES TECH. INFO. AGENCY,
DOCUMENT SERVICE CENTER
U.B. BLDG., DAYTON 2, OHIO

~~"This document contains information affecting the National defense of the United States within the meaning of the Espionage Laws, Title 18, U.S.C., Sections 793 and 794. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law."~~

RETAIN OR DESTROY IN ACCORDANCE
WITH AFR 205-1. DO NOT RETURN.

~~SECRET~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Wright Air Development Center

AIR RESEARCH AND DEVELOPMENT COMMAND

WRIGHT-PATTERSON AIR FORCE BASE,
OHIO

IN REPLY ADDRESS BOTH COMMUNICATIONS
AND ENVELOPE TO COMMANDER, WRIGHT
AIR DEVELOPMENT CENTER, ATTENTION
FOLLOWING OFFICE SYMBOL:

WCOS

20 APR 1955

SUBJECT: (U) Phase II Performance and Stability Tests of the YF-100A Aircraft

**TO: Commander
Air Force Flight Test Center
Edwards Air Force Base, California**

1. Reference is made to WADC Letter dated 17 June 1954, Subj: Phase II Performance and Stability Test of the YF-100A AFFTC Technical Report No. 53-33.

2. The attached comments regarding recommendations contained in the referenced report are submitted for your consideration. The comments reflect the latest information available on the status and progress of corrective action to eliminate the reported deficiencies.

3. Paragraph references noted in the comments are the original headings contained in AFFTC Report No. 53-33, and are listed for convenience in referring to the report.

FOR THE COMMANDER

Frank W. Williams

1 Incl:
Reply to Phase Rpt
AFFTC-53-33

FRANK W. WILLIAMS
Lt. Colonel, USAF
Chief, Operational Services Division
DCS/Operations
~~CLASSIFICATION CANCELLED~~

IF INCLOSURES ARE WITHDRAWN (OR NOT ATTACHED) THE CLASSIFICATION OF THIS CORRESPONDENCE WILL BE CHANGED OR CANCELLED IN ACCORDANCE WITH PAR 25E, AFR-295-1.

(OR CHANGED TO _____)

~~CLASSIFICATION CANCELLED~~

~~(OR CHANGED TO _____)
TH RDS MPA-6-17-E, 4 Jun 58
AUTHORITY: AMC REPORT NO. CD-356~~

AUTHORITY: AMC/ARDC Sec. Cls. Div. 2
as new 1 Oct 58
DATE 13 Jan 59

~~DATE 15 Sep 58~~

~~CONFIDENTIAL~~

55WCS-8914-A

BY: _____

~~CONFIDENTIAL~~

55-2111

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

20 APR 1955

WCOS

SUBJECT: (U) Phase II Performance and Stability Tests of the YF-100A Aircraft

**TO: Commander
Air Force Flight Test Center
Edwards Air Force Base, California**

1. Reference is made to WADC Letter dated 17 June 1954, Subj: Phase II Performance and Stability Test of the YF-100A AFFTC Technical Report No. 53-33.

2. The attached comments regarding recommendations contained in the referenced report are submitted for your consideration. The comments reflect the latest information available on the status and progress of corrective action to eliminate the reported deficiencies.

3. Paragraph references noted in the comments are the original headings contained in AFFTC Report No. 53-33, and are listed for convenience in referring to the report.

FOR THE COMMANDER

1 Incl:
Reply to Phase Rpt
AFFTC-53-33

FRANK W. WILLIAMS
Lt. Colonel, USAF
Chief, Operational Services Division
DCS/Operations

IF INCLOSURES ARE WITHDRAWN (OR NOT ATTACHED) THE CLASSIFICATION OF THIS CORRESPONDENCE WILL BE CHANGED OR CANCELLED IN ACCORDANCE WITH PAR 25E, AFR-205-1.

33779
144-

~~CONFIDENTIAL~~

55WCS-8914-A

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Reply to Phase Report

~~CONFIDENTIAL~~

AF Technical Report No. AFFTC-53-33

Par E. 1b - A mechanical stability device, referred to as a "Mach sensing stick force controller", was developed in conjunction with the original linear stabilizer control system to improve static longitudinal stability. The change from linear to non-linear stabilizer control system; however, has necessitated a redesign of this Mach box. The prototype of the redesigned Mach box will be made available to AFFTC on completion of present NAA tests.

Par E. 1c - A substantial improvement in low speed handling characteristics has been realized by the addition of the non-linear gearing system. Both the forces and stick motions necessary to control the airplane have been reduced for stabilizer trim positions used for landings. Increased aileron stick travel reduced sensitivity of lateral control and provided substantial improvement in low speed handling. Enlargement of the vertical tail, as a result of ECP F-100A-375, also contributed appreciably to the improvement of low speed handling characteristics.

Par E. 2a - This deficiency will be substantially reduced by the installation of the yaw damper on all F-100 aircraft. The production effectivity of this installation is F-100C, No. 164, with provisions made for retrofit of all inservice F-100 aircraft.

A prototype pitch damper is currently being flight tested by NAA and will be made available to AFFTC for testing as soon as the Contractor's tests are completed.

Par E. 2f - Pratt and Whitney has not provided a suitable configuration but this matter is still under study and it is anticipated that a final decision will be made in the next few weeks.

Par E. 2g - The stabilizer trim rate has been increased approximately 50%. This change in trim rate was achieved by increasing the speed of the trim motor. As a result of ECP F-100A-375, an impulse trim system has been installed in all F-100 aircraft which reduces the starting lag time thus producing an apparent increase in the trim rate.

Par E. 2h - There have been some minor changes initiated in the cockpit which have reduced the magnitude of this deficiency. Cognizance of this problem is maintained by the Laboratories and further improvement is anticipated with additional study.

Par E. 2k - A study has been made of an anti-skid system installation for the F-100A, F-100C, and F-100D airplanes. Approval has been granted for installation on F-100C, No. 164 and subsequent, and all F-100D airplanes. Retrofit of F-100A and C aircraft prior to No. 164 was disapproved by SMAMA, but is being further investigated by WADC.

~~CONFIDENTIAL~~
~~CONFIDENTIAL~~

Par E. 2l - There has been considerable improvement in the selection of drag chute configuration and mechanisms which have resulted in a high degree of reliability. The increase in reliability was accompanied by a slight decrease in actuating time; however, it is not anticipated that there will be any further reduction of time with the use of present equipment, which is considered to be the best available.

Par e. 2n - A substantial improvement in control response has been achieved by the modification required through implimentation of ECP F-100A-375. This modification included a redesign of the vertical tail and wing tip extensions and the installation of a non-linear gearing longitudinal control system. The merits and effects of this modification have been discussed elsewhere in this report and in WADC comments on Phase IV tests of the F-100A aircraft.

Par E. 2p - To correct this deficiency the bleed valves were staggered on production engines. In addition to this action, a study was initiated to consider the advantages of incorporating a modulated bleed system. This study is still current; however, the problem did not become as critical as it was anticipated. It is recognized that the staggered system may not be the optimum but it has proven to be acceptable.

Par E. 2q - It is believed that satisfactory and stall free engine operation to 50,000 feet altitude will be achieved in the near future. It has not been determined if the engine characteristics and/or fuel control system capabilities will permit totally stall free operation. J-57-P-21 engines with a redesigned compressor section are scheduled for F-100C No. 149 and subsequent.

Par E. 2r - The drag chute deployment handle has been relocated to a position on the shroud effective ship F-100A No. 154 and subsequent.

Par E. 2s - Refer to comment on Par E, 2b.

Par E. 2u - Flight experience has indicated that the F-100A does not have a pitch-up tendency as was originally anticipated during Phase II testing. No reports have been received by the WSPO indicating this characteristic is exhibited.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~
Reply to Phase Report

AF Technical Report No. AFFTC-53-33

~~CONFIDENTIAL~~

Par E. 1b - A mechanical stability device, referred to as a "Mach sensing stick force controller", was developed in conjunction with the original linear stabilizer control system to improve static longitudinal stability. The change from linear to non-linear stabilizer control system; however, has necessitated a redesign of this Mach box. The prototype of the redesigned Mach box will be made available to AFFTC on completion of present NAA tests.

Par E. 1c - A substantial improvement in low speed handling characteristics has been realized by the addition of the non-linear gearing system. Both the forces and stick motions necessary to control the airplane have been reduced for stabilizer trim positions used for landings. Increased aileron stick travel reduced sensitivity of lateral control and provided substantial improvement in low speed handling. Enlargement of the vertical tail, as a result of ECP F-100A-375, also contributed appreciably to the improvement of low speed handling characteristics.

Par E. 2a - This deficiency will be substantially reduced by the installation of the yaw damper on all F-100 aircraft. The production effectivity of this installation is F-100G, No. 164, with provisions made for retrofit of all inservice F-100 aircraft.

A prototype pitch damper is currently being flight tested by NAA and will be made available to AFFTC for testing as soon as the Contractor's tests are completed.

Par E. 2f - Pratt and Whitney has not provided a suitable configuration but this matter is still under study and it is anticipated that a final decision will be made in the next few weeks.

Par E. 2g - The stabilizer trim rate has been increased approximately 50%. This change in trim rate was achieved by increasing the speed of the trim motor. As a result of ECP F-100A-375, an impulse trim system has been installed in all F-100 aircraft which reduces the starting lag time thus producing an apparent increase in the trim rate.

Par E. 2h - There have been some minor changes initiated in the cockpit which have reduced the magnitude of this deficiency. Cognizance of this problem is maintained by the Laboratories and further improvement is anticipated with additional study.

Par E. 2k - A study has been made of an anti-skid system installation for the F-100A, F-100G, and F-100D airplanes. Approval has been granted for installation on F-100G, No. 164 and subsequent, and all F-100D airplanes. Retrofit of F-100A and C aircraft prior to No. 164 was disapproved by SMAMA, but is being further investigated by WADC.

~~CONFIDENTIAL~~

55WCS-8914 - 2

~~CONFIDENTIAL~~

55-2141

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Par E. 2l - There has been considerable improvement in the selection of drag chute configuration and mechanisms which have resulted in a high degree of reliability. The increase in reliability was accompanied by a slight decrease in actuating time; however, it is not anticipated that there will be any further reduction of time with the use of present equipment, which is considered to be the best available.

Par e. 2n - A substantial improvement in control response has been achieved by the modification required through implimentation of ECP F-100A-375. This modification included a redesign of the vertical tail and wing tip extensions and the installation of a non-linear gearing longitudinal control system. The merits and effects of this modification have been discussed elsewhere in this report and in WADC comments on Phase IV tests of the F-100A aircraft.

Par E. 2p - To correct this deficiency the bleed valves were staggered on production engines. In addition to this action, a study was initiated to consider the advantages of incorporating a modulated bleed system. This study is still current; however, the problem did not become as critical as it was anticipated. It is recognized that the staggered system may not be the optimum but it has proven to be acceptable.

Par E. 2q - It is believed that satisfactory and stall free engine operation to 50,000 feet altitude will be achieved in the near future. It has not been determined if the engine characteristics and/or fuel control system capabilities will permit totally stall free operation. J-57-P-21 engines with a redesigned compressor section are scheduled for F-100C No. 149 and subsequent.

Par E. 2r - The drag chute deployment handle has been relocated to a position on the shroud effective ship F-100A No. 154 and subsequent.

Par E. 2s - Refer to comment on Par E, 2b.

Par E. 2u - Flight experience has indicated that the F-100A does not have a pitch-up tendency as was originally anticipated during Phase II testing. No reports have been received by the WSPO indicating this characteristic is exhibited.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

FTDTP

Comments on Phase II Performance and Stability Tests YF-100A

Par E.1.a - It is believed that this comment is closely associated with the comment on low speed characteristics. Pilots conducting recent preliminary APGC/TAC evaluation tests consider the over-nose visibility of the F-100A during take-off and landing as satisfactory provided that the aircraft is operated at the recommended speeds; that is, approach at 180 knots and touchdown at 150 knots for the landing weight. To provide increased visibility for the airplane the pilot's seat adjustment has been increased to allow a 1-1/4 inch additional upward travel. Also a 5° wedge block is being provided for use behind the parachute. This provides the pilot with a more favorable seat position. The seat changes are effective in production airplanes 3, 5, 6, 7, 16, 21, and subsequent.

The camera reticle glass has been widened and its supports have been moved outboard to lessen the interference of pilot's vision. This change has been made on all production airplanes. Also, all production airplanes embody glass in the side panels which reduces the optical distortions that were apparent on the prototype airplane.

Par E.1.b - The Contractor is aware of the stability deficit due to speed changes in the Mach number range between .85 and 1.0. As pointed out previously this condition is apparent in all aircraft operating through the transonic speed range and is caused by the normal aft movement of aerodynamic center. The apparent decay of stability is only noted as speed is changed and is not present in maneuvering flight. The Contractor has designed and is presently flying a Mach box. The purpose of this Mach box is to artificially build stable forces into the control system such that the pilot does not sense the unstable condition. Present tests to date show the Mach box fulfills all the requirements of pilot forces, hysteresis and eliminates the transonic stick force reversal with speed. Preliminary NAA pilot comments indicate that the Mach box improves the flying qualities of the F-100A airplane in the Mach number range of .85 to 1.0, but the heavier stick forces and additional retrimming may have some disadvantage. Definite recommendations will be made as soon as more complete data and comments have been obtained.

Par E.1.c - The Contractor has installed an artificial stall warning device that shakes the rudder pedals and revised set of slats embodying a 15° rotation instead of the previous 10° on the F-100A aircraft. The results of these modifications have had a marked improvement as noted by the Company and Air Force pilots that have flown this configuration. The safe flight stall warning device and rudder pedal shaker has resulted in adequate stall warning for all low speed flight conditions, thus assuring that the pilot does not inadvertently enter the secondary stall. The revised slat configuration has provided better aileron control at the stall, improved longitudinal stability and has caused more uniform slat action than the original slat configuration.

CLASSIFICATION CANCELLED
(OR CHANGED TO 1)

(OR CHANGED TO)
TRD SM 4P 6-17-E J Jun 58
AUTHORITY: AMC REPORT NO. CD-356
DATE 15 Sep 58

Incl 1¹

~~CONFIDENTIAL~~

BY: Jy Holber

BY:

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Other items that are currently under design and study are as follows:

1. Rudder Aileron Interconnect

This is being proposed to help eliminate the adverse yawing characteristics and increase the aileron effectiveness in accelerated stalls. This modification is now being flight tested.

2. Slat Snatcher

The slat snatcher is a method of retracting the outboard slats when the aileron is deflected up. It is mechanically so arranged that it will operate only in the low speed range. The purpose is to further improve the aileron effectiveness and essentially eliminate the adverse yawing under low speed accelerated stalls.

3. Enlarge Aileron Span

The purpose of this study is to increase the aileron effectiveness in the low speed flight regime at the expense of some loss in rolling performance at low altitude high speed. It is anticipated that tests on this configuration can be made by July.

4. Boundary Layer Control Flaps

Studies and wind tunnel tests of this arrangement indicate that a significant decrease in landing touchdown speeds may be made (approximately 20 - 25 knots). At the present time model tests are near completion. It is anticipated a full scale flight test model will be flying by October 1954. The reason that BLC flaps is very attractive to the low speed flying range is because not only is the angle of attack decreased and the lift increased, but also there is a large decrease in drag which will result in minimum rates of descent occurring at a lower indicated air speed, thus allowing decreased touchdown speed.

Par E.2.a - The yaw damper has been installed in production F-100A #2. Pilot comments to date are that dead beat damping through the speed range has been achieved. The pitch damper is presently undergoing laboratory tests on the flight simulator. Flight tests indicate that the yaw damper improves directional stability considerably. AFFTC evaluation of the pitch and yaw damper will be requested at completion of the pitch damper installation in the Contractor's aircraft.

Incl 1²

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~
~~CONFIDENTIAL~~

Par E.2.b - The breakout forces have been materially decreased by reducing the size of the main feel bungees, (to 3-1/2 #/degree - initial slope, 1#/degree - final slope) increasing the bob-weight size and deleting the valve centering spring. These changes are effective on all production aircraft. Pilot comments on the configuration to date have been very favorable.

In addition to the above, a nonlinear gearing between the stick and stabilizer similar in principle to that recently tested on the F-86D with such excellent results, is being designed.

Par E.2.c - With installation of the 15° slats which provide a measure of improved stability at low speeds and with the installation of the stall warning device (rudder pedal shaker), it is felt that the conditions surrounding this initial comment have been eliminated.

Par E.2.d - The Contractor has developed and tested a linear nose wheel steering system and it was found to be a definite improvement over the system initially installed. It has been released for installation on all production airplanes.

Par E.2.e - NAA pilots have made landings and take-offs with dive brake extended. No difficulty was encountered or excessive take-off distances encountered. Dive brake drag at take-off and traffic speeds is similar to the F-86. Since complaints have not been raised against the lack of a light in the F-86, the WSPO has omitted this light. This was also done to simplify the aircraft, avoid excessive warning lights in the cockpit and divorce the aircraft from possible malfunctioning mechanical items not deemed essential.

Par E.2.f - Methods of accomplishing the intent of this comment have been under study. To date the best method has not been resolved. The most promising solutions are to locate provisions on the engine to actuate the nozzle position light already installed in the F-100A aircraft or to interlock the nozzle such that it cannot be open without afterburning. When those problems have been resolved, a proposal will be submitted.

Par E.2.g - The Contractor is currently pursuing a program to eliminate lag in the trim system. Some improvement has been obtained with the installation of the revised bungee and bob-weight and the deletion of the valve centering spring. Additional tests are scheduled to investigate the use of faster valves and improved trim mechanisms.

Incl 1³

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~ ~~CONFIDENTIAL~~

Par E.2.h - Additional adjustments will be made to minimize the canopy reflection, as more night flying is accomplished by the Contractor.

Par E.2.i - In tests to date, the operation of the production engine in the F-100A airplane indicates that this condition is completely eliminated.

Par E.2.j - There are three methods of removing the canopy from inside the aircraft; (1) by normal electrical operation, (2) mechanically unlocking and raising canopy, and (3) blowing canopy off with emergency escape procedure. The canopy may be removed by external operation by two methods; (1) electrical-ly canopy operation from exterior of the airplane, and (2) from exterior manual control unlock and manually raising the canopy. A parallel external blow-off control is considered dangerous due to inadvertent operation by unfamiliar ground personnel. No action taken on this deficiency.

Par E.2.k - During the course of the Contractor's flight testing, the wheel brake action has been found adequate; however, a number of mechanical difficulties have been experienced with the brakes. These problems are being taken care of by Bendix at this time.

The Contractor is further studying anti-skid braking devices which will allow the pilot to increase his braking effort without fear of skidding or destroying tires. A test installation will be made in the near future.

Par E.2.l - During the flight testing of the airplanes to date, reefed and non-reefed, parachute installations have been checked. The most successful parachute operations were made without reefing. The Contractor, AMC, and WADC have agreed to eliminate the reefing mechanism based on existing flight test experience. The pilot's drag chute deployment handle will be relocated to provide accessibility and simple operation.

Par E.2.m - The Contractor believes this comment arose from landings where some parachute panels failed. Subsequent testing did not cause any yawing to be apparent.

Par E.2.n - A number of studies are being pursued by the Contractor to improve response. The most notable are as follows:

Slat Snatcher
Aileron Rudder Interconnect
Extended Span Aileron

It is believed that the longitudinal control has improved and possibly the above comment is due, in some measure, to stick forces which subsequently have been improved by the addition of the revised bungee, bob-weight and elimination of the valve centering spring. In addition it is felt that they will be further improved by the revised variable gearing.

Incl 14

4

~~CONFIDENTIAL~~ ~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

Par E.2.c - Added complication for an automatic system does not warrant installation. Other aircraft have successfully operated without this feature. For simplification, this fix has been omitted. At times for identification, pilots want to extend light yet keep gear up. No action taken.

Par E.2.d - Both the left and right bleed valves on the XJ57-P7 engine (YF-100A) operate simultaneously, which results in a rapid thrust change of approximately 500 pounds when the valves open or close. To reduce the rapid thrust changes on the J57-P-7 engine used in the F-100A, the left and right bleed valves operate at different engine speeds. This arrangement should give two small thrust changes during an engine speed advancement on the F-100A instead of the one large thrust change. The Contractor is in the process of flight evaluating a "Holley" governor which will greatly reduce the tolerance of actuation presently required of the bleed doors.

Par E.2.g - This problem has virtually been eliminated on the production J57-P-7 engine. However, a thorough and complete power control tests program will be conducted on the F-100A airplanes to discover and correct any control system deficiencies. It is anticipated that the fuel schedule of the Bendix control used on the F-100A may require some alteration to adapt it to the performance characteristics of the aircraft-engine combination. The fuel control testing and rescheduling program for production aircraft is normal for new installations. This work will be done after the "Holley" governor evaluation has been made.

Par E.2.r - A study is being made for the relocation of the drag chute release handle to the upper left hand instrument panel similar to the drag chute deployment mechanism on the F-36D. The deletion of the reefing mechanism greatly aids in making this installation possible.

Par E.2.s - A Mach box was developed by NAA and test flown. Flight test showed the hysteresis band to be too large. The test unit is being redesigned. Proper functioning Mach box should improve longitudinal stick-free stability. Final analysis of this unit's improvement on aircraft longitudinal stability is to be made by AFFTC in Phase IV test on the F-100.

Par E.2.t - This statement is caused by a basic misconception of the slats on the F-100A airplane. These slats were designed primarily to improve the maneuverability, stability and limit lift coefficient at high Mach numbers. All indication to date show that these slats are operating essentially as they were designed to operate

The 15° slats give improved performance over the original 10° slats that were tested on the YF-100A airplane in that they have more uniform opening characteristics.

Incl 15

5

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~ ~~CONFIDENTIAL~~

Par E.2.u - Existing tunnel data show that the 15° rotation slats give sufficient improvement in stability at the stall that very little pitchup tendency should be noted. This item will be flight investigated during the normal testing required.

Par E.2.v - The defrosting duct has been redesigned to give adequate distribution of cockpit airflow.

Par E.2.w - Gear-down limit speed has been increased to 250 K.

Par E.2.x - For pilot comfort the seat back angle has been changed from 17° to 12° by addition of a 5° wedge. Installation will be in aircraft Serial Nos. 758, 760, 761, 763, 771, 776, and subsequent.

Par E.2.y - A delta pressure type gauge has been developed for installation in all F-100 aircraft to provide adequate means of determining thrust. In event delivery of this gauge does not parallel aircraft production, space provisions will be left on the pilot's panel for retrofit.

Incl 1⁶

6

~~CONFIDENTIAL~~ ~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

64-2

~~CONFIDENTIAL~~

WCG

17 June 1954

SUBJECT: (U) Phase II Performance and Stability Tests of the
YF-100A Aircraft AFFTC Technical Report No. AFFTC 53-33

TO: Commander
Air Force Flight Test Center
ATTN: DCS/Operations
Edwards Air Force Base
California

A study has been completed on the YF-100A final report, subject as above, on Performance and Stability Tests of the aircraft. Information as to action which has been or will be taken on the recommendations of the report is submitted herewith. Inclosure No. 1 is a list of remarks which are numbered to correspond to those sections of the report containing recommendations and conclusions.

1 Incl:
As stated above

ALBERT BOYD
Major General, USAF
Commander

IF INCLOSURES ARE WITHDRAWN (OR NOT ATTACHED) THE CLASSIFICATION OF THIS CORRESPONDENCE WILL BE CANCELLED IN ACCORDANCE WITH PAR. 25E, AFR-205-1.

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

54 - 3239

AF Technical Report No. AFFTC 53-55
December 1953

PHASE II FLIGHT TEST OF THE
NORTH AMERICAN YF-100A AIRPLANE
USAF No. 52-5754

ALFRED D. PHILLIPS
Flight Test Engineer

FRANK K. EVEREST, Lt. Col., USAF
Test Pilot

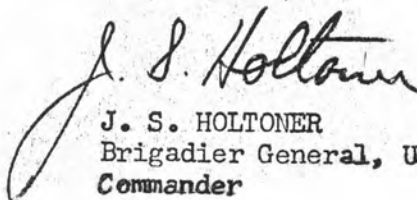
UNITED STATES AIR FORCE
AIR RESEARCH AND DEVELOPMENT COMMAND
AIR FORCE FLIGHT TEST CENTER
EDWARDS, CALIFORNIA

PUBLICATION REVIEW

This Report has been reviewed and is approved



H. A. HANES
Colonel, USAF
Director, Flight Test and Development



J. S. HOLTNER
Brigadier General, USAF
Commander

ABSTRACT

This report presents the results of Phase II performance and stability tests of the North American YF-100A air superiority fighter. The test results indicate the aircraft is superior in performance to any production fighter in the USAF. The most serious defects of the aircraft are the inadequate visibility over the nose during take-off and landing, the poor low-speed handling characteristics, and the negative to neutral static longitudinal stability experienced in level flight from approximately .8 Mach to maximum level flight speed.

TABLE OF CONTENTS

	<u>Page No.</u>
A. Purpose	1
B. Introduction	1
C. Summary of Test Results	2
D. Conclusions	4
E. Recommendations	4
APPENDIX I Factual Data	
A. Pilot's Comments	3--11
B. Performance	12--16
C. Stability and Control	17--26
D. Data Analysis Methods	28--35
E. Curves	36--136
APPENDIX II General Information and Photographs	
A. Three-View Drawing	2
B. Dimensions and Design Data	3--5
C. Operational Limitations	6
D. Fuel System	6
E. Power Plant	7
F. Weight and Balance	7
G. Test Instrumentation	8--9
H. Control System Diagrams	10--12
I. Photographs	13--31
APPENDIX III Flight Log and Original Data	
A. Flight Log	2--4
B. Original Data Corrected for Instrument Error	5--21
C. Take-Off Profiles	22--28
D. Landing Profiles	29--34

A. PURPOSE

This report presents the results of limited performance and stability flight tests of the YF-100A airplane, USAF No. 52-5754.

B. INTRODUCTION

1. Extent and Location of Flight Tests:

a. Flight tests of the aircraft were conducted to obtain stability and performance data to determine the aircraft's handling qualities and to compare actual performance with contractor's estimated performance.

b. The Phase II test program on the YF-100A airplane was conducted at the Air Force Flight Test Center, Edwards, California from 3 September through 17 September 1953. Thirty-nine flights, totaling 19 hours and 42 minutes, were required to complete the limited performance and stability tests.

2. Description of the Aircraft:

a. The YF-100A is the prototype version of the F-100A aircraft, an entirely new design of a single-place fighter capable of tactical operation in the high transonic and low supersonic speed range. The airplane is characterized by a thin, highly swept (45°), moderate aspect ratio wing, low drag fuselage and canopy, inboard ailerons (with no flaps), automatic leading edge slats, low positioned, one-piece horizontal stabilizer, and a thin-lip engine air inlet. The landing gear is of the tricycle type with the main gear retracting inboard into the wing and fuselage and the nose gear retracting aft into the nose section of the fuselage. Other noteworthy features of the airplane include a fuselage speed brake located in the bottom of the fuselage in the wing center section and an irreversible power control system with artificial feel.

b. The aircraft is powered by a single Pratt and Whitney XJ-57-P-7 continuous flow axial type gas turbine with afterburner rated at 13,800 lbs thrust uninstalled maximum power. The engine has a split compressor, split turbine, a hydro-electronic fuel control and is equipped with a two-position exhaust nozzle. The two spools of the compressor and turbine are mechanically independent of each other.

3. Test Configuration:

The aircraft was tested in the clean configuration and with two empty 275 gallon external fuel tanks attached to the undersides of the wings. The test weight of the aircraft was found to be 25,370 pounds in the clean configuration with fuel and oil tanks full. This weight included all test instrumentation, 1289 pounds ballast and the pilot. The aircraft was loaded to obtain a mid cg position and all tests were made at this one loading. The cg during flight was not controlled as the aircraft utilized automatic fuel sequencing. The cg position varied from a most aft position of 31.7 per cent MAC to a most forward position of 29.5 per cent MAC as fuel was consumed during flight. Detailed weight and balance data and photographs of the test aircraft are presented in Appendix II.

C. SUMMARY OF TEST RESULTS

1. Performance:

a. The results of the performance tests of the YF-100A type aircraft indicate that it is superior to any production aircraft in the United States at this time. The YF-100A can take-off at 25,000 pounds gross weight using 3000 feet of runway on a standard day at sea level. The aircraft can then climb at a rate of 14,800 feet per minute (at sea level) and reach an altitude of 45,000 feet in 4.3 minutes from best climb speed at sea level. Approximately two minutes must be added to obtain time to take-off and accelerate to best climb speed. The YF-100A aircraft can continue to climb to a combat ceiling (500 feet per minute) of 53,000 feet. With afterburner off, the climb is very slow (16 minutes from sea level to 40,000 feet). It is felt that climbs without afterburner will not be utilized by tactical organizations to a very large degree. At 35,000 feet a level flight Mach number of 1.04 is attainable. Maximum Mach number obtained was 1.39 indicated in a dive from 51,000 to 20,000 feet. There is no wing roll-off as previously experienced in F-86 type aircraft, no high Mach number buffet, no objectionable trim changes or other noticeable compressibility effects. Rates of descent reached approximately 50,000 feet per minute. This airplane is strictly a hit-and-run type fighter when utilized against present day fighters; i.e., it should not be expected to fight with an F-86 at the F-86's speed. This condition is comparable to the F-80 vs the F-51. Comparative performance between the YF-100A and other present day fighters is presented below:

COMPARATIVE PERFORMANCE

	<u>YF-100A</u> <u>Test</u> <u>Results</u> <u>Clean</u>	<u>F-86E</u> <u>Memo</u> <u>Report</u> <u>FTD 51-19</u> <u>Clean</u>	<u>F-86D</u> <u>Ref A F</u> <u>Tech</u> <u>Rpt 53-26</u> <u>Clean</u>	<u>F-94C</u> <u>Ref A F</u> <u>Tech</u> <u>Rpt 53-30</u> <u>Tip Tanks</u>	<u>F-94B</u> <u>Ref A F</u> <u>Tech</u> <u>Rpt 52-26</u> <u>Tip Tanks</u>	<u>F-89A</u> <u>Memo</u> <u>Report</u> <u>FTR 51-13</u> <u>Tip Tanks</u>
Max. Mach No. (max. power)	1.04 35,000 ft	.90 25,000 ft	.923 35,000 ft	.835 35,000 ft	.785 35,000 ft	.874 25,000 ft
Rate of climb at S. L. (ft/ min)(max. power)	14,800	8,800	12,900	9,500	7,100	7,700
Rate of climb at 30,000 ft (max. power)	10,500	2,850	4,900	4,500	3,000	4,900
Combat ceiling (max. power) 500 ft/ min.	53,000	46,000	47,000	47,000	43,000	39,500

COMPARATIVE PERFORMANCE
(Continued)

	F-86E	F-86D	F-94C	F-94B	F-89A
YF-100A	Memo	Ref A F	Ref A F	Ref A F	Memo
Test	Report	Tech	Tech	Tech	Report
Results	FTD 51-19	Rept 53-26	Rpt 53-30	Rpt 52-26	FTR 51-13
Clean	Clean	Clean	Tip Tanks	Tip Tanks	Tip Tanks

Time to
Climb to
45,000 ft
min (max.
power)

4.3 15.5 9.3 10.5

b. Engine performance and reliability are considered excellent. The afterburner is especially reliable inasmuch as afterburner "lights" can consistently be made at any altitude, speed, or configuration. Three objectionable characteristics associated with engine operation are: (1) slow, sluggish engine response to control application, (2) the opening and closing of compressor surge bleed valves when operating at power settings of approximately 90 percent rpm, which cause sudden increase and decrease in thrust, (3) compressor stall as occurred when applying power at low power settings (70 percent rpm).

2. Stability and Handling Characteristics:

a. The airplane is easy to taxi with the exception of high rudder forces. Take-off is satisfactory except at lift-off when visibility is limited. The airplane tends to wallow somewhat due to low airplane response to stabilizer movement. The airplane must be "pulled off" the runway as it exhibits no inclination to fly itself off even at speeds up to 180 knots. The visibility is not satisfactory during climb because of the nose high attitude at best climb speed.

b. The airplane exhibits negative to neutral static longitudinal stability from approximately .8 Mach number to maximum Mach number. Trimming the aircraft for normal cruise is very difficult because of this factor. It is very annoying for formation flying and would possibly be detrimental to the airplane as a gun platform. In the approach configuration, the static longitudinal stability is positive; however, it is felt that it could possibly be increased and low speed handling characteristics could be improved. The YF-100A airplane exhibits a lateral, directional, and longitudinal poorly damped dynamic stability characteristic which occurs at high altitude and high Mach. The longitudinal damping improves with decreased altitude but lateral and directional damping is unsatisfactory at all altitudes. This, too, is detrimental for formation flying and accurate gun firing. Longitudinal control is sensitive, and at transonic speeds of approximately .8 to .9 Mach number, it is similar to that exhibited by the F-86D aircraft. This is unsatisfactory, however, it is felt that pilots could live with this condition.

c. The speed brake is very effective and creates a considerable amount of drag with little objectionable buffeting. The speed brake may be used on landing; however, because of the small buffet encountered at full extension, it gives the pilot a false impression that the airplane may be nearing the stall.

d. The stalling characteristics are poor. At 10,000 feet, from 160 knots to 110 knots, there is a yawing and pitching tendency that is quite uncomfortable to the pilot. At 110 knots the left wing drops uncontrollably; full right ailerons and rudder not being capable of controlling the wing drop. The stall itself is unsatisfactory because of this drop but is considered acceptable because the normal touchdown speed is well above the stall speed.

e. Landing of the YF-100A airplane is considered more difficult than any other production fighter. It exhibits poor lateral and longitudinal low speed handling characteristics. The airplane response to control movement is slow and the pilot finds himself constantly over-controlling during final approach, both because of the high stabilizer breakout forces and the low airplane response rates. During the latter part of the final approach, the nose is very high and it is extremely difficult to judge the altitude and position of the airplane when nearing touchdown point. When landing at night the limited visibility makes landings dangerous inasmuch as the pilot has little or no reference to either side as he does during daylight hours. On one occasion during a landing, the pilot landed 2000 feet short of the runway as a result of the high rate of sink and poor visibility. Touchdown speeds are high because of the ease of dragging the tail. On one landing the tail touched the runway at a touchdown speed of 136 knots. Normal touchdown speeds will be approximately 150 knots or above. It is anticipated that all landings will be made utilizing the drag chute. The drag chute is unacceptable at this time owing to the considerable amount of yaw it creates after deployment.

D. CONCLUSIONS:

It is concluded that:

a. The YF-100A series airplane out-performs any other production fighter in the USAF at this time.

b. The YF-100A airplane is unacceptable for service use at this time because of poor visibility over the nose during take-off and landing, neutral and negative static longitudinal stability, and the poor low-speed handling characteristics.

c. The YF-100A aircraft has a good potential and can be made into a useful tactical weapon.

E. RECOMMENDATIONS:

1. The following recommendations pertain to major deficiencies which should be changed prior to acceptance or delivery of the aircraft to tactical units. It is recommended that:

a. Forward visibility be increased and improved during take-off and landing.

b. The neutral and negative static longitudinal stability exhibited be corrected to insure ease of formation flying and to increase the potential of the airplane as a gun platform.

c. The poor handling characteristics during take-off and landing configuration be improved.

2. The following recommendations pertain to deficiencies which are considered unsatisfactory and should be corrected as soon as possible. It is recommended that:

- a. The lateral-directional and longitudinal dynamic stability be improved.
- b. The stabilizer breakout forces be decreased.
- c. The poor stalling characteristics be improved.
- d. The rudder forces during taxiing and low-speed flight be decreased.
- e. A dive brake warning light or position indicator be installed on the pilot's panel to prevent inadvertent take-off with dive brakes extended.
- f. An afterburner light be installed to give indication that the afterburner is on.
- g. The stabilizer trim response rate be increased.
- h. The canopy reflections of cockpit instrument and flood lights be eliminated for night flying.
- i. The slow, sluggish engine response to control application be eliminated.
- j. A switch be installed on the exterior of the fuselage so that the canopy may be blown off by external operation on the ground.
- k. The wheel braking action be increased.
- l. The time of actuation of the drag chute be decreased.
- m. The yawing moment caused by drag chute deployment be eliminated.
- n. The rudder, aileron, and stabilizer airplane response rates at low speeds be increased.
- o. An automatic landing light retraction be coupled with gear retractors to prevent inadvertent flight with the landing light extended.
- p. The sudden increase or decrease of thrust during actuation of the surge bleed valves be eliminated.
- q. The compressor stall experienced at low power settings be eliminated.
- r. The drag chute deployment handle be relocated to a position behind the throttle.
- s. A study be made to determine the feasibility of installation of a q device for better aerodynamic longitudinal control forces throughout the entire altitude and speed range.
- t. The slats be modified to insure that they do not extend in unaccelerated flight above .9 Mach number at high altitude.
- u. A study be made to insure that the pitch-up tendency will not cause inadvertent over-stressing of the airplane at low altitude, and high speed when making abrupt control movements.
- v. The defrosting duct presently under the left console be relocated to prevent excess heating of the left console.
- w. The limit gear down speed be increased to 250 knots IAS.
- x. The pilot seat tilt back angle be reduced to approximately 7 to 9° for pilot comfort.
- y. An adequate means of determining thrust during take-off and during flight be provided by an engine pressure ratio type gage.

APPENDIX I
FACTUAL DATA
YF-100A USAF No. 52-5754

	<u>Page No.</u>
A. Pilot's Comments	
1. Cockpit Description	3
2. Taxiing and Ground Handling	3
3. Take-off and Initial Climb	4
4. Climb	4
5. Level Flight	5
6. Flight Control System	5
7. Approach and Landing	9
8. High Altitude Operation	10
9. Night Operation	11
10. Engine Operation	11
11. General Aircraft and System Functions	11
B. Performance	
1. Take-offs	12
2. Climbs	12
3. Level Flight	13
4. Airspeed Calibrations	15
5. Landings	15
6. Engine Static Thrust Calibration	16
C. Stability and Control	
1. Test Configuration	17
2. Mechanical Characteristics of the Control System	17
3. Stabilizer Control Effectiveness	19
4. Stalls	20
5. Static Longitudinal Stability	20
6. Dynamic Longitudinal Stability	21
7. Maneuvering Flight Characteristics	22
8. High Speed Flight Characteristics	24
9. Speed Brake Operation	24
10. Static Directional Stability	25
11. Dynamic Lateral Directional Stability	25
12. Adverse Yaw	26
13. Lateral Control	26

TABLE OF CONTENTS
(continued)

		Page No.
D.	Data Analysis Methods	
	1. Introduction	28
	2. Notation	29
	3. Take-Offs	31
	4. Climbs	31
	5. Level Flight	32
	6. Thrust and Drag	33
	7. Lift and Drag Coefficient	33
	8. Airspeed Calibration	34
	10. Temperature Recovery Factor	34
	11. Landings	34
	12. Stability and Control	35
E.	Performance and Stability Curves	
	Figure Number	
	1 Mach number versus altitude	36
	2-3 Climb Performance	37--40
	4-5 Nautical Air Miles per Pound of Fuel	41--42
	6-7 $N\sqrt{288/T_a}$ versus Mach number	43--44
	8-9 RPM versus Calibrated Airspeed	45--46
	10-13 C_L versus C_D and C_L^2 versus C_D	47--50
	14-15 Drag versus Equivalent Airspeed	51--52
	16-17 Static Thrust	53--54
	18-22 Tail Pipe Probe Thrust Calibration	55--59
	23 Airspeed Calibration	60
	24-25 $N_2\sqrt{288/T_{t2}}$ versus Fuel Flow	61--62
	26 N_2 rpm versus Inlet Temperature	63
	27 Airflow versus N_1 rpm	64
	28-30 Control Friction	65--67
	31 Landing Time History	68
	32-34 Stalls	69--74
	35-37 Static Longitudinal Stability	75--77
	38-47 Dynamic Longitudinal Stability	78--87
	48-53 Maneuvering Flight Characteristics	88--93
	54-55 Abrupt Pull-Ups	94--95
	56-57 Limit "g" Diving Turn	96--97
	58-59 High Speed Dive	98--101
	60-61 Speed Brake Opening	102--103
	62-63 Sideslip Characteristics	104--105
	64-83 Dynamic Lateral Directional Stability	106--125
	84-87 Adverse Yaw	126--129
	88-89 Summary of Rolling Characteristics	130--131
	90-92 Lateral Control Characteristics	132--134
	93-94 Maximum Allowable Deflection Aileron Rolls	135--136

A. PILOT'S COMMENTS

1. Cockpit Description:

a. Access to the cockpit is by use of a ladder that hooks over the edge of the cockpit. There are foot steps and handholds installed in the side of the fuselage; however, they are too high to be utilized without the use of an extension ladder to begin the initial climb up the side of the fuselage. Generally, the cockpit is considered acceptable; however, the following discrepancies should be corrected in the production aircraft:

b. The lever for drag chute deployment is located directly outboard of the throttle. It is quite awkward to manipulate, especially when the throttle is in the retarded or low power position. The drag chute deployment handle should be relocated to a position directly behind the throttle.

c. The windshield defrosting handle is too close to the landing gear handle when the gear is down. The windshield defrost handle should be relocated approximately one inch or more outboard from its present position and for ease of operation, the landing gear handle should also be relocated or extended 0.5 inches outboard from its present position.

d. The pilot's seat is presently too difficult to move up. A stronger spring or bungee should be installed for ease of operation.

e. The map case cover opens inboard and should be changed to lift up and outboard for ease of operation.

f. The present warning lights located on the left console should be relocated up and more in line with the pilot's vision, such as those presented at the F-100 Contractor Technical Compliance Board. On several occasions the pilot had the oil overheat warning light come on and did not notice it until after some time had elapsed.

g. It is too difficult to read the position indicators for the rudder pedal adjustment. The rudder pedals should be made adjustable fore and aft with one control instead of individual controls.

h. Visibility from the cockpit is considered excellent in the taxi and level flight attitudes, and visibility inside the cockpit is considered satisfactory with the exception of the above mentioned warning lights. (See page 24, Appendix II for photographs of the cockpit).

2. Taxiing and Ground Handling:

The YF-100A Aircraft is fairly simple to taxi and is considered similar to the F-86 type aircraft in ground handling. One difficulty noticed is the high rudder pedal forces when utilizing nose wheel steering. This makes the use of nose wheel steering somewhat difficult and the pilot finds himself prone to use the brakes more than is necessary. The nose wheel steering ratio appears to be satisfactory during both low and high taxi speeds. The YF-100A

aircraft was taxied in cross winds up to 20 knots with little or no difficulty. Braking action during taxiing is considered good. The brake pedal forces are light and are considered far superior to those of the F-86 type aircraft. During taxiing the rudder is of little or no use and brakes and/or nose wheel steering must be used at all times.

3. Take-Off and Initial Climb:

Take-offs were made with full power (afterburner on, speed brakes in). The brakes were released when the afterburner cut in and nose wheel steering was used up to approximately 100 knots IAS. Rudder control is available at approximately 60 knots; however, the response of the airplane to the rudder is slow and it is recommended that nose wheel steering be utilized up to approximately 100 knots. Inasmuch as the horizontal stabilizer, if held in full nose up position, creates quite a bit of drag, the stick is left at the take-off trim position (approximately 4 degrees nose up) until an indicated airspeed of 140 knots is obtained. At this time the stick is pulled firmly to the full nose up position and the airplane is flown off the runway at 150 knots IAS. Because of the unusually high angle of attack required for take-off with the highly swept, low aspect ratio wing with no flaps, a conscious effort must be made to pull the aircraft off the ground. On one occasion, an indicated airspeed of 180 knots was attained on the ground when the nose wheel was held approximately 6 to 12 inches off the ground and it was still necessary to increase the pitch attitude for take-off. After the airplane broke ground, the control stick back pressure was released slightly and the gear was retracted immediately. There is little or no trim change during gear retraction; however, the pilot found that there is a slight tendency to over-control laterally because of the light aileron breakout forces. Take-off is considered satisfactory; but, because of the above mentioned directional problem of the high rudder forces and low airplane response to rudder, it appears that formation take-offs will be difficult. During the transition time from lift-off to gear retraction, there is little forward visibility because of the nose high attitude of the airplane. This is not considered unsatisfactory, however, because the transition time between lift-off and gear retraction is short and the pilot can level the airplane out almost immediately after lift-off to increase the speed to the recommended climb speed. The longitudinal stick forces are considered too high and this, added to the fact that immediately after breaking ground the response of the airplane to longitudinal control position is slow, may result in over-controlling slightly. This is not considered satisfactory; however, it is acceptable because of the previously mentioned rapid acceleration of the airplane during the transition period to higher speeds where longitudinal control is satisfactory.

4. Climb:

After take-off the YF-100A aircraft accelerates very rapidly to the best climb speed and visibility is considered acceptable during this part of the flight. Once the recommended climb speed is attained, the nose is pulled up sharply and the YF-100A climbs very rapidly to its combat ceiling. The airplane has an extremely nose high attitude during climb and visibility is not considered satisfactory. No unusual or unsatisfactory control characteristics are evident during the climb, but care must be exercised to prevent the airplane from exceeding the best climb speed, inasmuch as the nose high attitude and rapid rate of climb

combine to give the pilot a tendency to level out for better visibility.

5. Level Flight:

The maximum level flight Mach number attained was approximately 1.04 at 35,000 feet and the maximum level flight indicated airspeed attained at sea level was 631 knots (.96 Mach no.). Aerodynamic characteristics are not considered satisfactory inasmuch as there is a slight yawing and pitching moment that makes formation flying difficult and also makes the airplane a poor gun platform. Further, the longitudinal breakout forces are considered high. These, coupled with negative longitudinal static stability throughout the cruising range, cause the pilot to be constantly retrimming and moving the stick in an attempt to fly the airplane at a constant altitude. The cockpit pressurization is excellent. Cockpit refrigeration is inadequate during low level, high speed flight as the pilot is too warm even when dressed in a light flying suit. It is believed that part of this heat results from the defroster duct that runs along the left console since, it is not possible to touch any part of the left console with the exception of the throttle due to the high temperature of the metal parts. Some consideration should be given to relocating the defroster-heater duct and/or control valve to keep this temperature problem to a minimum. The canopy fogged over at times when at extreme altitudes but it cleared rapidly upon actuation of the de-fogging system. The seat is comfortable but it is tilted back too much and should be straightened possibly three to five degrees.

6. Flight Control System:

a. All three flight control systems are irreversible and are powered by two engine driven hydraulic pumps, one taken off each rotor section. There is an emergency air driven hydraulic pump which was never utilized, and it is not known whether it would be usable in case of emergency or not. Control "feel" is obtained by use of bungees at the control stick and on the rudder. In the case of the stabilizer control, there is also a bob-weight. There is no aerodynamic or "q" device installed that decreases or increases the stick forces with increase or decrease of airspeed and/or altitude. Incorporation of a "q" device would improve the control "feel". Stabilizer breakout forces are considered too high. Lateral control forces are considered satisfactory at all speeds. Rudder forces are considered acceptable at high and intermediate speeds, but are too high at the approach and take-off speeds. Lateral and longitudinal trim is obtained by use of the trim button located on top of the control stick and there is a toggle switch on the left console to actuate rudder trim. Aileron and rudder trim are satisfactory and the stabilizer trim speed is also considered satisfactory; however, stabilizer trim motor response is too slow and is considered unsatisfactory.

b. Longitudinal Stability

(1) Stabilizer control and effectiveness for take-off.

Stabilizer control during take-off is considered satisfactory. It was possible to lift the nose wheel from the ground between 110 and 120 knots IAS.

The airplane was flown only at one mid. cg (32 percent), however, it was felt that, inasmuch as the airplane lifts off at 150 knots, the stabilizer control effectiveness would be satisfactory even at a forward cg. As mentioned previously immediately after breaking ground there is a tendency to over-control because of the high breakout forces and low response rate. If possible, the high breakout forces should be eliminated inasmuch as some pilots might have difficulty during take-off in turbulent and/or cross wind conditions. There is little or no trim change during the transition period from lift-off during gear retraction until recommended climb speed is attained.

(2) Dynamic stability.

Longitudinal dynamic stability is satisfactory in the low and intermediate speed ranges. At high Mach number (.98 and above) and high altitude (45,000 feet) the controls free tests indicate marginal dynamic stability as it takes 4 to 5 cycles before a longitudinal oscillation dampens. When an attempt is made to stop the longitudinal oscillations by use of the controls the pilot tends to over-control, similar to the F-86F and F-86D models. This condition is considered unsatisfactory. The sensitivity of the controls makes it very easy to overshoot the desired acceleration in maneuvering flight. Attempts to correct the "overshoot" usually cause the pilot to induce a longitudinal oscillation similar to that encountered in the F-86D aircraft. This condition is not as severe as that experienced in the F-86D; however, it does warrant some consideration for correction because this condition is unacceptable when flying formation and in rough air.

(3) Maneuvering stability.

Maneuvering stability is considered to be unsatisfactory. It is felt that at a high indicated airspeed it would not be too difficult to exceed the limit load factor by abrupt displacement of the control stick because of the inherent pitch-up tendency experienced in swept wing fighters. This pitch-up is more noticeable than that of earlier swept wing fighters, such as the F-86 aircraft, and it is considered that this is due to the increased wing sweep angle. When maneuvering the YF-100A aircraft, upon application of any "g" force, the drag increase becomes quite apparent and it may be determined now that this fighter can only be utilized as a hit and run gun platform as any attempts to linger and try to out-turn any other present day fighter would result in a great loss of speed. This, coupled with poor turning capabilities at lower speeds, would mean that the opponent would invariably end up in a stern attack on the YF-100A aircraft. As long as the YF-100A is kept at high speeds and low "g" factors, it should have a very good potential as a fighter. During one phase of the test program, at an indicated Mach number of 1.1 and at 35,000 feet altitude, the stick was pulled full back. A maximum acceleration of 5.6 "g's" was obtained. The Mach number decreased to less than 1.0. A pitch-up was noticed at 5 "g's" which resulted in an acceleration of 5.6 "g's" even though the pilot was attempting to hold a constant 5 "g" pull up. This may be an indication of the difficulty to be expected at lower altitudes where more "g's" may be obtained from abrupt application of the stick and over-stressing the airplane may be possible. Further study is recommended to insure that the airplane will not be easily over-stressed during accelerated maneuvers at medium to low altitudes and at high airspeed.

(4) Static Stability

Static stability of the YF-100A aircraft is considered unsatisfactory. At cruising and high Mach numbers the YF-100A exhibits negative stability. When increasing speed from a pre-determined trim speed, stick position and forces required to maintain the slower speeds are in the wrong direction (pull), and when decreasing speed from a pre-determined trim speed, the stick position and force changes indicate neutral to very slight positive stability. This is considered an unacceptable condition and every effort should be made to correct this condition prior to delivery of production airplanes. In the approach configuration the airplane exhibits positive static stability; however, because of high breakout forces it is difficult to fly in a trim condition. The unsatisfactory static stability tends to make the airplane difficult to fly in formation and results in a poor gun platform.

(5) Trim Change Characteristics

There is a slight trim change when extending the gear, as the left gear apparently extends first, causing a yawing moment to the left; however, it is easily controlled by use of the right rudder. When extending speed brakes at high speeds there is little noticeable trim change and this is considered excellent. There is approximately a five to ten pound nose down trim change when igniting the afterburner at the intermediate to higher speed ranges. This is also controllable and is considered acceptable. If possible, it is recommended that this trim change be alleviated, since in the case of formation flying, it would present a small problem.

(6) Stabilizer Control and Effectiveness for Landing

Stabilizer control and effectiveness for landing is considered satisfactory; however, the response of the airplane to control movement is considered unsatisfactory. Coupled with the high breakout forces and low response rates, the pilot finds himself constantly overshooting his desired attitude during the approach and landing flare. No trouble exists on control effectiveness inasmuch as it is quite easy to drag the tail during touchdown. The tail was dragged on one landing when touchdown was made at an indicated speed at 136 knots. Most touchdowns were made at approximately 150 knots IAS. The stabilizer breakout forces should be reduced and a study should be made to increase airplane response to control movement during the approach configuration. Perhaps the control effectiveness would be improved by return to the old type elevator control linkage, similar to that of the F-86E, for slow speed operation.

(7) Stalls

Stalls were made only at the mid cg position. The stalling characteristics are considered poor. Several full stick back stalls were made at both low and high altitudes. At low altitudes (10,000 ft.) from approximately 160 knots down to the stall of 110 knots there is a yawing and pitching tendency that is quite uncomfortable to the pilot. At 110 knots the left wing drops uncontrollably inasmuch as full right aileron and full right rudder will not hold the wings level. The stall itself, when reached at approximately 110 knots, is

unsatisfactory but is considered acceptable because there is no reason why the airplane will ever be flown at this low speed, especially as the normal touch-down speed will be 150 knots or above. Stall recovery is made by releasing back pressure on the stick while still holding full right aileron and right rudder; thus permitting the speed to increase until such time as the aileron and rudder effectiveness return. Accelerated stalls are considered satisfactory inasmuch as the airplane buffets and pitches similar to other swept wing aircraft and stall recovery is made by releasing back pressure on the control stick. At high altitudes (35,000 ft) during unaccelerated stalls the stick can be held back to full aft position and although the airplane is prone to pitch and roll, there is enough aileron and rudder control to rudder-walk the airplane through loss of several thousand feet of altitude.

c. Directional stability

(1) Cross wind take-off Characteristics.

The cross wind take-off characteristics of the YF-100A are considered satisfactory inasmuch as, with the nose wheel left on the ground to 140 knots, there is little or no tendency for the upwind wing to rise. Directionally the airplane has a tendency to turn into the wind and the normal procedure is to correct with nose wheel steering. The best procedure is to use nose wheel steering to correct for turning tendencies up to higher indicated speeds as the velocity of the cross wind is increased. Brakes could be used for directional control. However, brake response is quite sensitive and the pilot is somewhat prone to over-control; thus, it is best if the nose wheel steering is relied upon for directional control during cross wind take-offs.

(2) Dynamic lateral-directional Stability.

Dynamic lateral-directional stability is considered unsatisfactory throughout the entire combat speed range. Directional-lateral oscillations, resulting from rudder displacements, are not damped after 5 cycles and show no tendency to damp at all during slightly turbulent air conditions. This is considered unacceptable and should be corrected prior to acceptance or delivery of the aircraft to tactical units.

(3) Static directional Stability.

Static directional stability of the YF-100A aircraft is considered satisfactory at all speeds. The rudder and aileron position and forces are in the right direction for a given yawed condition. At airspeeds of 160 knots and below, with full right rudder, the airplane will roll to the right even though full left aileron is applied. This is undesirable but acceptable because full rudder sideslips need not be made in this airplane during the approach and landing flare. The rudder forces are satisfactory in the high and intermediate speed ranges, but during approach speeds the rudder forces are too high, and coupled with the low airplane response to the rudder, rudder control is unsatisfactory.

d. Lateral Control

The lateral control of the YF-100A aircraft is satisfactory. The rate of roll is high and the time to reach peak rate of roll is satisfactory at all speeds down to approximately 175 knots. From 175 knots down to the touchdown speed of 150 knots, the rate of roll is satisfactory, but the response to aileron deflection is slow and some difficulty may be experienced during turbulent and/or cross wind landing conditions. There is a slight amount of adverse yaw present in the lower speed range; however, it is considered acceptable inasmuch as this speed is below the practical combat speed of the airplane.

e. Dive and High Speed Flight Characteristics

High speed dives were made up to an indicated mach number of 1.39. During the dives there was no rolling tendency such as that experienced in F-86 type aircraft and the YF-100A exhibits the best diving characteristics of any fighter flown to date. Recovery from dives can be made by either actuating the speed brakes, pulling power off and slowing down, or by pulling out at the desired altitude. Again, caution must be exercised when making high "g" pull-outs because of light stick forces necessary to recover from high speed dives. As previously mentioned, it is believed that it would be quite easy to over-stress the airplane if caution were not exercised during high speed flight. One dive was made with drop tanks attached (empty) to an indicated Mach number of 1.2. There is no perceptible difference in handling characteristics of the airplane. There is no buffet exhibited with the tanks installed and the installation is considered to be superior to that on any previous model or type.

7. Approach and Landing:

Approach and landing of the YF-100A aircraft is difficult and at night is considered to be dangerous. Initial approach is made at an altitude of approximately 1000 ft. above the runway. Indicated airspeed at peel-off is between 300 and 350 knots and, at the time of peel-off, speed brake is extended and power reduced to 80 percent. As the airplane slows down to gear extension speed of 210 knots, the gear is extended and the speed brake is retracted. The gear limit speed of 210 knots is too low and should be increased to 250 knots. Speed during base leg and turn onto final approach is held at 170-180 knots and, after roll-out onto final is made, 170 knots is held until approximately $\frac{1}{4}$ mile from the end of the runway. At this time power is decreased and flare started by pulling the nose up. The speed brake is extended, if necessary, to increase the drag. Forward visibility during the final approach is unacceptable. Poor forward visibility plus high stabilizer breakout forces and low airplane response to stabilizer movement cause the pilot, during the latter part of the approach, to more or less fly blind and "hunt" for the runway. During one night landing the landing was 2000 feet short of the end of the runway because the pilot was unable to see the runway during the final approach. In the approach configuration the airplane exhibits a yawing tendency and feels quite sluggish. The pilot has a tendency to keep his airspeed high and is definitely reluctant to reduce power to slow the airplane to the normal touchdown speed of 150 knots. It is essential that the low speed handling characteristics be improved prior to accepting the first production airplane and that a study be initiated immediately to increase the visibility over the nose. In view of the trend toward low

aspect ratio and the resulting higher angles of attack for landing, the requirement for good visibility during the approach and landing conditions may make it advisable to eliminate from consideration the use of nose inlets so that the cockpit may be placed in the extreme forward part of the fuselage to enable the pilot to have good visibility over the nose. Speed brakes were used on some of the landings but the high rate of sink with brakes retracted does not require their use. When speed brakes are utilized in the full down position the rate of sink is so high that it is necessary to add power. This is accompanied by a slight buffet which is objectionable because it feels as if the airplane is nearing a stalled condition. It would be desirable to incorporate a speed brake position indicator so that partial speed brakes could be used during the approach and landing flare. Immediately after touchdown speed brakes are extended to full open position and if the drag chute is not utilized, the nose wheel is allowed to contact the runway and immediate braking is applied. If the drag chute is used, the speed brake is extended immediately after touchdown and the drag chute employed shortly thereafter. When the drag chute opens there is a strong yawing tendency and it is possible that the airplane, if not controlled, could yaw off the runway to one side or the other. The best method of controlling this yaw is by use of nose wheel steering, inasmuch as airplane response to rudder is slow and no immediate corrective action is noticed. This yawing tendency during drag chute operation is unacceptable and should be corrected prior to delivery of the first production airplane. Because of the high landing speed of this airplane the drag chute should be used for all landings. Braking action is good but after the brakes heat up they become noticeably weak during the latter part of the landing roll. The wheel brakes need to be improved to insure full braking action throughout the entire landing roll. Use of the landing chute during all landings will decrease the wear on both the brakes and tires. The drag chute effectiveness is excellent but the time delay from actuation of the control handle to chute deployment is too long. Landings were made with cross winds up to 20 knots at 90 degrees to the landing path and they were executed by dropping the wing and crabbing into the wind. At touchdown, the wings were leveled and the airplane kicked straight, thus aligning the airplane with the runway immediately before or at touchdown.

8. High Altitude Operation:

The airplane performance is satisfactory at altitudes up to 40,000 feet. Maximum level flight speed at 35,000 feet is approximately 1.04 Mach number. Above 40,000 feet there is a 1 "g" buffet condition that is unexplained; it is possible that this may result from partial extension of the slats. On almost every flight above 40,000 feet, outboard portions of the slats were observed by the pilot to be extended from 1" to 2". If they were fully closed, it is believed that both the performance and the maneuverability of the airplane would be improved. The highest altitude attained during the Phase II tests was 55,000 feet. The airplane at this altitude, of course, is unusable as it barely flies and cannot be maneuvered at all without loss of altitude. It is believed that the highest practical altitude at which the airplane could be used would be approximately 50,000 feet. As previously mentioned, cabin pressure differential is excellent and the pilot's cockpit is comfortable. The cockpit heating system is adequate and the de-fogging system is superior to that of any other aircraft flown. The 2.75 PSI differential pressure provided in the YF-100A aircraft was never used and it should be eliminated from the production airplanes because it is doubtful if it will ever be used in tactical organizations.

9. Night Operation:

The YF-100A airplane was flown at night on two occasions during the Phase II tests. The first flight was aborted immediately after take-off due to the failure of the gear to retract. At this heavy gross weight, speed was increased during the final approach but because of the poor visibility over the nose and the inability of the landing light to light up the runway far enough ahead of the pilot to be able to see, touchdown was 2000 feet short of the runway at an indicated speed of 170 knots. During taxi and take-off the landing light, utilized as a taxi light, is satisfactory. During take-off roll the technique is no different than during daylight hours, except that, when the nose wheel is lifted prior to breaking ground, the pilot practically is on instruments inasmuch as he is unable to see over the nose to judge his attitude. Reference was made to the altimeter, and, upon reaching 200 feet, the airplane was leveled off and speed was increased to recommended climb speed. The cockpit lighting is excellent; however, the right instrument panel lights reflect on the left forward canopy and the left forward flood light reflects on the right forward canopy. This should be easily remedied and the night lighting, as a whole, is considered superior to that in any other fighter flown. Inasmuch as the landing light has a separate switch for extension or retraction, it is recommended that a safety device be installed to automatically retract and turn off the landing light when the gear is retracted to prevent inadvertent failure of the pilot to operate the switch.

10. Engine Operation:

The engine operation during the Phase II tests was satisfactory except for two major deficiencies that should be corrected prior to delivery of the first production airplane. One deficiency is the engine surge bleed valves. These bleed valves shut at approximately 90 percent rpm, thus increasing the thrust appreciably. This would make it quite difficult to fly formation at these power settings. A study should be made to eliminate sudden thrust increases when the bleed valves shut. The other deficiency is the engine power control. The present engine power control is hydro-electrical and gives an extremely non-linear engine response to throttle movement. This results in a slow, sluggish engine response to throttle control application. Further, there is a compressor stall which occurs when accelerating the engine from low power settings at approximately 70 percent rpm. This is undesirable as, upon one occasion, a compressor stall resulted when adding power during the final approach.

11. General Aircraft and Systems Functions:

a. Emergency systems.

The emergency gear and speed brake system, although not actuated, appears to be satisfactory. The standard emergency flight control system, which is actually one-half of the normal system, also appears to be satisfactory inasmuch as there is little chance that both rotors in J-57 type engines would fail at one time. Further, in case of engine failure, it is very doubtful that, due to the weight and touchdown speeds of this aircraft, dead stick landings will be attempted except under the most ideal conditions. The additional air driven turbine provided as a third emergency hydraulic source in the YF-100A airplane will not be installed in the production airplanes. A few power-off landings were made on the dry lake surface, but it was quite difficult to judge the touchdown point, and the pilot would be reluctant to attempt a dead stick landing on a runway unless he had considerably more experience

in the airplane and a fairly long runway on which to make the attempt.

B. PERFORMANCE

1. Take-Offs:

Performance take-offs to determine the minimum distance required to take-off and climb to 50 feet were made at 25,000 lbs. take-off weight and a mid cg position of 31 percent MAC. Minimum distance take-offs were accomplished by using full throttle with afterburner, contractor's recommended trim setting, and full nose up stabilizer at the point of take-off. Several take-offs were made by applying full nose up stabilizer prior to the point at which take-off speed was reached. Due to the increased drag on the aircraft with the stabilizer deflected in the full nose up position the take-off distance was appreciably increased when this technique was used. Take-offs with afterburner off are not recommended because of the excessive distance required. On one occasion, without afterburner and on a 30°C day at 2300 feet altitude, the airplane went off the end of an 8,000 foot runway before becoming airborne. The following table is a summary of the average take-off data corrected to NACA standard day, no wind, sea level conditions.

TAKE-OFF PERFORMANCE

Gross Weight lbs	Ave. Ground Run feet	Ave. Total Distance to 50'	Ave. IAS at T.O. knots	Ave. IAS at 50' knots	S. L. V_t at T.O. knots	S. L. V_t at 50' knots	Ave. Gross Thrust at T.O. -lbs	Afterburner Operation
25,000*	2990	4390	146	-	142	165	11,500	on
25,000**	5270	--	147	-	143	-	7,700	off

* average of six take-offs

** one take-off only

2. Climbs:

Climbs were flown using maximum and military rated power for gross weight of 24,300 pounds at sea level and best climb speed. The oil cooler was set on automatic and the gap varied from 0° at low altitudes to 18.4° at 54,000 feet. Climb speeds used were those recommended by the contractor for optimum rate of climb. These data have been corrected to standard day atmospheric conditions and are presented in Figures 2 and 3. A summary of these data is presented in the following tables:

MAXIMUM POWER CLIMB PERFORMANCE
Afterburner On

Altitude feet	R/C ft/min	T/C min	N ₂ rpm	Fuel used - lbs	Naut. air miles travelled	Gross Wt - lbs	True Airspeed knots
S. L.	14,800	0*	9,510	0**	0	24,300	510
10,000	14,500	.7	9,510	300	6	24,000	520
20,000	13,900	1.4	9,510	575	12	23,725	535
30,000	10,700	2.2	9,350	825	20	23,475	540
40,000	6,600	3.35	9,195	1070	30	23,230	525
50,000	2,000	5.9	9,195	1380	52	22,920	520
53,800(SC)	100	10.6	9,195	1850	100	22,450	510

* Add 2.0 minutes to get time from brake release. (approx.)

** Add 800 lbs of fuel to account for start, taxi, take-off and accelerate to best climb speed. (approx.)

MILITARY POWER CLIMB PERFORMANCE
Afterburner Off

Altitude feet	R/C ft/min	T/C min	N ₂ rpm	Fuel used - lbs	Naut. air miles travelled	Gross Wt - lbs	True Airspeed knots
S. L.	4,300	0*	9,510	0**	0	24,650	425
10,000	4,000	2.3	9,510	300	19	24,350	455
20,000	3,500	5	9,410	550	40	24,100	475
30,000	2,250	8.4	9,180	800	67	23,850	492
40,000	650	15.9	9,070	1170	130	23,480	500
43,400(sc)	100	26	9,100	1525	220	23,075	500

*Add 3 minutes to get time from brake release. (approx.)

**Add 700 lbs of fuel to account for start, taxi, take-off and accelerate to best climb speed. (approx.)

3. Level Flight:

a. Level flight performance data were obtained at various altitudes for gross weights that would be obtained on check climbs at each respective altitude. Comparative performance data with two empty 275 gallon external fuel tanks attached were also obtained. Oil cooler operation was automatic and the gap varied from 3° to 18.5° open. All data have been corrected to an NACA standard day at the gross weights and altitudes presented in the following tables. True airspeed versus altitude data are presented in Figure 1 and are summarized in the following table:

TRUE SPEED -- KNOTS

Altitude feet	Gross Wt. lbs	TRUE SPEED AND MACH NUMBER Maximum Power			Military Power		
		V _t knots	Mach No.	N ₂ rpm	V _t knots	Mach No.	N ₂ rpm
S. L.*	24,300	634	.96	9,510	542	.82	9,510
1,500	24,200	631	.96	9,510	552	.84	9,510
11,500	24,000	615	.97	9,510	589.5	.93	9,510
27,000	23,550	609	1.02	9,508	580.5	.973	9,420
35,000	23,350	600	1.04	9,345	562	.977	9,185
41,500	23,150	586	1.015	9,305	533	.929	9,125
44,000	23,100	582	1.005	9,295	520	.905	9,115
49,500	22,900	562	.98	9,260	-	-----	-----

*Values extrapolated to sea level.

b. Curves of nautical air miles per pound of fuel versus Mach number are presented in Figures 4 and 5. The results of these data indicate that the best range of the aircraft can be realized by flying at as high an altitude as practicable. These results are summarized in the following table for recommended cruise settings. JP-4 fuel was used for all tests.

RECOMMENDED CRUISE PERFORMANCE

Altitude feet	Gross Wt. lbs	Mach No	CAS knots	TAS knots	N ₂ rpm	Nautical air mi. per lb. of fuel
11,500	24,500	.675	366	428	8,790	.14
27,000	23,550	.78	315	465	8,620	.22
27,000 *	23,550	.775	313	462	8,790	.188
41,500	23,150	.90	265	517	9,030	.253

*External tanks attached.

c. Power required data were obtained from all level flight test points and are presented in Figures 6 through 9. It is of interest to note that between 234 and 322 knots CAS less rpm is required to maintain level flight at 27,000 feet than is required at 11,500 feet. This same trend of less rpm at higher altitude for the same CAS was experienced in the XB-52 aircraft which is also powered by J-57 engines. The results of these tests are summarized in the following table:

POWER REQUIRED AT ALTITUDE

Altitude feet	Gross Wt. lbs	Max rpm N ₂	Mil rpm N ₂	Calibrated airspeed -- knots					
				Max rpm	Mil rpm	9200 rpm	9000 rpm	8800 rpm	8600 rpm
11,500	24,000	9,510	9,510	535	512	464	424	368	287
27,000	23,550	9,508	9,420	425	402	385	364	341	310
27,000*	23,550	9,490	9,400	405	389	369	347	314	-
41,500	23,150	9,305	9,125	306	276	-	257	-	-

*External tanks attached.

d. Drag data for non-afterburning operation were obtained by use of calibrated tail pipe pressure probes located aft of the turbine and forward of the afterburner. No drag data were obtained for afterburner operation. These data are presented in Figures 10 through 15 and are summarized in the following table:

AIRCRAFT DRAG SUMMARY

	Clean Aircraft	2 Empty 275 gal. tanks
Minimum drag--lbs.	1,690	2,080
Equivalent airspeed for Minimum drag -- knots	250	265
f--sq. feet at min. drag	8.67	--
C _{Dp} (at minimum C _L)	.012	--

4. Airspeed Calibrations:

No standard production airspeed system was installed in the test aircraft. A test nose boom in which both the total and static pressure sources were located, was attached to the nose section of the aircraft above the air intake as shown in Figure 25. Angle of sideslip and angle of attack vanes were also attached to the nose boom aft of the static source. The system was calibrated by means of tower fly-bys at 2300 feet altitude and airplane fly-bys past the AFFTC F-86 pacer aircraft, USAF No. 48-209 at 40,000 feet. Two pace points were also obtained by using the pacer aircraft. One point was obtained from the ground radar for a Mach number of 1.39 in a dive. These data are presented in Figure 25 as a plot of ΔV versus indicated mach number.

5. Landings:

Performance landings to determine the minimum runway distance required to land over a 50 foot obstacle were made at an average gross weight of 22,000 pounds. No actual test data were obtained over a 50 foot obstacle due

to the extremely low approach made on the test landings. The aircraft was below 50 feet before its height could be recorded. Maximum performance landings were made by extending the speed brakes and chute immediately after touchdown and then using maximum braking to stop. Normal performance landings were made utilizing the same procedure as outlined above except the drag chute was not used. The landing data have been corrected to standard day, sea level, no wind conditions and the results are presented as follows:

LANDING SUMMARY

Ave. Gr. Wt--lbs	RPM N ₂	Chute Deployed	IAS T.D.	S. L. V _t at T. D _r -knots	Ground Roll - ft.
22,000	idle	yes	140	131	2880
22,000	idle	no	140	131	5440

6. Engine Static Thrust Calibration;

a. The engine used during the test program was operated on the AFFTC thrust stand to calibrate the exhaust gas total pressure probes and to determine the static performance of the XJ57-P-7 engine. These data are presented in Figures 16 through 22 and are summarized in the following table for standard day conditions at sea level:

STATIC ENGINE PERFORMANCE

N ₂ rpm	N ₁ rpm	Gross Thrust lbs	Exhaust Gas. - Temp °C	Specific Fuel Consumption lbs/hr -- lb	Surge Bleed Valve Position
9,495*	5,750	10,200	565	2.2	Closed
9,495**	5,750	6,800	565	.91	Closed
9,400	5,580	6,250	540	.91	Closed
9,200	5,300	5,250	488	.915	Closed
9,000	5,030	4,480	440	.92	Closed
8,500	4,300	2,630	397	.975	Open
8,000	3,630	1,800	320	.98	Closed
7,500	3,100	1,300	270	1.02	Closed
7,000	2,720	980	258	1.14	Closed
6,500	2,430	730	255	1.39	Closed
6,000	2,160	520	255	1.73	Closed
5,640***	1,960	420	255	2.01	Closed

*A/B on - Max. power
 ** A/B off - Mil. power
 *** Idle rpm

b. Reference to the above table indicates a considerable disagreement with the Pratt and Whitney model specification (maximum thrust of 13,200 pounds and military thrust of 8,450 pounds). A large percentage of this loss is attributed to the extremely low efficiency of the sharp lipped air intake on the YF-100A while the aircraft is not in motion. When the aircraft is airborne the ram efficiency reaches approximately 96 percent at 450 knots CAS.

C. STABILITY AND CONTROL

1. Test Configuration:

a. The configurations used throughout the test program and referred to in this report are based upon those set down in USAF Specifications 1815-B and conform to the following table. RPM, altitude, gross weight and trim speeds that were used for each test are listed on the individual curves.

TEST CONFIGURATIONS

<u>Configurations</u>	<u>N₂ rpm</u>	<u>Trim airspeed</u>	<u>Landing Gear</u>
Take-off	max.	-----	Down
Power*	max.	level flight	Up
Cruise	level flight	as noted	Up
Power approach	level flight	1.4V _{s1} * *	Down
Landing	idle	1.4V _{s1}	Down

* All curves note whether afterburner is on or off.

** 1815 B states that the trim speed for the power approach configuration shall be 1.2V_{s1} but since the landing speed of the YF-100A is greater than 1.2V_{s1}, all tests in the power approach configuration were conducted with the airplane trimmed at approx. 1.4 V_{s1}.

b. All Phase II tests were conducted at one cg range. No control over cg location was exercised while in flight as the fuel selection system was entirely automatic. The cg at engine start was 31.9 percent MAC and this value varied as fuel was consumed to a most forward cg of 29.5 percent MAC and to an aft cg of 31.7 percent MAC.

2. Mechanical Characteristics of the Control System:

a. The flight control hydraulic system of the YF-100A aircraft powers both the stabilizer and the ailerons. Two completely independent systems, each of equal output are provided and operate in parallel. Should

either system fail the other has sufficient capacity to power the controls. The rudder operates from the regular utility hydraulic system. Should the utility system fail, the rudder may be operated manually since the air loads on the rudder may be overcome by normal pilot effort. In the event of engine failure the windmilling engine RPM is sufficient to provide adequate hydraulic pressure at airplane speeds down through landing. In the event the engine "seized" (both rotors locked) an emergency air turbine pump is provided which operates by ram air. This ram air turbine will not be included in the production airplanes.

b. Longitudinal control is obtained through movement of the horizontal stabilizer, hydraulically actuated by means of two individual sets of cables and push-pull rods connected to the stick. The stabilizer is operated by an irreversible hydraulic power control and artificial feel is provided by means of an artificial feel bungee. Forces in maneuvering flight are augmented by means of a two pound per "g" bobweight. The bobweight force varies from a maximum of 2.5 pounds per "g" at zero degrees attitude with full nose down stabilizer to a minimum of .12 pounds per "g" at 20° attitude with full nose up stabilizer. The two pounds per "g" figure quoted above is for the aircraft trimmed for cruise conditions at zero degrees attitude. A schematic drawing of the longitudinal control system is presented on page 10, Appendix II. There is no airspeed sensing device ("q" box) incorporated in the system; therefore, control forces are always the same for stick deflection from trim position in static flight. Longitudinal trim is obtained by means of moving the stabilizer through an electrical trim motor.

c. Lateral control is obtained by irreversible hydraulically operated ailerons actuated by push-pull rods and cables connected to the stick. The aileron control system consists of two mid-span ailerons on the trailing edge of each wing. Artificial feel is provided, as with the stabilizer, by an artificial feel bungee. During the Phase II tests, "in-flight" deflection of the ailerons was restricted to two-thirds stick throw above .6 Mach number due to the lack of structural integrity tests; however, some data were obtained at approximately 80 percent stick throw above .6 Mach number with no adverse results. No static balance is incorporated in the lateral control system. Lateral trim is maintained through an electric trim actuator which moves both the inboard and outboard ailerons on each wing by means of the hydraulic power unit. A schematic diagram of this system is found on page 11, Appendix II.

d. Directional control is maintained in a manner similar to the longitudinal and lateral systems. An irreversible hydraulic actuating cylinder moves the rudder and is controlled by cables and push-pull rods connected to the rudder pedals. Artificial feel is provided by a bungee and directional trim is maintained by an electric trim actuator which moves the rudder by means of actuating the hydraulic cylinder. A schematic drawing of the directional control system is illustrated on page 12, Appendix II.

e. During all Phase II tests it was found that all trim devices would reduce the control forces to zero and maintain settings.

f. Control friction was measured in a closed hangar at a temperature of 20 °C. Breakout forces were measured as that force required to start the control surface to move at any point over the entire range of a control system.

Both the rudder and the stabilizer break-out forces are excessive and above the required values of Specification 1815B. The aileron break-out forces are within the limits of 1815B and are satisfactory. Break-out forces in flight were checked on several occasions and though the in-flight forces are somewhat less than the static break-out force, the stabilizer and rudder break-out forces are still felt to be excessive. Plots of force versus control surface deflections are presented in Figures 28, 29 and 30 for the longitudinal, lateral and directional control systems respectively. The results of these tests are briefly summarized in the following table:

CONTROL FRICTION
Ground Test Data
20 °C Air Temperature

Control	Break-Out Force at Trim--lbs	Ave. Break-Out Force Over Control Range -- lbs	Allowable Limit 1815B--lbs
Stabilizer	8	10	4.5
Aileron	2	3	3
Rudder	22	15	10.5

3. Stabilizer Control Effectiveness:

a. Take-off tests were conducted from a concrete runway and from Rogers Dry Lake bed to determine the ability of the stabilizer to produce take-off attitude. No quantitative data were obtained but, using the contractor's recommended trim setting of 4° nose up and full nose up stabilizer at take-off, control is more than adequate at a weight of 25,000 pounds and a cg of 31 percent MAC. With the large amount of stabilizer control which is available, it is believed that stabilizer control effectiveness will be satisfactory for all conditions of loading and cg. Stick forces at the point of nose gear lift-off are light and estimated to be under the maximum of 35 pounds set forth in USAF Specification 1815B.

b. Landing tests were conducted to determine the ability of the longitudinal control to hold the aircraft off the runway until the maximum allowable ground angle was reached. Qualitative tests were conducted at gross weights of 21,000 to 22,000 pounds and approximately 30 percent MAC. Quantitative data recorded on Figure 31 for one landing indicates that the tail can be dragged at speeds of 136 knots or below at a cg of 30 percent MAC. As during the take-offs, adequate stabilizer is available at all times to produce the maximum ground angle. Care had to be exercised on all landings so as to insure that the tail was not dragged inadvertently prior to touchdown as was done on the above mentioned landing. It is estimated that sufficient control would be available for all practicable landing cg positions and gross weights. Though the control break-out forces are excessive and aircraft response to control movement is slow and sluggish on landings, the forces are light and are under the 1815B limit of 35 pounds pull force.

4. Stalls:

a. The stall characteristics are considered generally unsatisfactory. In the power approach configuration at 10,000 feet and power configuration at 35,000 feet with the speed brake out, there is an uncontrollable left roll-off at minimum airspeed. In the power configuration at 35,000 feet and 45,000 feet there is a slight pitching tendency 20 to 30 knots above the minimum flying speed and a rolling and yawing moment from about $1.25 V_S$ down to the minimum flying speed. The pitch-up is not violent and is easily controlled. The banking and yawing may be controlled by judicious use of the ailerons and rudders until the aircraft reaches the minimum flying speed, at which point, with the stabilizer full nose up, the aircraft "mushes" through several thousand feet of altitude with no apparent tendency to roll-off on either wing. Control is adequate during the approach to, and recovery from, the stall. Since the stall occurs 30 to 40 knots below the landing speed of the aircraft, it is felt that the poor stall characteristics are tolerable but should be remedied as soon as possible. No other unsatisfactory characteristics are apparent at high altitude or under accelerated conditions.

b. Time histories of the power approach configuration stalls and the power configuration stall at 45,000 feet are presented in Figures 32 through 34 and are summarized in the following table:

STALL CHARACTERISTICS

<u>Configuration</u>	<u>Altitude feet</u>	<u>Minimum IAS - knots</u>	<u>Type of and IAS for First Warning</u>
Power 21,500 lbs 30 percent MAC	45,000	115	Slight pitch-up -141 knots
Power Approach 21,200 lbs 30.5 percent MAC	8,000	105	Yawing tendency --131 knots

5. Static Longitudinal Stability:

a. Static longitudinal stability tests were conducted at 15,000 and 35,000 feet in the power configuration and 10,000 feet in the power approach configuration at an average cg of 31 percent MAC and an average gross weight of approximately 22,000 pounds. No neutral points were obtained since the tests were flown at one cg range only. Pilot technique involved trimming the aircraft at the desired airspeed and altitude and then varying the airspeed by use of the stabilizer alone without changing either the trim or throttle setting.

b. In the power configuration at both 15,000 and 35,000 feet, the static longitudinal characteristics are unsatisfactory. At 35,000 feet between .80 and .97 Mach number and at 15,000 feet between .86 and .99 Mach number

APPENDIX I

the aircraft exhibits either neutral or negative stability. A pull force of 7.5 pounds is required to maintain a trim speed of 323.5 knots at 35,000 feet when the afterburner is ignited, indicating a slight nose down pitching moment. However, this is not too objectionable as the forces are easily trimmed out. This trim change is graphically presented in Figure 35. Data obtained in the power approach configuration indicate no unsatisfactory characteristics. It is possible that, if a steeper force gradient could be obtained, the slow and sluggish low speed handling characteristics may be improved somewhat. Plots of stick force and stabilizer position versus calibrated airspeed are presented in Figures 35 through 37.

6. Dynamic Longitudinal Stability:

a. The dynamic longitudinal stability was investigated at 45,000 and 10,000 feet in the power configuration and at 10,000 feet in the power approach configuration at an average of 31 percent MAC. The testing technique involved making abrupt stabilizer deflection from trimmed level flight in order to obtain a positive acceleration of approximately 2.0 "g's" and a negative acceleration of approximately 0.0 "g". Tests were conducted with the controls free and with the pilot attempting to reduce the oscillations as rapidly as possible. No control fixed tests were made because, with the irreversible control system, the result would be the same as with controls free.

b. Test results indicate that at 45,000 feet the aircraft fails to satisfy the requirement of Specification 1815B (the short period normal acceleration oscillation shall be damped to $1/10$ amplitude in one cycle). With controls free the oscillation persists for as long as 7 to 8 seconds and 4 to 5 cycles. When an attempt is made to stop the oscillations, pilot technique is the determining factor as to the number of cycles required to damp out the normal acceleration oscillations. In one case the oscillations were damped to $1/10$ amplitude in two cycles by the pilot's judicious use of the controls. At 10,000 feet and 475 knots CAS, the dynamic longitudinal stability with controls free is satisfactory. When controls are used to damp the oscillations there may be a "g" overshoot, which in one case, was greater than the initial acceleration. If care is not exercised during a control displacement of this type, it is possible to cause a pilot induced oscillation similar to that experienced in the early F-86D model aircraft. In the power approach configuration the dynamic longitudinal stability is satisfactory. The normal acceleration oscillation damps in from one-half to one cycle. Under no conditions are there any undamped short period stabilizer oscillations.

c. These data appear as time histories in Figures 28 through 47 and are summarized in the following table:

DYNAMIC LONGITUDINAL STABILITY

<u>Configuration</u>	<u>CG %MAC</u>	<u>CAS knots</u>	<u>Alt. ft</u>	<u>Cycles to damp to 1/10 amplitude</u>	<u>Sec. to damp to 1/10 amplitude</u>
CONTROLS FREE					
Power(A/B on)	31	274	45,300	5	8
Power(A/B on)	31	274	45,300	4+	7+
Power(A/B off)	31.2	475	10,500	1.5	3.5
Power(A/B off)	31.2	475	10,500	2	3.5
Power approach	30.1	164	9,900	1/2	3.5
Power approach	30.1	164	9,900	1	4.5
Pilot Attempting to Reduce Oscillations as Rapidly as Possible					
Power(A/B on)	31	274	45,300	5	7.5
Power(A/B on)	31	274	45,300	2	3
Power(A/B off)	31.2	475	10,500	1.5	2.5
Power(A/B off)	31.2	475	10,500	2	3.5

7. Maneuvering Flight Characteristics:

a. Maneuvering flight characteristics were tested in the power and cruise configurations at 46,000 feet and at 12,000 feet in the power and power approach configurations. Tests were conducted by placing the aircraft in steady turns to obtain stabilized conditions of acceleration, stick force and speed while allowing the altitude to vary as little as possible.

b. Although the design limit load factor for this aircraft is 7.33 "g's", the contractor imposed a temporary restriction of 5.0 "g's" for all Phase II tests. With the exception of the power approach configuration the maneuvering characteristics are generally satisfactory in the range between stick force break-out and force lightening or pitch-up. The stabilizer position gradients are positive, up to pitch-up or force lightening, for all test conditions except the power approach configuration, thus indicating the stick fixed neutral points to be well aft of the cg at which the tests were conducted. After the initial break-out force was applied the forces were, in all cases, proportional to stabilizer deflection. At 45,000 feet, with the aircraft trimmed at 282 and 271 knots CAS, the stick force gradient is slightly higher than the 1815-F maximum of 8.85 pounds per "g". This is not excessive and it is felt that this condition is tolerable. For all other conditions in the power and cruise configurations the force gradient is less than the Specification maximum limit. In the power approach configuration the maneuvering characteristics are unsatisfactory. The stick fixed and stick free neutral points for the power approach configuration appear to be at approximately the cg at which the tests were conducted (31 percent MAC). The true force picture, in the approach configuration, was masked by the high break-out forces but the data indicate a force gradient approaching zero pounds per "g". These data are presented in Figures 48 through 53 and are

summarized in the following table:

MANEUVERING FLIGHT CHARACTERISTICS

<u>Configuration</u>	<u>CG</u>	<u>Ave. Force</u>	<u>Max. "g"</u>	<u>Max.</u>	<u>Min.</u>
		<u>gradient</u>	<u>attainable</u>	<u>1815-B</u>	<u>1815-B</u>
	<u>% MAC</u>	<u>lbs/"g"</u>	<u>"g"</u>	<u>lbs/"g"</u>	<u>lbs/"g"</u>
Power (A/B on) 282 knots CAS 45,800 feet 23,400 lbs	30.5	10.5	1.8	8.85	3.32
Power (A/B on) 271 knots CAS 47,000 feet 22,000 lbs	29.9	10.0	1.85	8.85	3.32
Cruise 254 knots CAS 45,900 feet 21,670 lbs	29.8	8.0	2.0	8.85	3.32
Cruise 237 knots CAS 45,700 feet 21,300 lbs	30.3	5.5	1.75	8.85	3.32
Power (A/B off) 484 knots 11,950 feet 23,000 lbs	31.7	4.3	5.0*	8.85	3.32
Power Approach 159 knots IAS 11,190 feet 21,900 lbs	31	0	1.5	No requirement	

*Flight test limit load factor

c. Additional data were obtained in the power configuration at 45,000 and 10,000 feet to demonstrate that the stick force gradient during quick pull-ups exceeded that obtained in steady turns. These data are presented in Figures 48 and 52, and as time histories in Figures 54 and 55, and show that Specification 1815-B requirements are satisfied.

d. Data were also obtained to determine the maximum "g" available in supersonic dives to approximately 1.07 Mach number at 35,000 feet. This test was accomplished by diving the aircraft from 40,000 feet and applying back stick in a diving turn at approximately 35,000 feet. The time histories of these tests, presented in Figures 56 and 57 indicate that a maximum of 6 "g's" are available at 349 knots IAS 1.0 Mach number, at 35,000 feet. However, an acceleration of this magnitude and speed cannot be held as there is a very rapid bleed-off in airspeed.

8. High Speed Flight Characteristics:

The high speed flight characteristics were investigated in dives from altitudes above 50,000 feet. The flying qualities of the YF-100A aircraft are excellent throughout the high speed range up to a Mach number of 1.39, the highest speed obtained. A time history of one dive to this Mach number is presented in Figures 58 and 59. A slight trim change, as shown by the change in stabilizer position, aileron position and rudder-force, was encountered at approximately 1.15 Mach number. This is not considered unsatisfactory as it was not noticed by the pilot at the time of the test. Data obtained from the AFFTC Radar-Phototheodolite are also plotted in Figure 58. These data show that the indicated Mach number of 1.39, which was obtained in this dive, was the same as the true Mach number determined from the true airspeed obtained from the radar data.

9. Speed Brake Operation:

a. Time histories of speed brake operations are presented in Figures 60 and 61. From trimmed, level flight at a calibrated airspeed of 590 knots at 3,500 feet, the speed drops to 370 knots in 200 seconds when the speed brake is open and the altitude held constant. At this speed, the time required for the brake to open fully is 13 seconds. In this period, the airspeed drops to 520 knots. The initial movement of the speed brake is accompanied by a nose down trim change. In the test shown in Figure 60, this trim change was at least 0.5 "g" but was easily controlled with less than 15 pounds of force. It is noted that, in this instance, the attempt to compensate for the trim change resulted in a longitudinal oscillation, because of the poor dynamic stability characteristics of the airplane, which produced a normal acceleration of 2.5 "g" even though the pilot was attempting to maintain level flight.

b. In the power approach configuration, the speed brake was opened at a stabilized speed of 160 knots at 4,300 feet. The airspeed was held constant and the resulting altitude loss was only 700 ft/min after the brake was extended. At this speed the speed brake extends fully in approximately 2.5 seconds. A slight nose down trim change occurs only after the speed brake reaches a half open position. The pull force required to compensate for the trim change is less than 10 pounds.

10. Static Directional Stability:

a. The static directional stability was investigated at 35,000 feet in the power configuration and at 10,000 feet in the power approach configuration. The airplane was placed in steady sideslips by deflecting the rudder while using sufficient lateral control and bank angle to hold a constant heading.

b. The sideslip characteristics are satisfactory. With the exception of the high rudder break-out forces (18 to 24 pounds), the sideslip angle was substantially proportional to the rudder pedal force and rudder deflection. In the power configuration 5.8 degrees of sideslip is attained at 57 pounds of rudder force. In the power approach configuration 11 degrees of left sideslip is attainable before full aileron is required to hold a constant heading.

c. The results of these tests appear in Figures 62 and 63 and briefly in the following table:

STATIC DIRECTIONAL STABILITY

Configuration CAS Alt Wt.	Stabilizer force at 50 lbs Rud. Force - lbs	Maximum sideslip angle degrees		Maximum rudder deflection degrees		Rudder force at Maximum deflection-lbs		L. outer aileron deflection at max. rudder degrees		
		<u>SIDESLIP</u>		<u>RUDDER</u>		<u>RUDDER</u>		<u>RUDDER</u>		
		Lt.	Rt.	Lt.	Rt.	Lt.	Rt.	Lt.	Rt.	
Power 281 knots 45,970 feet 21,700 lbs	5	--	--	5.7	18.5	--	57	--	--	2
Power approach 160 knots 11,620 feet 23,500 lbs	3	--	13	12	18	14	60	45	15	15

11. Dynamic Lateral Directional Stability:

a. Dynamic lateral directional stability tests were flown to determine the response of the airplane to either lateral or directional disturbances and to determine if there was any tendency for any oscillation of the rudder or aileron control surfaces. Tests were made at 45,000 feet and 10,000 feet in the power configuration and at 10,000 feet in the power approach configuration. Only control free tests were made as it was felt that results of control fixed tests would be the same because of the irreversible control system. Tests were first made by kicking the rudder abruptly to induce an oscillation. Oscillations were then induced by abruptly deflecting the ailerons. Both procedures were repeated again but with the pilot attempting to reduce the oscillations as rapidly as possible by using the controls.

b. Data obtained are presented in Figures 64 to 83. In all cases, except in the power approach configuration, the dynamic lateral directional stability is unsatisfactory. The time for the oscillations to damp to $1/2$ amplitude is excessive and there is a tendency for small amplitude oscillations to persist for 10 to 12 seconds. There is no indication of control surface oscillations. In the power approach configurations, the time to damp to $1/2$ amplitude is satisfactory and all oscillations appear to damp out in 8 to 10 seconds. The oscillations may be reduced more rapidly by use of the controls.

12. Adverse Yaw:

a. The sideslip developed during rudder fixed abrupt rolls from a steady banked turn was measured at 167 knots indicated airspeed at 10,000 feet. This was approximately sixty percent above the stalling speed for this configuration. With full aileron deflection (15°), the adverse yaw produced indicated values of sideslip of 19° to the left in left rolls and 11° to the right in right rolls. Rolls were then repeated using the rudder to keep the ball in the cockpit turn indicator centered. At this speed, 13° of rudder (20° available) is sufficient to keep the ball centered. Time history plots of these tests are presented in Figures 84 and 85. It should be noted that, when rolling with the ball in the center, the sideslip vane mounted on the nose boom gave indicated sideslip angles of as much as 6 to 8 degrees in the direction of the roll. It is apparent that, owing to the location of the vane and to the rotation of the airplane about its cg in a maneuver of this nature, at a high angle of attack, erroneous sideslip indications are obtained. Assuming that the error also exists in the rudder fixed abrupt aileron rolls, it is likely the actual sideslip developed as a result of the adverse yaw is only in the neighborhood of 8 to 10 degrees. Although these tests were run at a speed somewhat faster than the more critical speed of $1.4V_{SL}$ required by the specification, it is felt that directional stability and rudder power are both adequate to satisfactorily overcome the effects of the adverse yaw.

b. Similar tests were made in the power configuration at 245 knots ($2.0 V_S$) at 45,000 feet. Maximum aileron travel in these tests was limited to 13° because of a temporary restriction on the airplane. The data obtained are presented in Figures 86 and 87. Again it appears that the indicated values of sideslip are in error. At this speed only 5° of rudder is required to keep the ball in the cockpit turn indicator in the middle when using the rudder to offset the effects of the adverse yaw. The true sideslip resulting from the rudder fixed rolls is only 2 or 3 degrees.

13. Lateral Control:

a. Rate of roll tests were conducted at 45,000 feet in the power configurations (A/B on and off), at 10,000 feet in the power configurations (A/B on and off), and in the power approach configurations at 10,000 feet. These tests were repeated using rudder to assist the roll for all conditions except the power configuration with the afterburner on. All rolls were accomplished by starting from a 30 to 60 degree banked turn and then abruptly applying maximum allowable aileron deflection to roll out of the turn and continue through at least 360° (70 to 90 degrees in the power approach configuration). The rudder pedals were either held fixed at the trim position or rudder was applied

with the ailerons to coordinate the roll. As mentioned previously in the report, during all Phase II tests the aileron deflection was restricted to $\frac{2}{3}$ stick throw above .6 Mach number because of the lack of structural integrity tests on the aircraft.

b. Because the pilot was unable to use full aileron, a direct comparison to the maximum requirement of Specification 1815-B for the power configuration tests cannot be made. By extrapolation of the data available, it is felt that the rolling characteristics are generally satisfactory. At 45,000 feet, at maximum level flight speed (278 knots CAS), roll rates of 190 degrees per second can be achieved at approximately 80 percent aileron deflection. The tests at 10,000 feet in the power configuration indicate that at high indicated airspeeds (above 500 knots) the aileron effectiveness is decreased somewhat. With the aircraft trimmed at 462 knots roll rates of 230 degrees/sec can be obtained at approximately 80 percent aileron deflection while at 538 knots roll rates of 160 degrees/sec are all that are obtainable at the same aileron deflection. The rudder has little or no effect on the roll rate at either 45,000 or 10,000 feet in the power configuration. With the aircraft trimmed at approximately $1.4V_{SL}$ in the power approach configuration the roll rate is inadequate. For right rolls the helix angle obtainable is .06 while for left rolls the helix angle is only .038 as compared to the 1815-B requirement of .07. The use of rudder increases the roll rate by 10 to 30 percent but is still under the specified helix angle requirement of .07. In all cases, both in the power and power approach configurations the aileron forces are well under the specification requirement of 30 pounds stick force.

c. The aileron roll data are presented in summary form in Figures 88 and 89 and as aileron characteristics in Figures 90 through 92. Time histories of the roll data are presented in Figures 93 and 94. The data are summarized in the following table:

AILERON ROLLS

Configuration	Dir. of Roll	Rudder Pos. deg.	L. Outer ail. pos. deg.	Aileron force lbs.	Helix angle Pb/2 V radians	Roll Rate deg/sec
<u>RUDDER PEDALS FIXED</u>						
Power(A/B on) 45,000 ft. 278 knots CAS	Left	2 rt.	11.5 up	12	.047	145
	Right	2 rt.	12.5 dn	12.5	.063	192
Power(A/B off) 45,000 ft 245 knots CAS	Left	2 rt.	11.5 up	12	.051	140
	Right	2 rt.	13.5 dn	13	.066	175
Power(A/B on) 10,500 ft 538.5 knots CAS	Left	0	11.7 up	15	.043	145
	Right	0	10.3 dn	14	.04	134

(AILERON ROLLS con't)

<u>Configuration</u>	<u>Dir. of Roll</u>	<u>Rudder Pos. deg.</u>	<u>L. Outer ail. pos. deg.</u>	<u>Aileron force lbs.</u>	<u>Helix angle Pb/2V radians</u>	<u>Roll Rate deg/sec</u>
Power(A/B off)	Left	0	12.4 up	15	.08	233
10,500 ft	Right	0	11.5 dn	14	.076	218
462 knots CAS						
Power App- roach	Left	2 rt.	14 up	--	.037	40
10,400 ft	Right	2 rt.	15 dn	--	.061	65
167 knots CAS						

RUDDER USED TO ASSIST ROLL

Power(A/B off)	Left	3 lt.	1.2 up	--	.055	147
45,000 ft	Right	8 rt.	13.5 dn	--	.067	183
244 knots CAS						
Power(A/B off)	Left	4 lt.	12.4 up	--	.079	230
10,500 ft	Right	7 pt.	12.3 dn	--	.078	225
462 knots CAS						
Power App- roach	Left	12 lt.	14 up	--	.0495	53
10,500 ft	Right	14 rt.	15 dn	--	.067	70
167 knots CAS						

Note: Full static aileron deflection= 15° up and down

D. DATA ANALYSIS METHODS

1. Introduction:

This section briefly discusses the methods of data reduction that were used in analyzing the test data. The following references were used and will be referred to in the succeeding discussion.

- No. 1 "Flight Test Engineering Manual", USAF, Technical Report No. 6273.
- No. 2 "Performance Flight Testing Methods in Use by the Flight Section", USAF, Technical Report No. 5069.
- No. 3 "Pressure Altitude Method of Flight Test Data Reduction", AMC Memorandum Report No. TSFTE 2060.
- No. 4 "A Method of Determining Delta Rate of Climb for Turbo-Jet Powered Aircraft," AMC Memorandum Report No. MCRFT 2157.

- No. 5 "Flying Qualities of Piloted Airplanes", USAF Specification 1815-B.
- No. 6 "Performance Flight Testing Methods Jet Propelled Aircraft," USAF Technical Report No. 5239.
- No. 7 "Specification No. A-1639-B, Model XJ57-P-7 Engine," Pratt and Whitney Aircraft.
- No. 8 "Standardization of Take-Off Performance Measurement for Airplanes," AFFTC Technical Note R-12.

2. Notation:

The following notation will be used throughout the succeeding discussion of methods of data analysis.

<u>Symbol</u>	<u>Description</u>	<u>Units</u>
A/R	Aspect ratio	----
b	Wing span	feet
C _L	Lift coefficient	----
C _D	Drag coefficient	----
D	Airplane drag	pounds
dh/dt	Apparent rate of climb	ft/min
e	Airplane efficiency factor	----
F _{gt}	Test gross thrust	pounds
F _{gs}	Standard gross thrust	pounds
F _e	Air flow momentum or ram drag	pounds
F _n	Net thrust	pounds
F _g	Gross thrust	pounds
ΔF _n	Net thrust on a standard day at standard rpm minus net thrust on a test day	pounds
g	Acceleration due to gravity	ft/sec ²
Δh	Altimeter position error	feet
K	Temperature recovery factor	----
K _{tA_j}	Equivalent nozzle area	feet ²
L	Lift	pounds
M	Mach number	----
P _a	Ambient air pressure	in. Hg
P _{t2}	Compressor inlet total pressure	in. Hg

<u>Symbol</u>	<u>Description</u>	<u>Units</u>
P_{t7}	Total turbine impact pressure	in. Hg
q_c	Impact air pressure	in. Hg
R/C_{std}	Standard day rate of climb	ft/min
N_{2s}	Standard day high pressure compressor rpm	rpm
N_{2t}	Test day high pressure compressor rpm	rpm
R_{at}	$W_t (E_t + 50)/S_{at}$	pounds
S	Area of wing	feet ²
S_{gs}	Standard day sea level ground distance	feet
S_{gt}	Test day ground distance	feet
S_{as}	Standard day sea level air distance to 50 feet	feet
S_{at}	Test day air distance to 50 feet	feet
S_{ts}	Standard day, sea level total distance from start to 50 feet or from 50 feet to stop	feet
T_{as}	Standard day ambient air temperature	°C
T_{at}	Test day ambient air temperature	°C
T_{ic}	Indicated free air temperature	°C
T_{t2}	Compressor inlet total temperature	°C
t	Time from 50 feet to touchdown or from T. O. to 50 feet	seconds
V_{to}	Velocity at take-off	ft/sec
V_{50}	Velocity at 50 feet	ft/sec
V_{td}	Velocity at touchdown	ft/sec
V_w	Velocity of wind down runway	ft/sec
V_c	Calibrated airspeed	knots
V_e	Equivalent airspeed	knots
V_t	True airspeed	knots
ΔV	Airspeed position error	knots
W_s	Standard gross weight	pounds
W_t	Test gross weight	pounds
ΔW	Standard minus test gross weight	pounds
W_a	Airflow through jet engine	lbs/sec
ϵ	Pressure ratio($P_a/29.92$)	----
η_R	Ram or duct efficiency	percent
σ	Density ratio	----
θ	Temperature ratio(T_{at}/T_{as})	----

3. Take-Offs:

Take-offs were made to determine the minimum total distance required to clear a 50 foot obstacle at both maximum and military power. These data have been corrected to standard day, sea level, no-wind conditions at standard gross weight and power. These corrections were accomplished by use of the following relationships derived in Reference No. 8:

$$S_{gs} = S_{gt} \left[\frac{V_{to} + V_w}{V_{to}} \right]^2 \left(\frac{F_{gt}}{F_{gs}} \right)^{1.3} \left(\frac{W_{std}}{W_t} \right)^{2.3} \left(\frac{\sigma_t}{\sigma_s} \right)$$

$$S_{as} = (S_{at} - tV_w) \left(\frac{F_{gt}}{F_{gs}} \right)^{1.6} \left(\frac{W_{std}}{W_t} \right)^{2.3} \left(\frac{\sigma_t}{\sigma_s} \right)^{.7}$$

Since no tailpipe instrumentation was available for measurement of after-burner thrust, the net thrust augmentation ratio (F_{na}/F_n) was used to approximate the total thrust being delivered. This ratio was determined from data obtained on the static thrust run (Ref. 1, page 3-37). All maximum and military power test thrust data were corrected to standard day sea level conditions and plotted as thrust versus rpm to determine the average gross thrust at take-off for rated engine rpm. These data showed an average gross thrust of 11,500 pounds at 9,495 rpm at the point of take-off for maximum power operation. A gross thrust of 7,700 pounds at the same rpm was obtained for military power operations.

4. Climbs:

a. Climb tests were flown using military and maximum rated power. Prior to starting the Phase II tests the engine was set to produce an rpm which Pratt and Whitney had predetermined as being the rpm which would produce 8,450 pounds of static thrust at military power and 13,200 pounds of static thrust at maximum power in a ground test cell on a standard day at sea level. This (N_2) was 9,495 rpm. Since the fuel control unit of the XJ57-P-7 engine maintained a predetermined schedule of engine rpm (N_2) with respect to compressor inlet temperature, it was necessary to determine this schedule. These data appear in the form of a bias curve in Figure 26, N_2 being plotted versus compressor inlet temperature. All climb data have been corrected to standard day atmospheric conditions; a limit exhaust gas temperature of 570°C up to 25,000 feet, 610°C up to 40,000 feet and 640°C above 40,000 feet, or military rpm and the gross weight that would have been obtained on a standard day if the climb started at sea level and best climb speed.

b. These corrections were accomplished as follows:

$$(1) \quad R/C_{std} = dh/dt \sqrt{T_{at}/T_{as} + \frac{101.2 V_t \Delta F_n}{W_t}} \sqrt{\frac{T_{as}}{T_{at}}}$$

Thrust data for these corrections were obtained from Reference No. 7. The derivation of this method is outlined in Reference No. 4. Weight corrections were applied by use of the following equations which are presented in Reference No. 2.

$$\Delta R/C_1 = dh/dt \times \frac{\Delta W}{Wt} = \text{Rate of climb change for } \Delta W \text{ at constant power}$$

$$\Delta R/C_2 = \frac{19010}{V_c} \frac{\Delta W}{e b^2} = \text{Rate of climb change for induced drag change because of } \Delta W.$$

$$e = C_L^2 / (C_D - C_{DP}) \pi AR$$

(2) The exhaust gas temperature was corrected to the temperature corresponding to the standard day value of $T_{t7} \times 288/T_a$ in the case where this parameter was less than that for the limit exhaust gas temperature. For the case where the test ratio of $T_{t7} \times 288/T_a$ exceeded the maximum, it was necessary to correct the rpm. These corrections were accomplished in the following manner:

(a) A plot of $T_{t7} \times 288/T_a$ versus $N \sqrt{\frac{288}{T_a}}$ was constructed from the test data.

(3) For the case where $T_{t7} \times 288/T_a$ was less than the allowable limit, the ratio was increased or decreased to a value corresponding to the standard day value of $N_{2s} \times \sqrt{288/T_a}$.

(4) For the case where $T_{t7} \times 288/T_{at}$ exceeded the allowable limit, the parameter $N \sqrt{288/T_a}$ was decreased to a value corresponding to the limit standard day value of $T_{t7} \times 288/T_{as}$. The rate of climb change resulting from the change in thrust for this rpm correction was applied using the thrust data from Reference No. 7 and the equation:

$$\Delta R/C = \frac{101.2 V_t \Delta F_n}{Wt} \sqrt{T_{as} / T_{at}}$$

which has already been discussed.

5. Level Flight:

Level flight data were obtained by flying each successive test point at an altitude selected to give a constant ratio of gross weight to atmospheric pressure, to provide a minimum weight correction for the basic data. All weight pressure ratios were preselected to represent approximately the values which would be obtained on a standard day by climbing to the selected altitude from a sea level gross weight of 24,300 pounds. The methods of reduction are outlined in Reference No. 1. The equations used were as follows:

$$N_{2s} = \left[N_{2t} + \frac{(\Delta N \sqrt{288/T_a})}{\Delta W/\delta} \Delta (W/\delta) \right] \sqrt{T_{as}/T_{at}}$$

$$\frac{\Delta (N \sqrt{288/T_a})}{\Delta W/\delta} = \text{Weight correction factor obtained from test data plots similar to Figure 6 at test Mach number.}$$

To determine military or maximum rated rpm in level flight it was necessary to estimate from the test data the maximum attainable Mach number under the desired conditions and then to determine the standard day compressor inlet temperature which in turn determined the standard day military rpm by use of the rpm bias curve presented in Figure 26.

6. Thrust and Drag:

Thrust required and drag data were obtained during the level flight calibrations. These data were obtained from the total pressure and temperature pick-ups installed in the tail pipe aft of the turbine and forward of the afterburner. Thrust required and drag data for afterburner operation were obtained from the augmentation ratios as described in Paragraph 3 of this discussion. The pressure probes were calibrated during the static thrust tests conducted on the Edwards Air Force Base thrust stand. The calibration data have been presented in Figures 16 through 23. The calibration data have been extrapolated to include all conditions of flight by use of the ideal gas flow equations which are derived in Reference No. 6 and presented briefly as follows:

$$\text{Subsonic: } F_g/P_a = 570.06 (K_t A_j) \left[\left(\frac{P_{t7}}{P_a} \right)^{0.2481} - 1 \right] \text{ for } \frac{P_{t7}}{P_a} < 1.85$$

$$\text{Supersonic: } F_g/P_a = 70.727 (K_t A_j) \left[1.26 \frac{P_{t7}}{P_a} - 1 \right] \text{ for } \frac{P_{t7}}{P_a} > 1.85$$

No airflow data were obtained. The Pratt and Whitney Division recommended method for determination of airflow was utilized. The curve used for this determination is presented in Figure 27, Appendix I. To obtain the net thrust of the engine installation, the ram drag was subtracted from the gross thrust.

$$F_n = F_g - F_e$$

$$F_e = 0.0524 W_a V_t$$

7. Lift and Drag Coefficients:

The lift and drag coefficients for incompressible flow were calculated from the following equations:

$$C_L = \frac{295 W_t}{V_e^2 S}$$

$$C_D = \frac{295 F_n}{V_e^2 S}$$

8. Airspeed Calibration:

The test boom airspeed system with a North American Aviation Inc. manufactured head was used throughout all tests and was calibrated in the clean configuration only. Calibrations up to approximately .98 Mach number were made by use of the Air Force Flight Test Center F-86A pacer, USAF No. 48-209. Calibration points above .9 Mach number were obtained by making airplane and tower fly-bys and using the altimeter method of airspeed calibration. Supersonic data were obtained in a dive and calibrated against a radar computed plot of the dive. The data and a schematic diagram of the nose boom installation are presented in Figure 23. The airspeed calibration curve is presented as airspeed correction to be added (ΔV) versus indicated Mach number. It is realized that this is a rather unorthodox presentation but by plotting the correction in this manner the data approached a common correction curve for all altitudes, for Mach numbers above .88.

9. Altimeter Correction:

The altimeter corrections for position error were computed from the airspeed calibrations data by use of the following expression derived in Reference No. 1.

$$\frac{\Delta h}{\Delta V} = \frac{V_c \left[\frac{1 + V_c}{2,183,944} \right]^{2.5}}{11,282 \sigma}$$

10. Temperature Recovery Factor:

The variations of indicated free air temperature with airspeed were obtained from the pacer airspeed calibration data. Test data indicated that 100 percent adiabatic temperature rise was obtained with the thermocouple type bulb located on the under side of the fuselage 95 inches to the rear of the nose air intake. The expression for determining the percentage of adiabatic temperature rise is developed in Reference No. 6 and is as follows:

$$K = \left(\frac{T_{ic} - 1}{T_{at}} \right)^{5/M^2}$$

11. Landings:

Normal and maximum performance landings were conducted to determine the minimum distance from a 50 foot obstacle to stop under different conditions of

operation. The data obtained have been corrected to standard day, sea level, no wind conditions by the following expressions which are presented in Reference No. 2:

$$S_{ts} = S_{as} + S_{gs}$$

$$S_{as} = S_{at} + V_{wt}$$

$$S_{gs} = S_{gt} \left(\frac{V_{td} + V_w}{V_{td}} \right)^{1.85} \sigma$$

12. Stability and Control:

a. Stability and control data were obtained to evaluate the handling characteristics of the aircraft at some of the conditions where the compliance with USAF Specification was estimated to be marginal or unsatisfactory. These data are presented in Figures 28 through 94 of Appendix I. Certain limitations of instrumentations and simplifying assumption for data analysis were tolerated in the analysis of these data and are worth noting:

b. Instrumentation was not dynamically balanced.

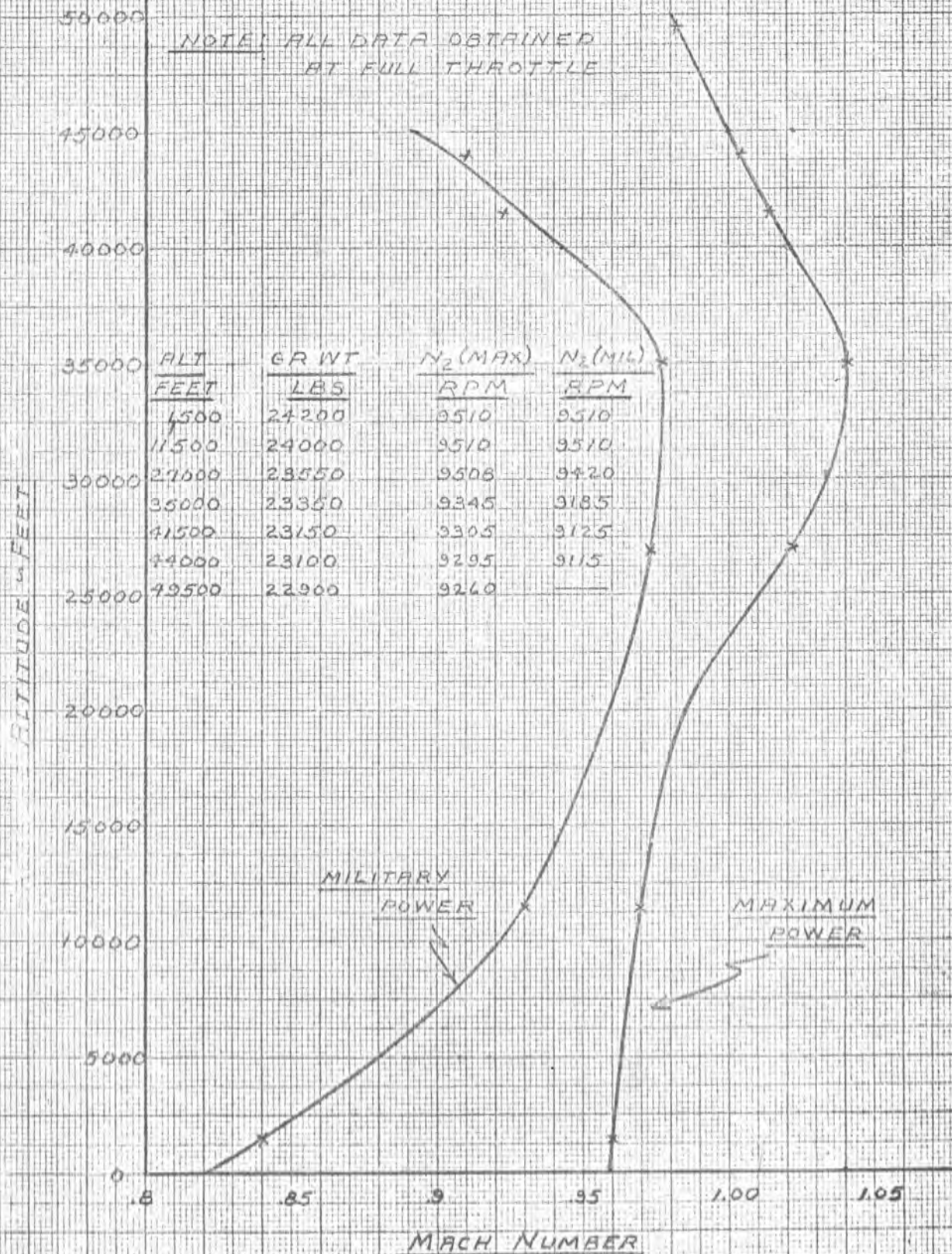
c. Aileron roll rates were obtained from a Schwein rate gyro and the peak values obtained are presented. Since only the left aileron was instrumented, the right aileron was assumed to have the same deflection but in an opposite direction.

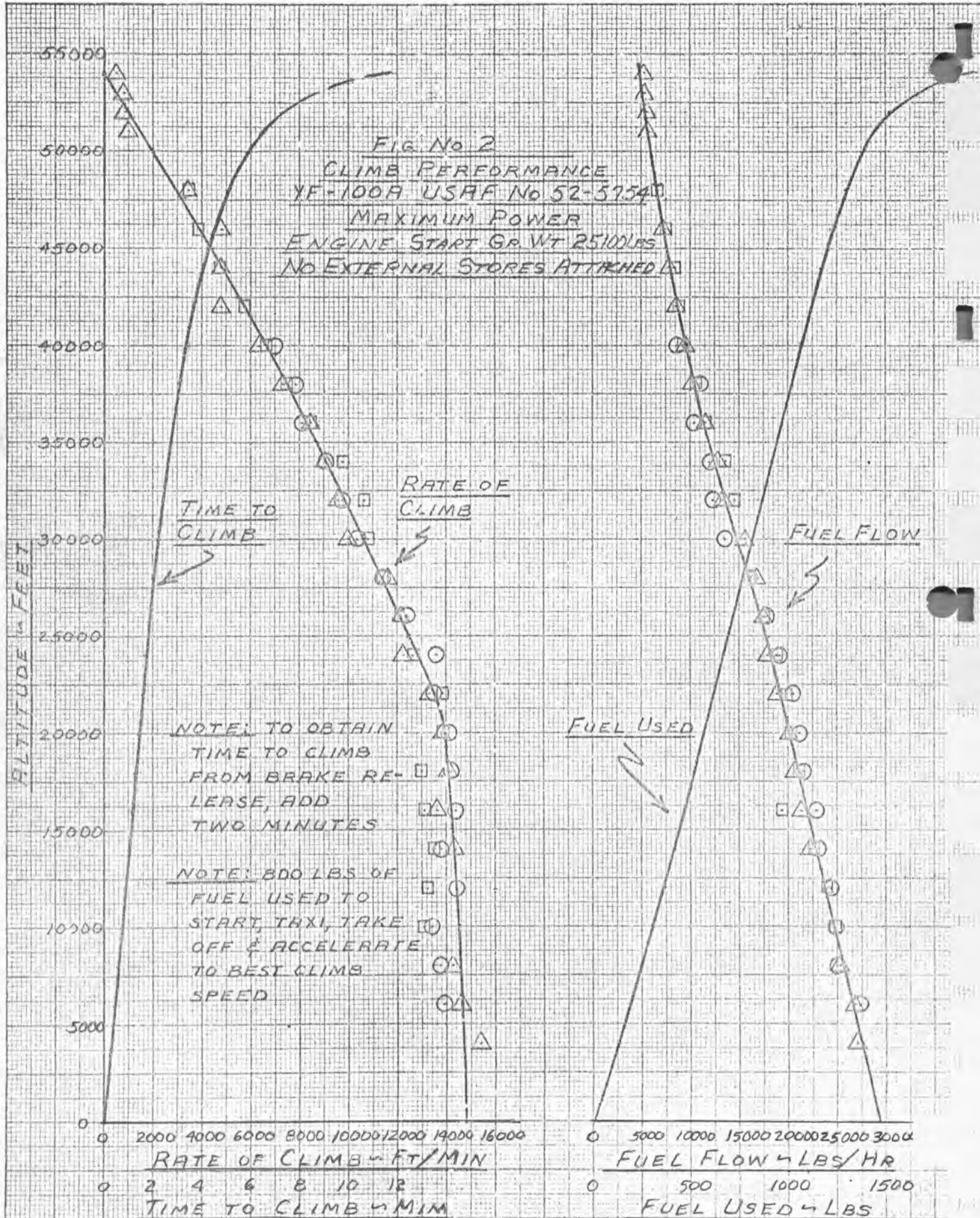
d. Stick force gradients have been calculated from the slopes of the stick force versus normal acceleration curves. These gradients have been used to compare test results with the requirements presented in Specification 1815-B. It has been observed that this procedure was not in strict accordance with the method outlined in 1815-B, which defines the gradient as the ratio of the force to the "g" at the test point. The 1815-B criteria were used for comparison since they generally represented the desired condition.

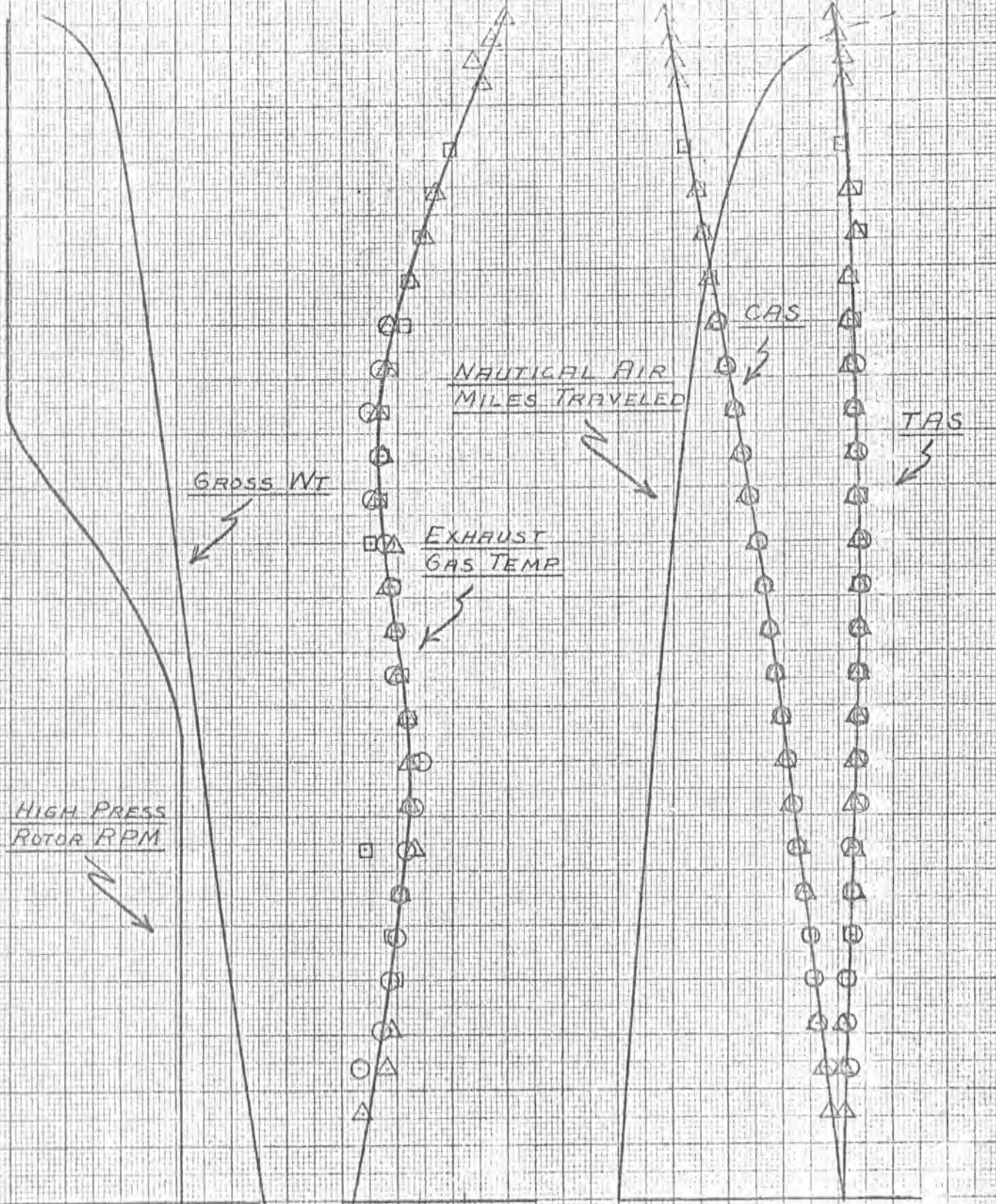
FIG No 1

MACH NUMBER VS ALTITUDE
 YF-100A USAF No. 52-5754
 ENGINE START GR WT. 25100LBS
 NO EXTERNAL STORES ATTACHED

NOTE: ALL DATA OBTAINED
 AT FULL THROTTLE



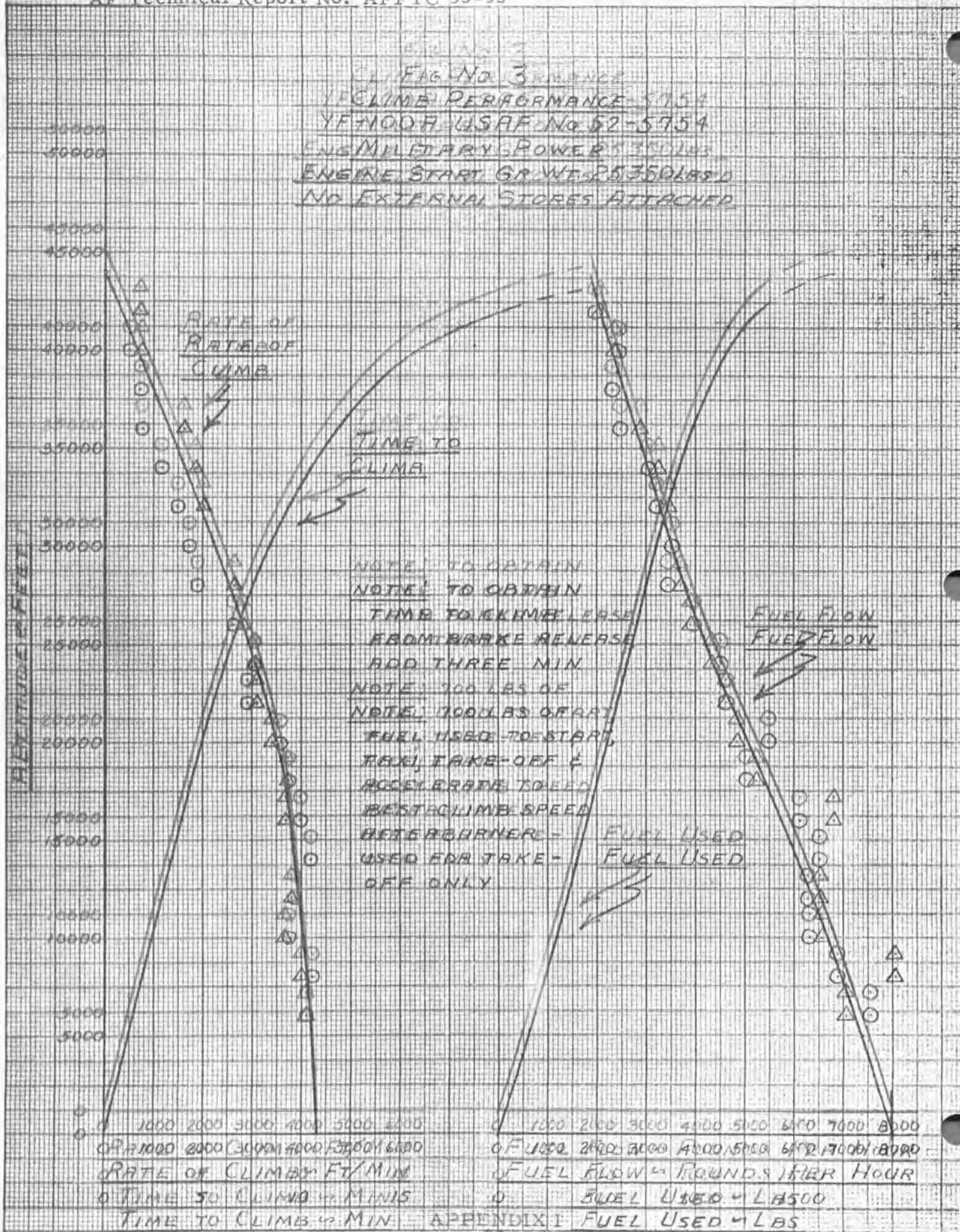




22000	23000	24000	500	550	600	650	100	200	300	400	500	600
<u>GROSS WEIGHT - LBS</u>			<u>EXHAUST GAS TEMP</u>				<u>AIR SPEED - KNOTS</u>					
9200	9300	9400	9500	9600			0	20	40	60	80	100
<u>№ of RPM</u>							<u>NAUTICAL AIR MI TRAVELED</u>					

APPENDIX I

CLIFF No 3
 YF CLIMB PERFORMANCE 5754
 YF 1100H USRF No 52-5754
 ENGINE MILITARY POWER 3300LBS
 ENGINE START GR WT 25350LBS
 NO EXTERNAL STORES ATTACHED



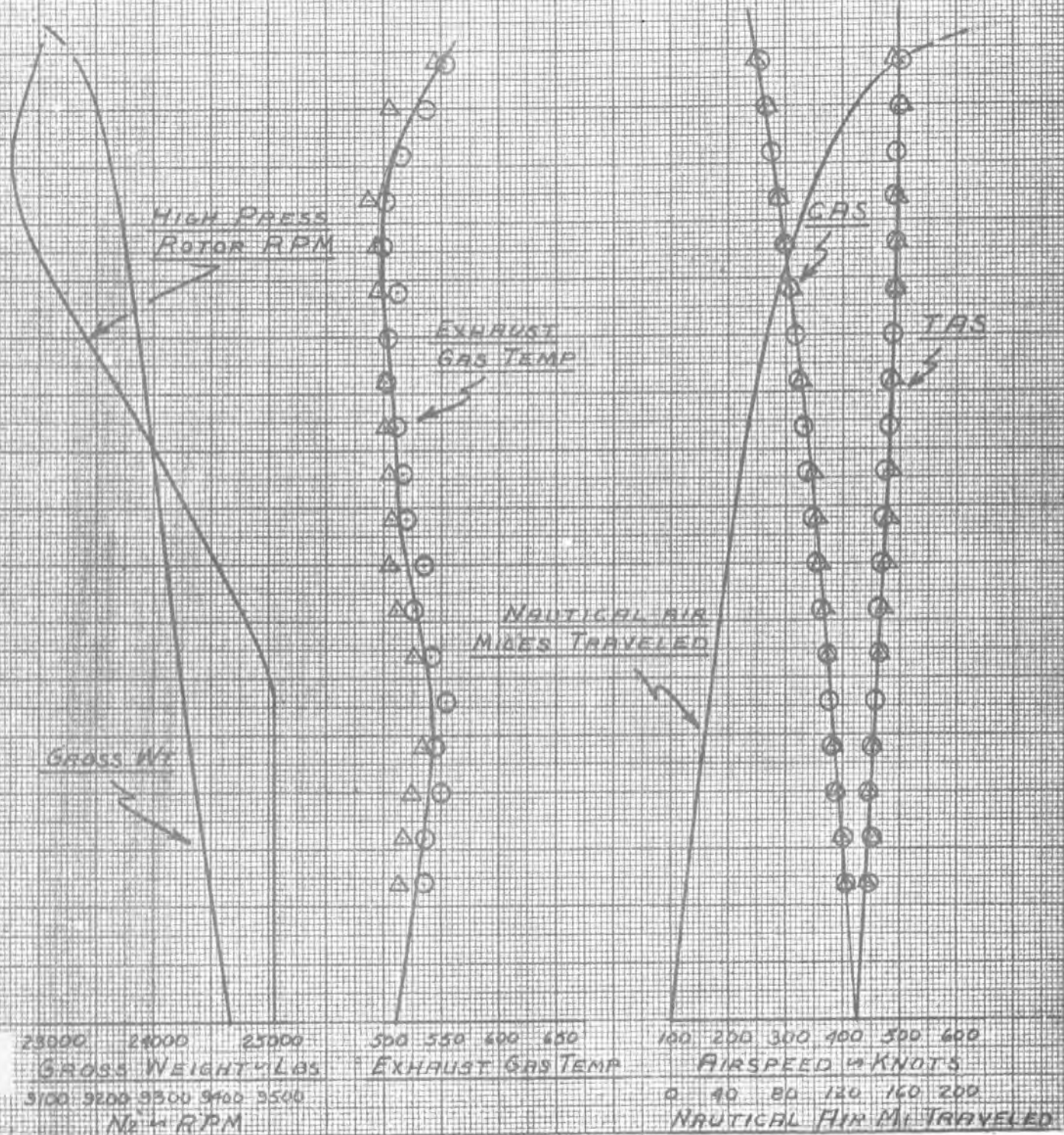


FIG No 4
NAUTICAL AIR MILES PER POUND OF FUEL
YF-100A USAF No 52-5754
JP-4 FUEL
ENGINE START GR. WT. 25100 LBS.
NO EXTERNAL STORES ATTACHED.

<u>SYM</u>	<u>ALT</u>	<u>GROSS WT</u>
	<u>FEET</u>	<u>LBS</u>
▽	1500	24200
○	11500	24000
△	27000	23550
□	35000	23350
◇	41500	23150
▽	49000	23100
△	49500	22900

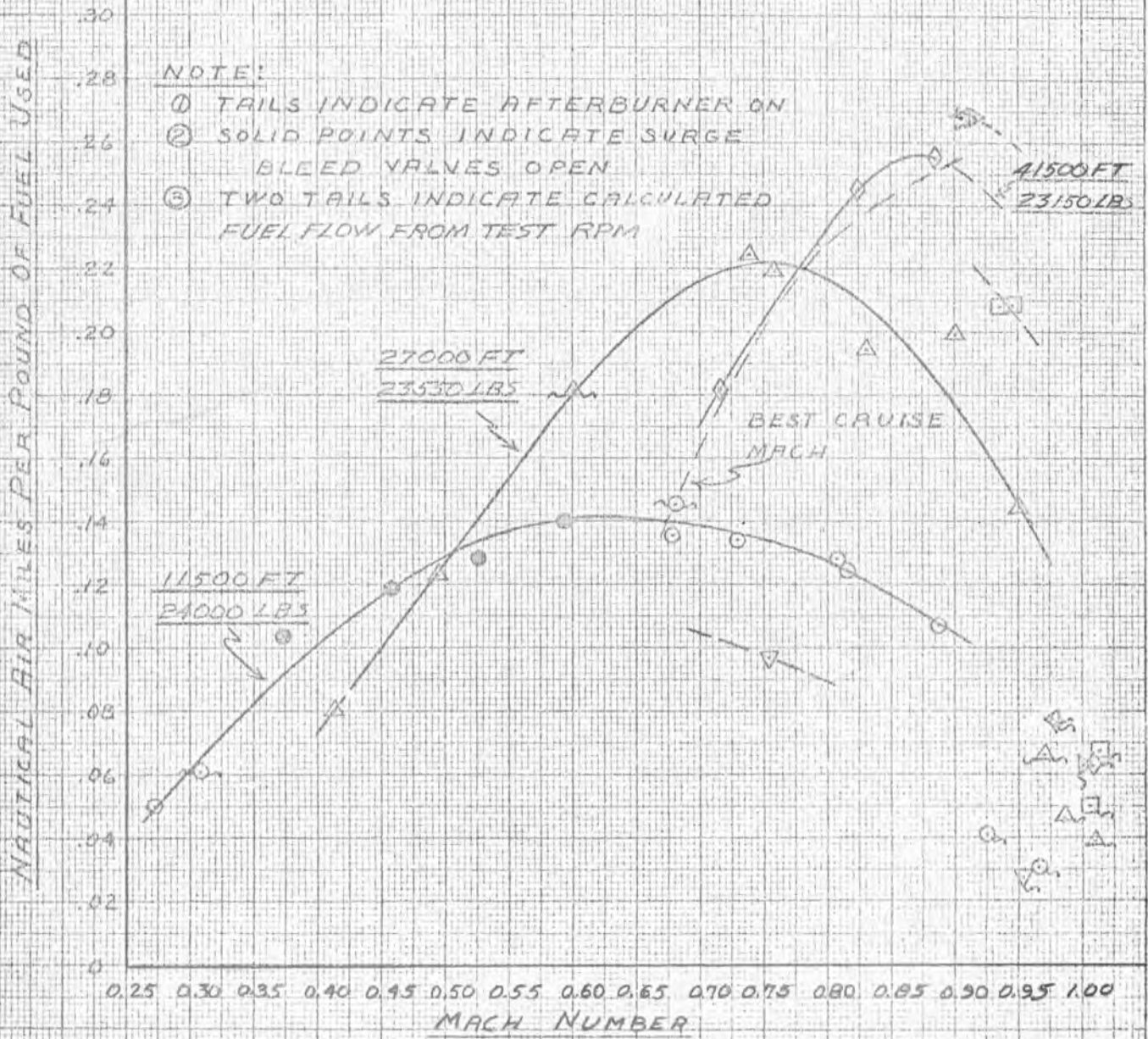


FIG No 5

NAUTICAL AIR MILES PER POUND OF FUEL

YF-100A USAF No 52-5754

JP-4 FUEL

ENGINE START GR. WT 25100LBS

COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE

SYM	ALT FEET	GROSS WT LBS	CONFIG
△	27000	23550	CLEAN
▲	27000	23550	2-275 GAL TANKS

NOTE:

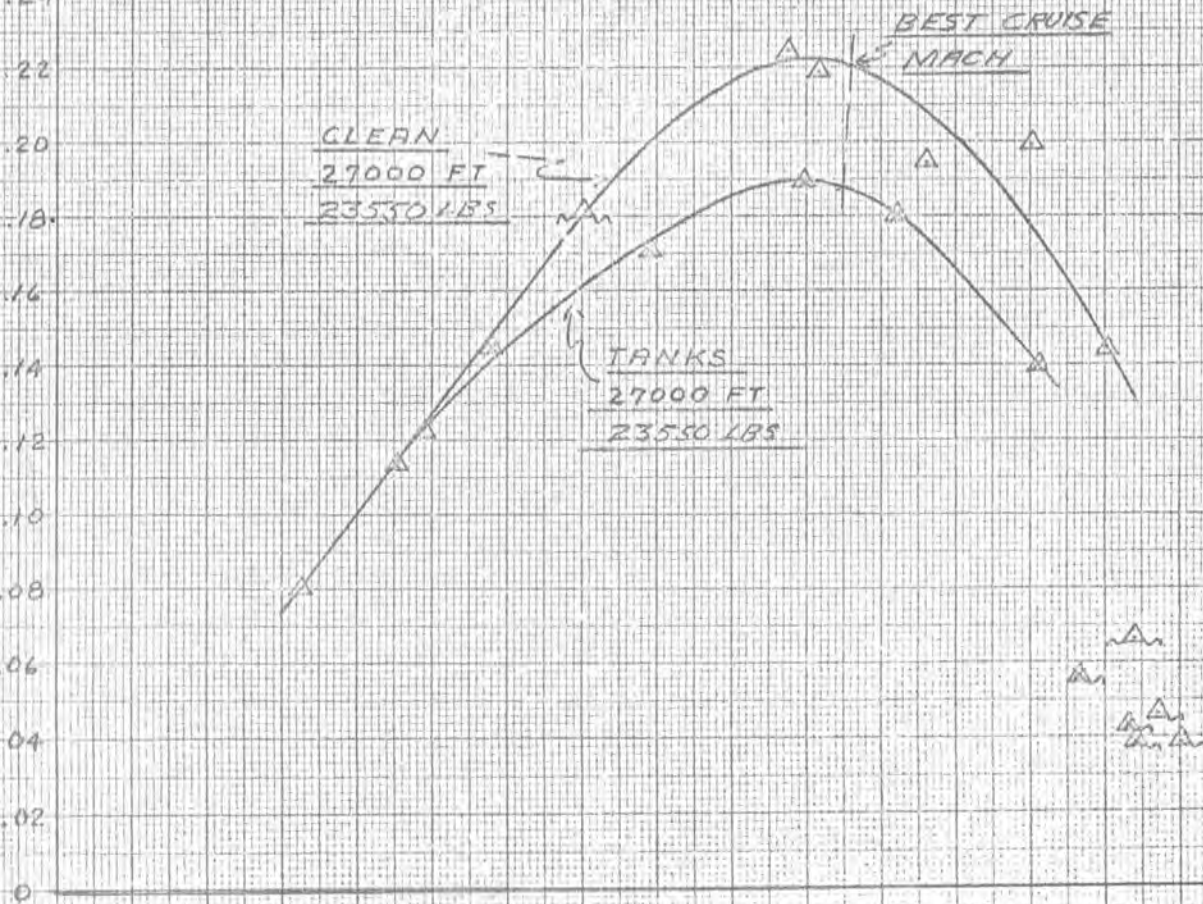
- ① TAILS INDICATE AFTERBURNER ON
- ② TWO TAILS INDICATE CALCULATED FUEL FLOW FROM TEST RPM

NAUTICAL AIR MILES PER POUND OF FUEL USED

0
.02
.04
.06
.08
.10
.12
.14
.16
.18
.20
.22
.24
.26
.28
.30

0.25 0.30 0.35 0.40 0.45 0.50 0.55 0.60 0.65 0.70 0.75 0.80 0.85 0.90 0.95 1.0

MACH NUMBER



F-5 No 6
 N_2 TV2BR/ T_0 VS MACH NO
 YF-100A USAF No 52-5754
 ENGINE START GR. WT 25100 LBS.
 NO EXTERNAL STORES ATTACHED

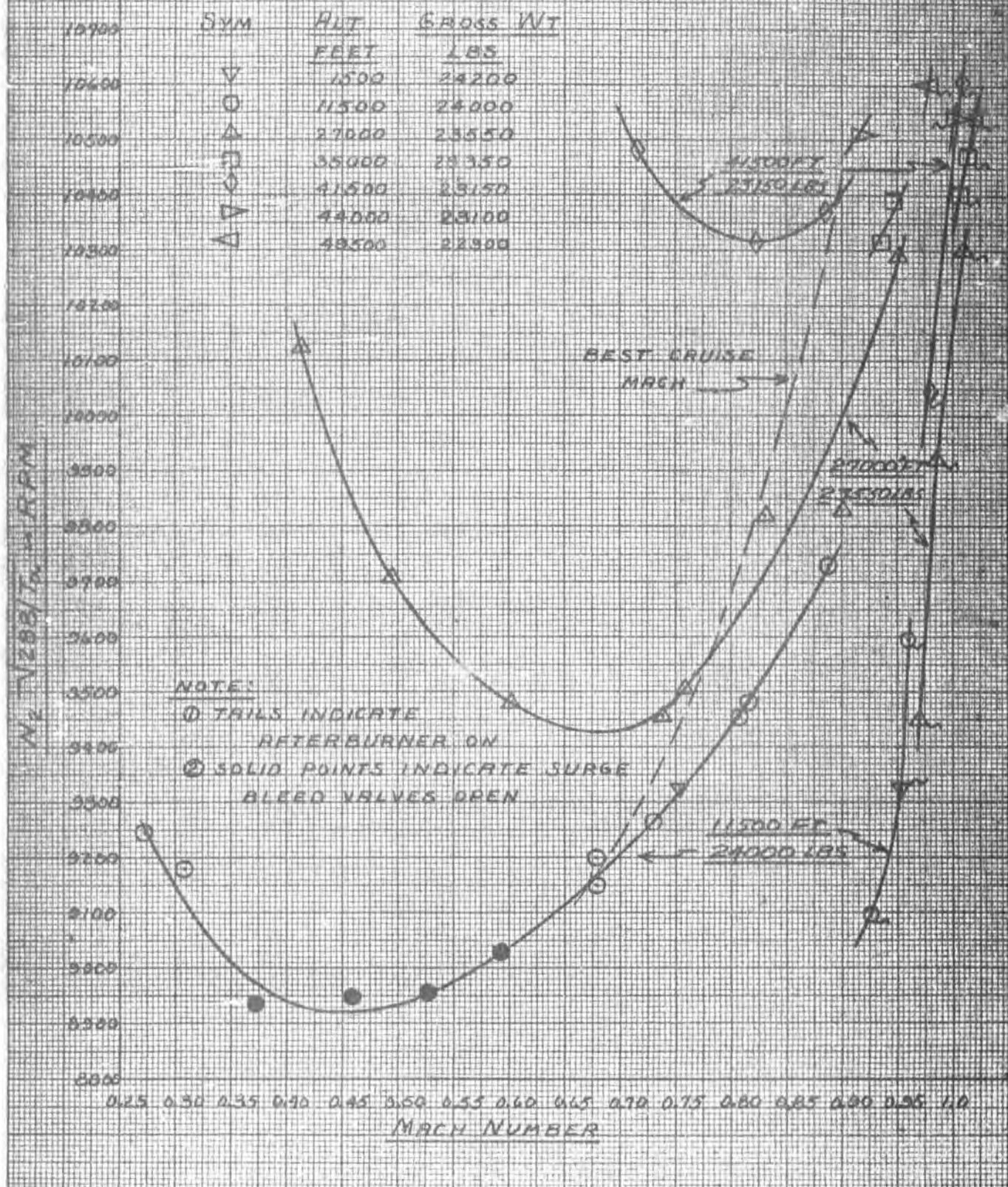


FIG No 7

$N_2 \sqrt{288/T_0}$ vs MACH No
 YF-100A USAF No 52-5754
 ENGINE START GR. WT. 25100 LBS

COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE

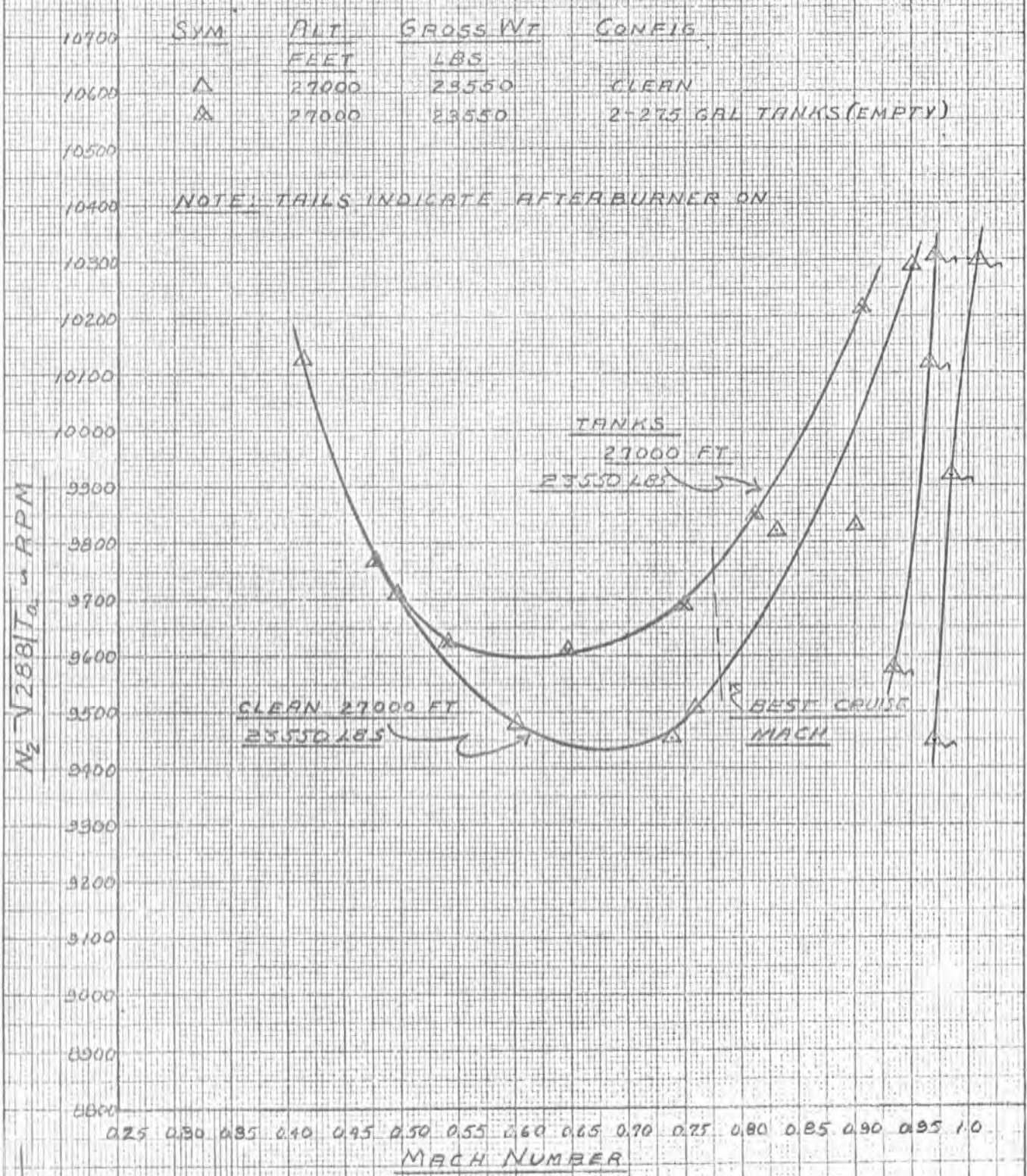


Fig No 8

RPM vs CALIBRATED AIRSPEED

YF-100 USAF No 52-5754

ENGINE START GR WT 25100LBS

NO EXTERNAL STORES ATTACHED

SYM	ALT	GROSS WT
	FEET	LBS
▽	1500	24200
○	11500	24000
△	27000	23550
□	35000	23350
◇	41500	23150
▽	44000	23100
△	49500	22900

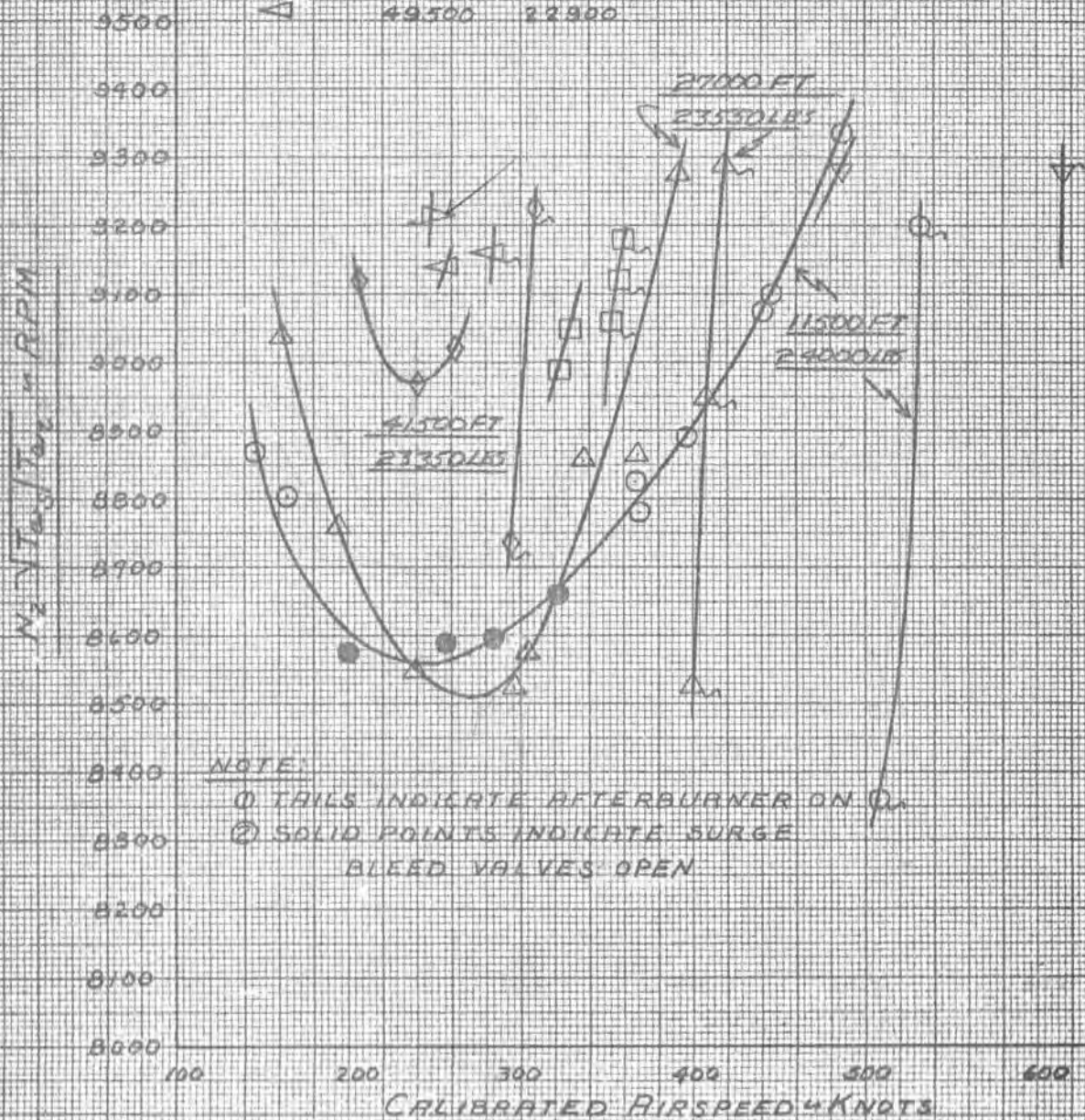


FIG No 9

RPM vs CALIBRATED AIRSPEED

YF-100A USAF No 52-5754

ENGINE START GROSS WT 25100 LBS

COMPARISON OF TANKS ON AND TANKS OFF PERFORMANCE

SYM	ALT FEET	GROSS WT LBS	CONFIG
△	27000	23550	CLEAN
△	27000	23550	2-275 GAL TANKS (EMPTY)

NOTE: TAILS INDICATE AFTERBURNER ON

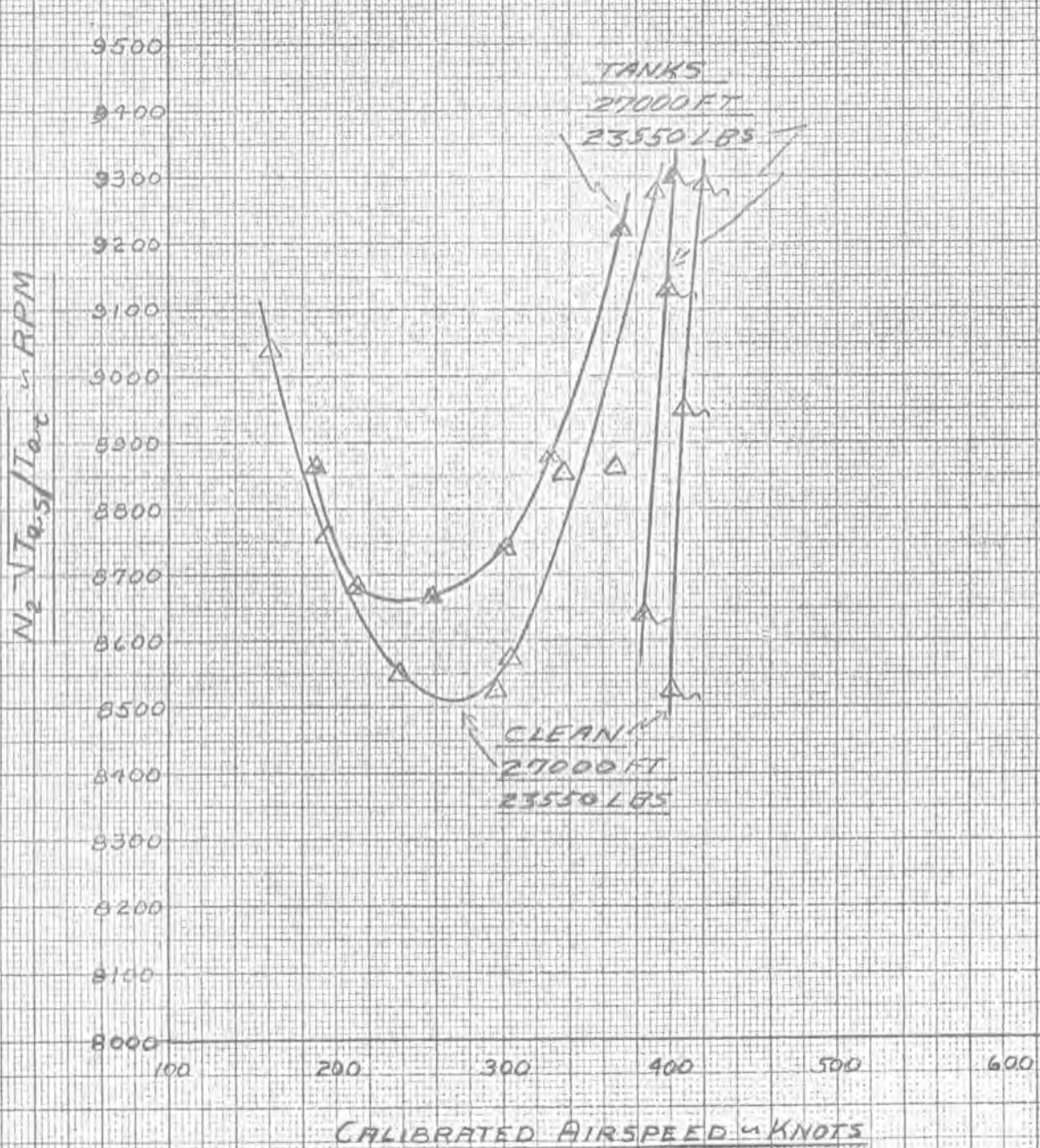


FIG No 10

LIFT COEFFICIENT

VS

DRAG COEFFICIENT

YF100A USAF NO52-5754

NO EXTERNAL STORES ATTACHED

SYM	ALT FEET	AVG WEIGHT LBS	TEST W/S
◇	41500	23150	134300
□	35000	23350	99200
△	27000	23550	69400
○	11500	24000	37000
▽	1500	24200	25520

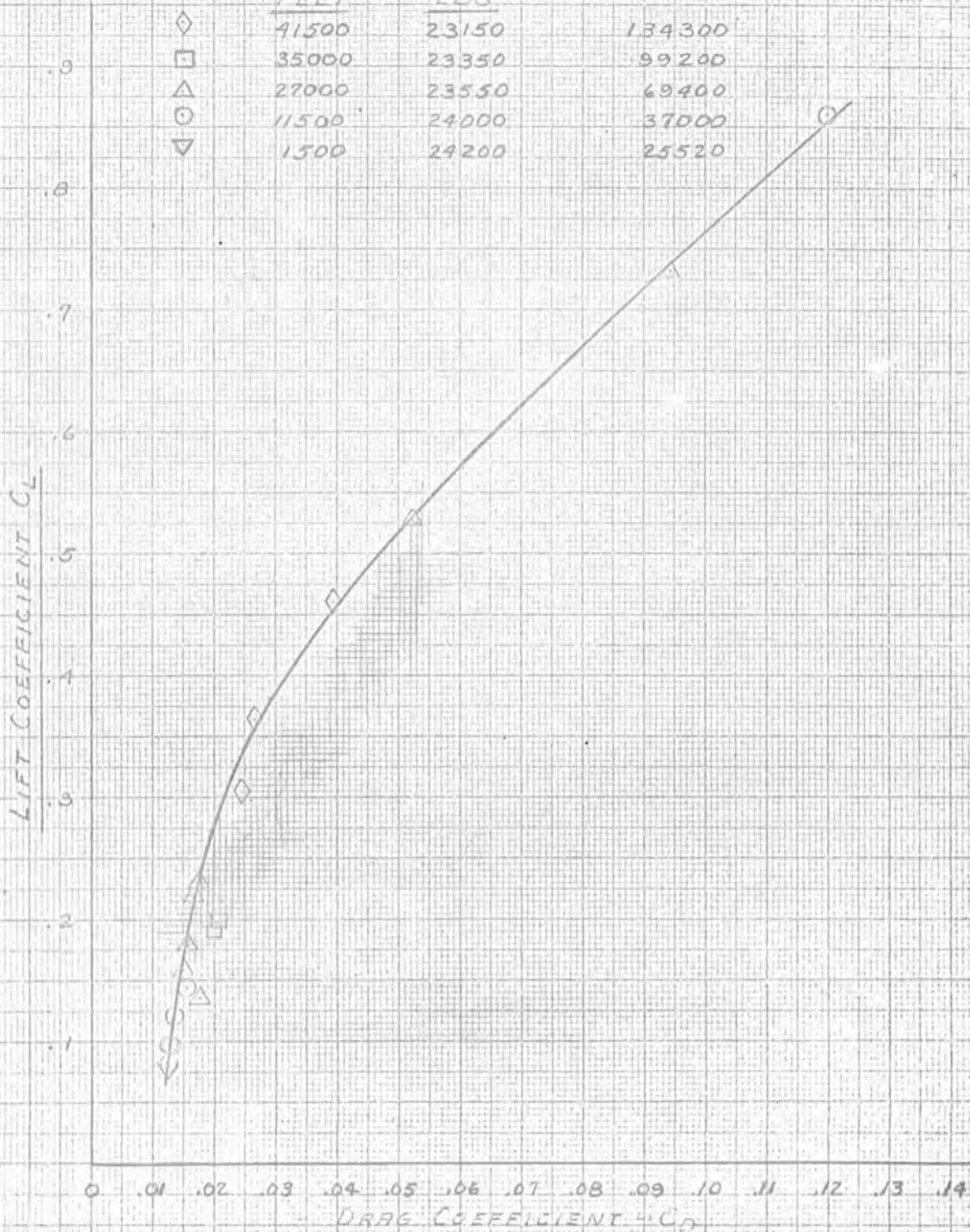
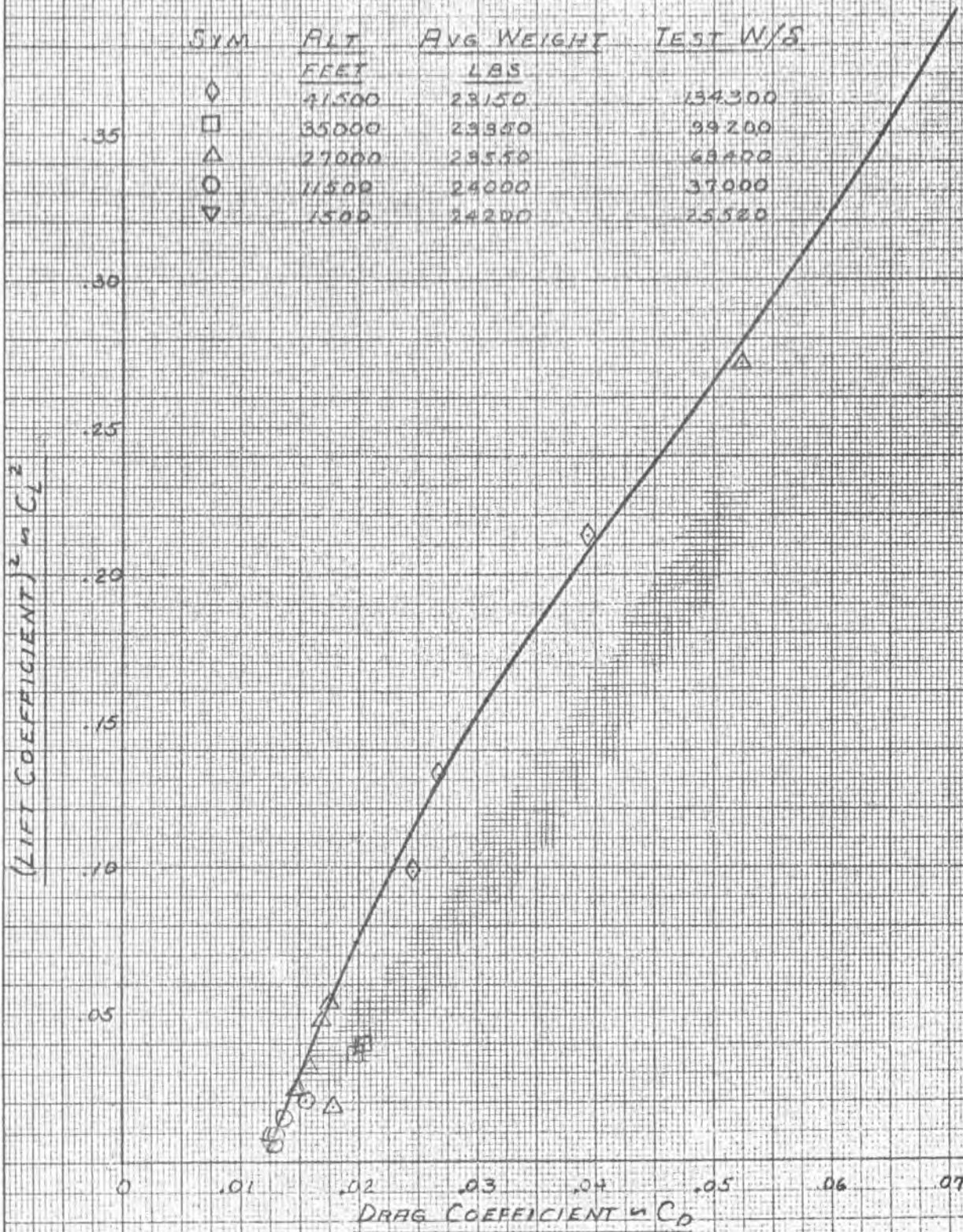


FIG No 11
 C_L^2 VERSUS C_D
 YF-100A USAF No 52-5754
 NO EXTERNAL STORES ATTACHED



FIS No 12

LIFT COEFFICIENT

VS

DRAG COEFFICIENT

YF-100A USBF No 52-5754

DRAG COMPARISON WITH TANKS ON AND TANKS OFF

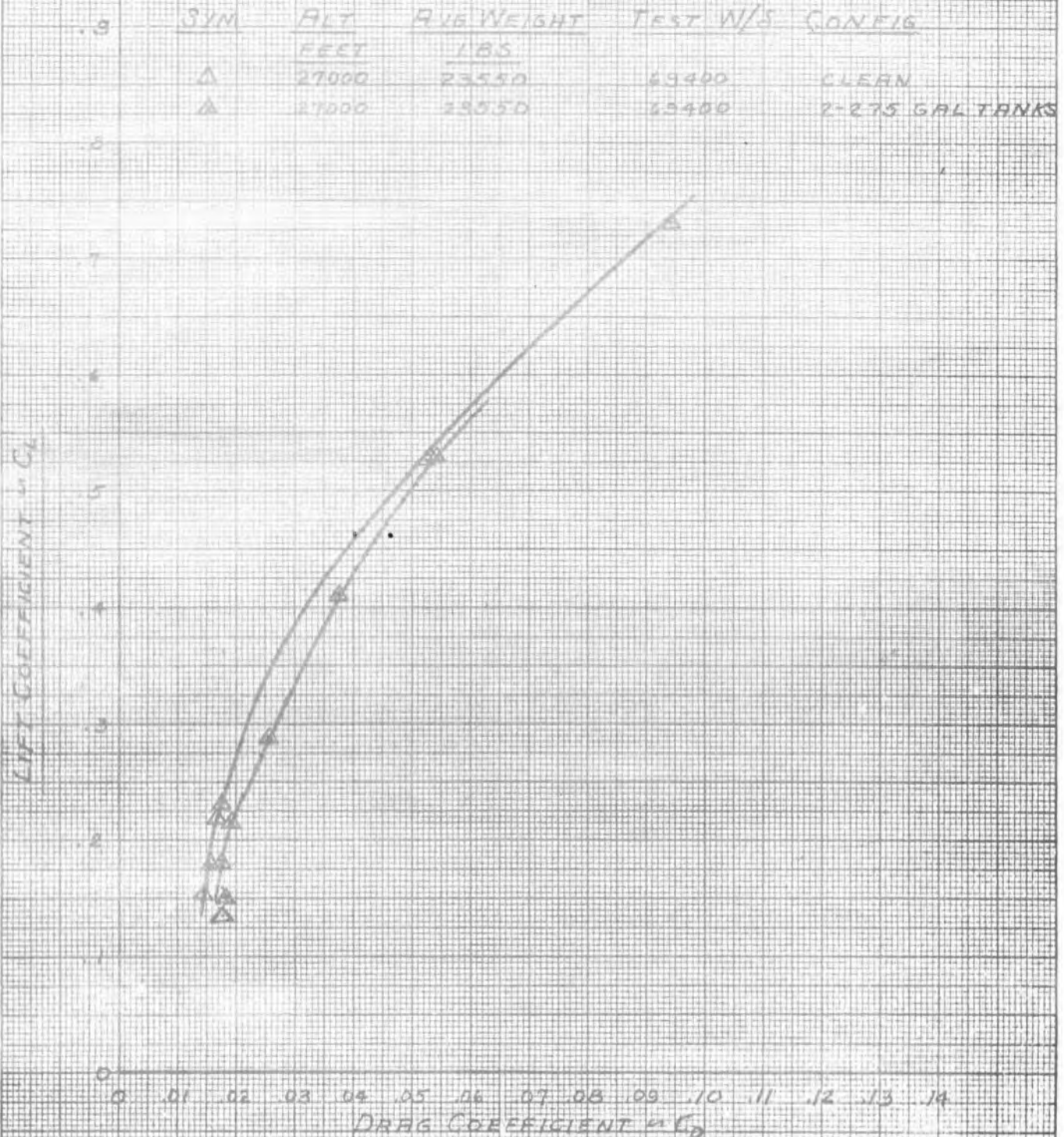


FIG No 13
 C_L^2 VERSUS C_D

YF-100A USAF No 52-5754

DRAG COMPARISON WITH TANKS ON AND TANKS OFF

SYM	ALT	Avg WEIGHT	TEST W/G	CONFIG
	FEET	LBS		
Δ	27000	23550	69400	CLEAN
▲	27000	23550	69400	2-275 GAL TANKS

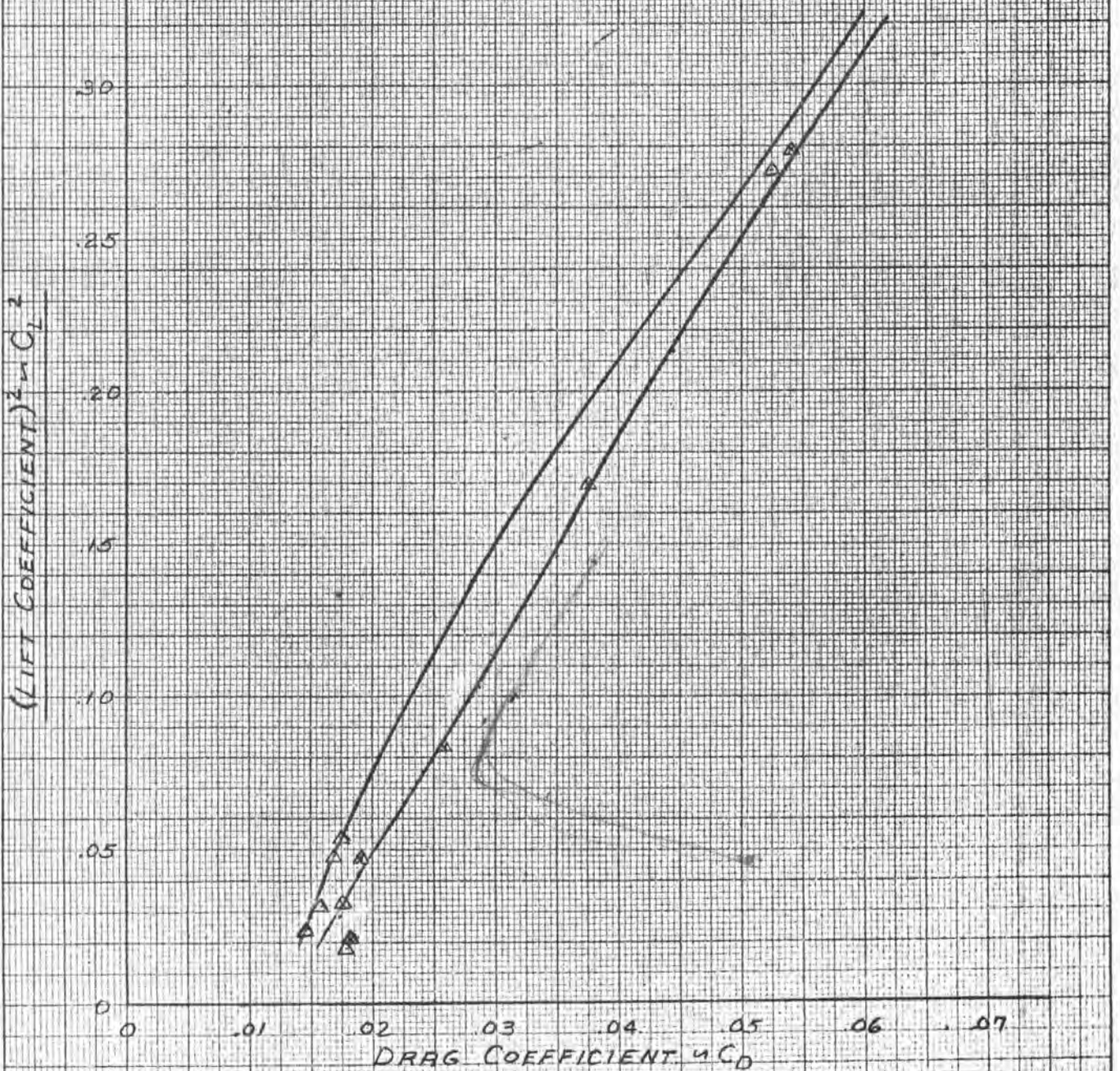
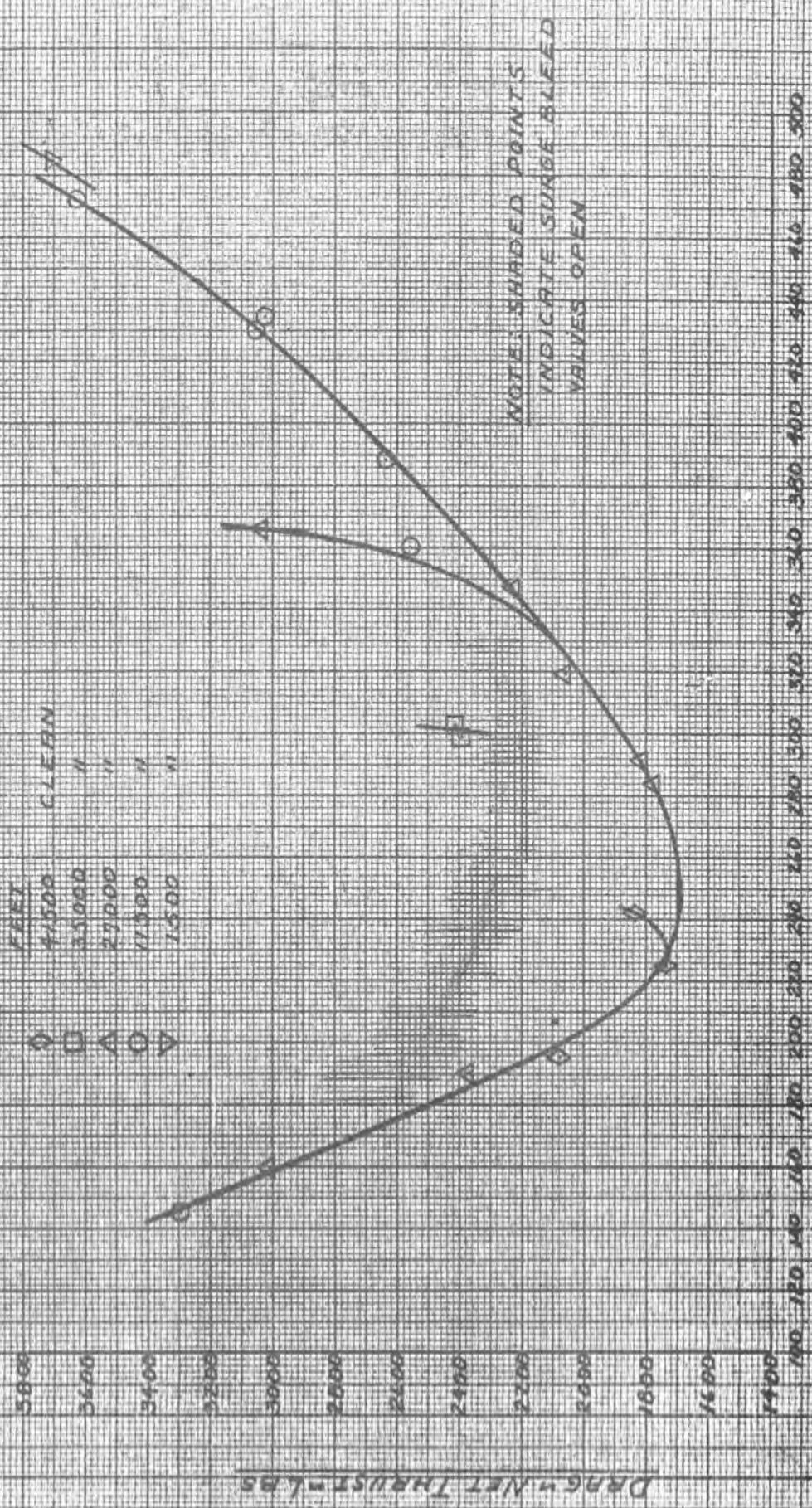


Fig No 14
 DRAG VS EQUIVALENT AIRSPEED
 YF-100A USAF No 52-3754
 23,550 LBS GROSS WEIGHT
 NO EXTERNAL STORES ATTACHED

SYM	HAT	CONFIG
◇	41500	CLEAN
□	33000	"
△	27000	"
○	11500	"
▽	7500	"



EQUIVALENT AIRSPEED - KNOTS

DRAG - NET THRUST - LBS

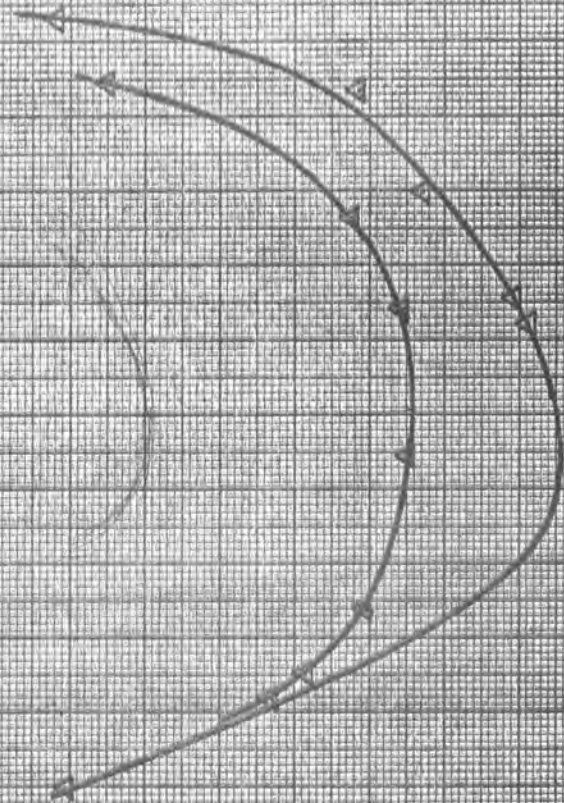
Fig No 15
 Drag vs Equivalent Airspeed

YF-100A USAF No 52-5754
 23550 Lbs Gross Weight

Drag Comparison with Tanks On and Tanks Off

SYM	ALT FEET	CONFIG
Δ	17000	CLEAN
Δ	21000	2-215 GAL TANKS

DRAG NET THRUST - LBS.

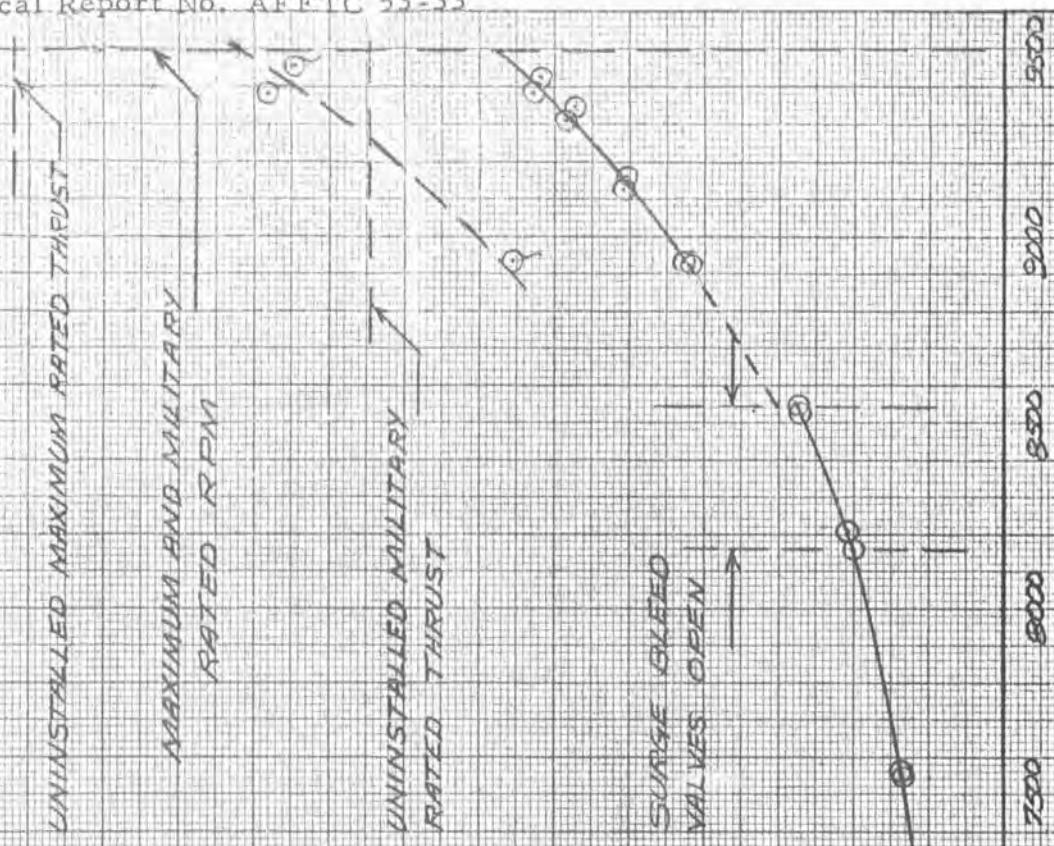


EQUIVALENT AIRSPEED - KNOTS

FIGURE No 16
STATIC THRUST
XF-100A, USAF No 52-5754
XJ-57-R-7 ENGINE
JP-4 FUEL

NOTE: THRUST RATINGS
OBTAINED FROM PRATT &
WHITNEY SPECIFICATION
NO. A-1639-B FOR UNINSTALLED
ENGINE RATINGS

SYM
 ○ AIR ON - NOZZLE OPEN
 ○ AIR OFF - NOZZLE CLOSED
 ● AIR OFF - NOZZLE OPEN



THRUST ~ $F_0 \frac{2992}{P_0}$ ~ POUNDS

RPM ~ $N_2 \sqrt{\frac{2897}{T_2}}$

FIG No 17

STATIC THRUST
 YF-100A USAF No 57-5754
 XJ57-P-7 ENGINE
 JP-4 FUEL

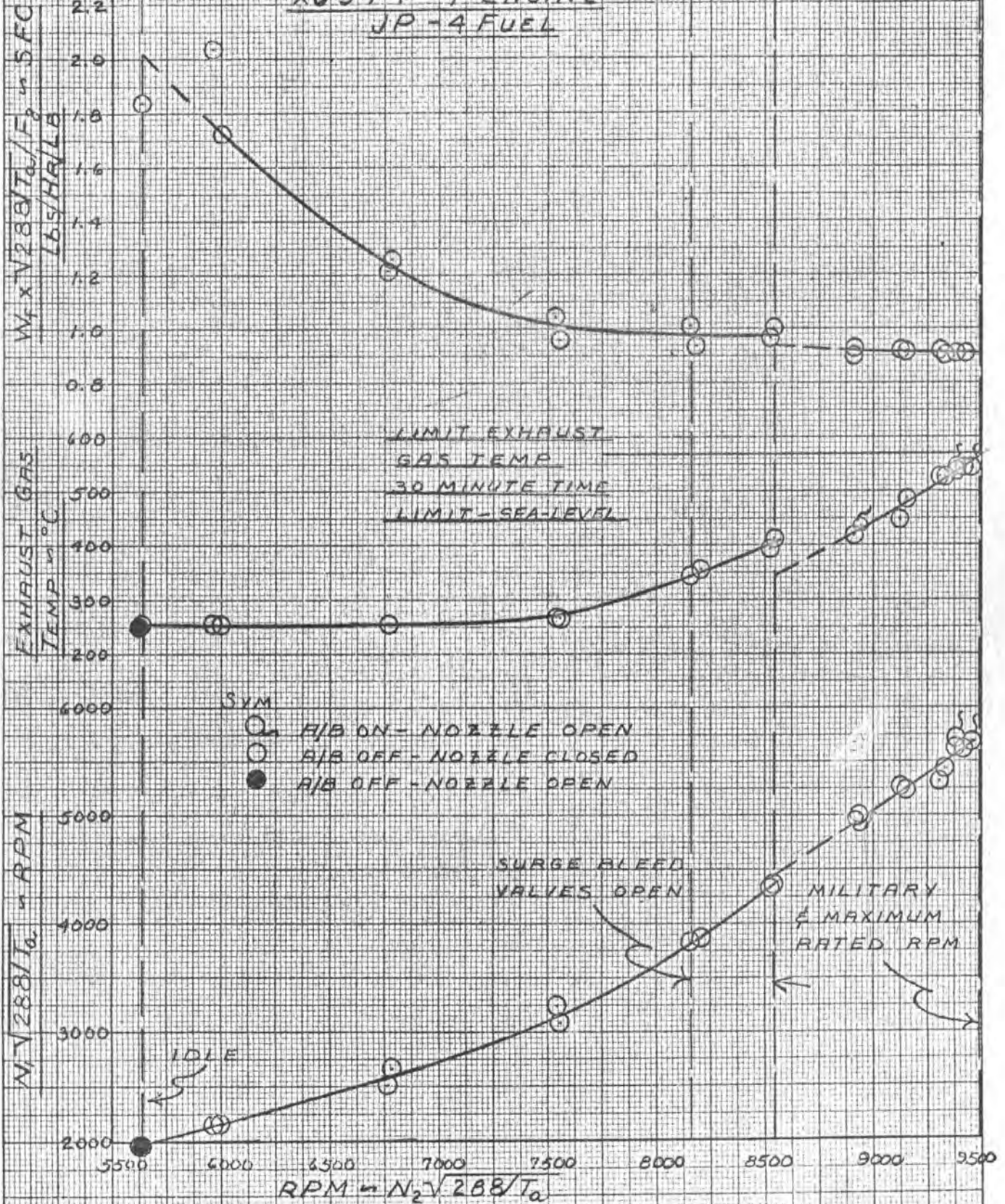


FIG. No. 18
 TAIL PIPE PROBE CALIBRATION
 YF-100A USAF No 52-5754
 XJ57-P-7 ENGINE
 NOZZLE CLOSED

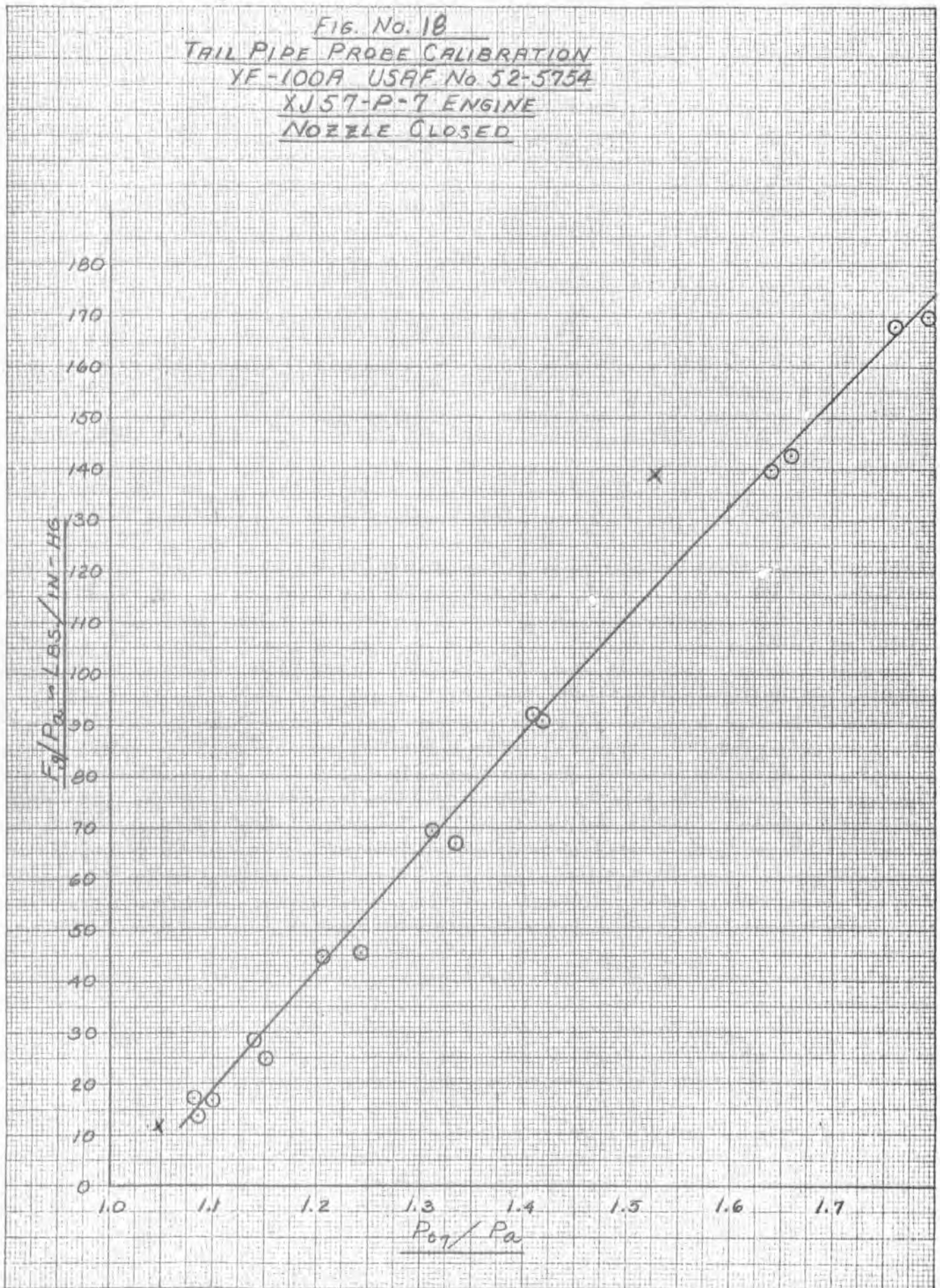


FIG No. 19
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED

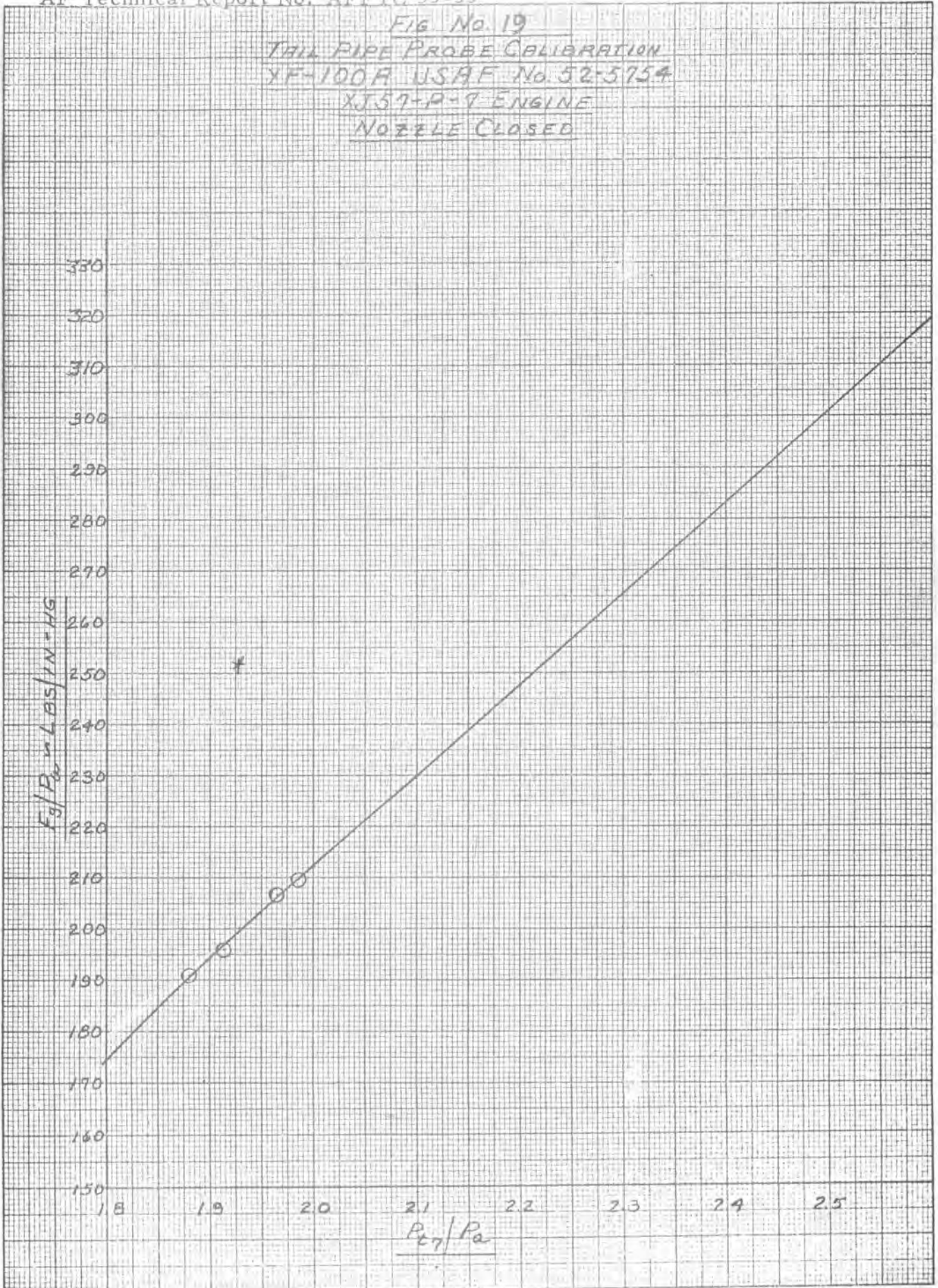


FIG. No. 20
TAIL PIPE PROBE CALIBRATION
YF-100A USRF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED

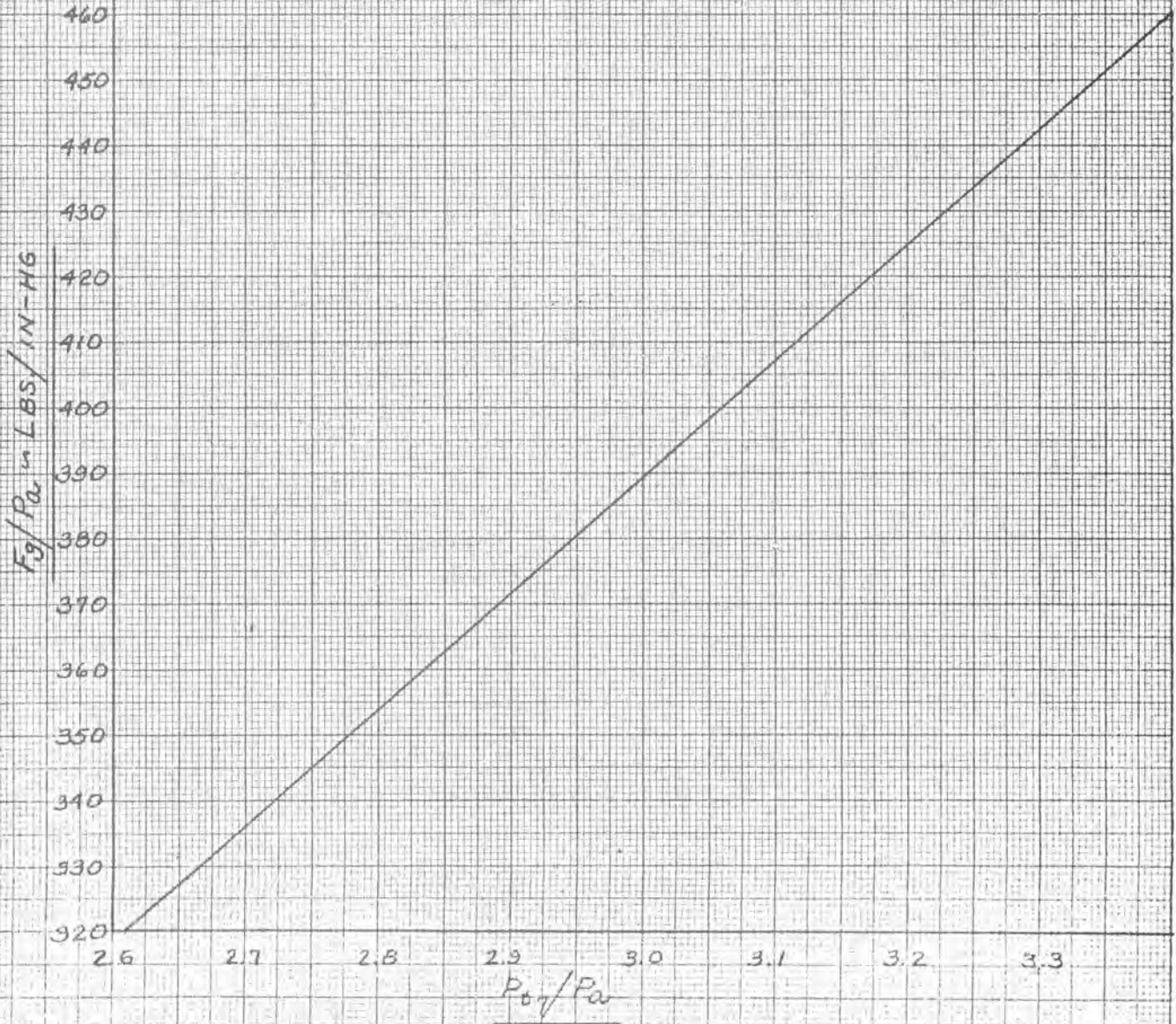


FIG. NO. 21
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED

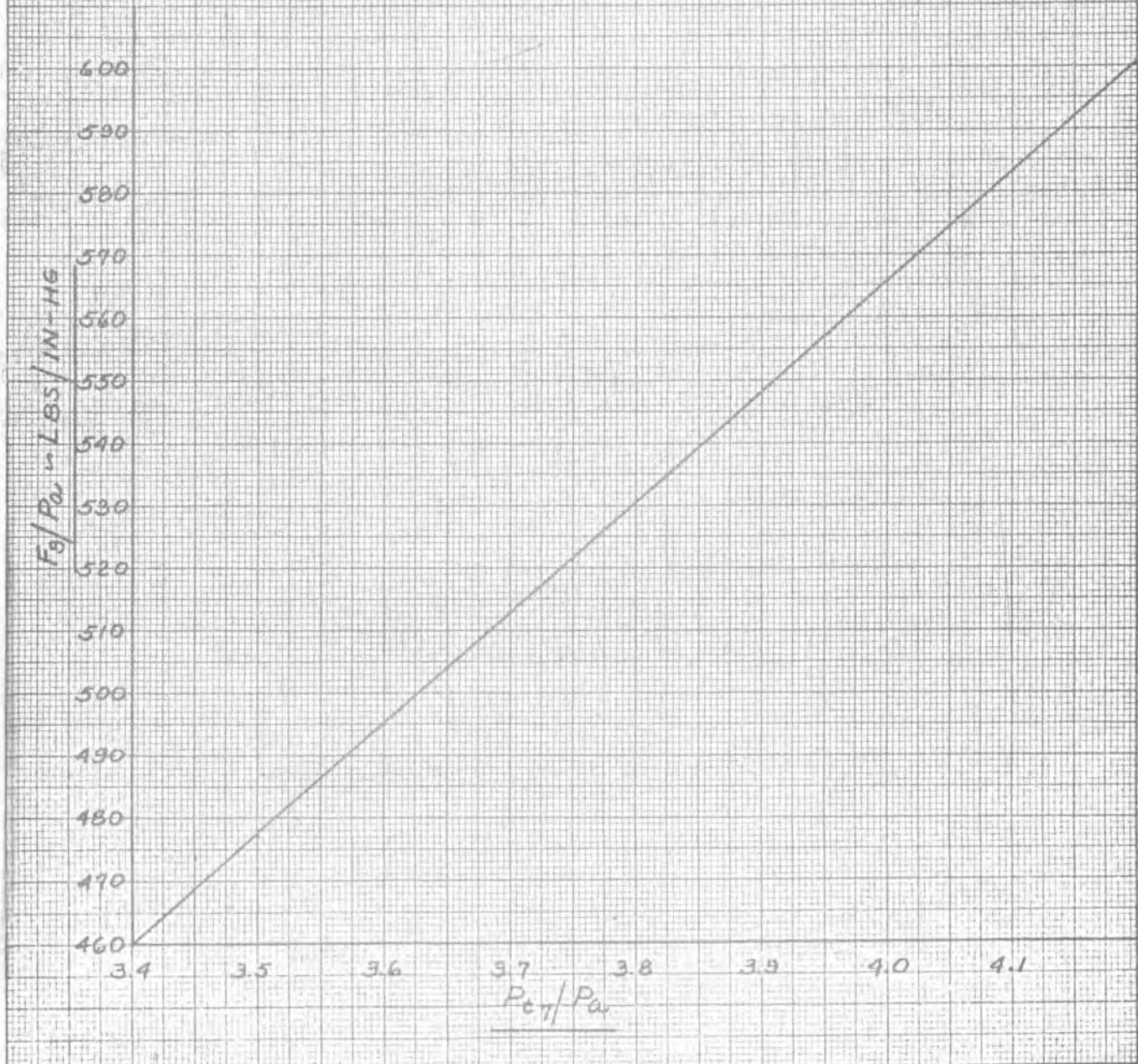


FIG. No. 22
TAIL PIPE PROBE CALIBRATION
YF-100A USAF No. 52-5754
XJ57-P-7 ENGINE
NOZZLE CLOSED

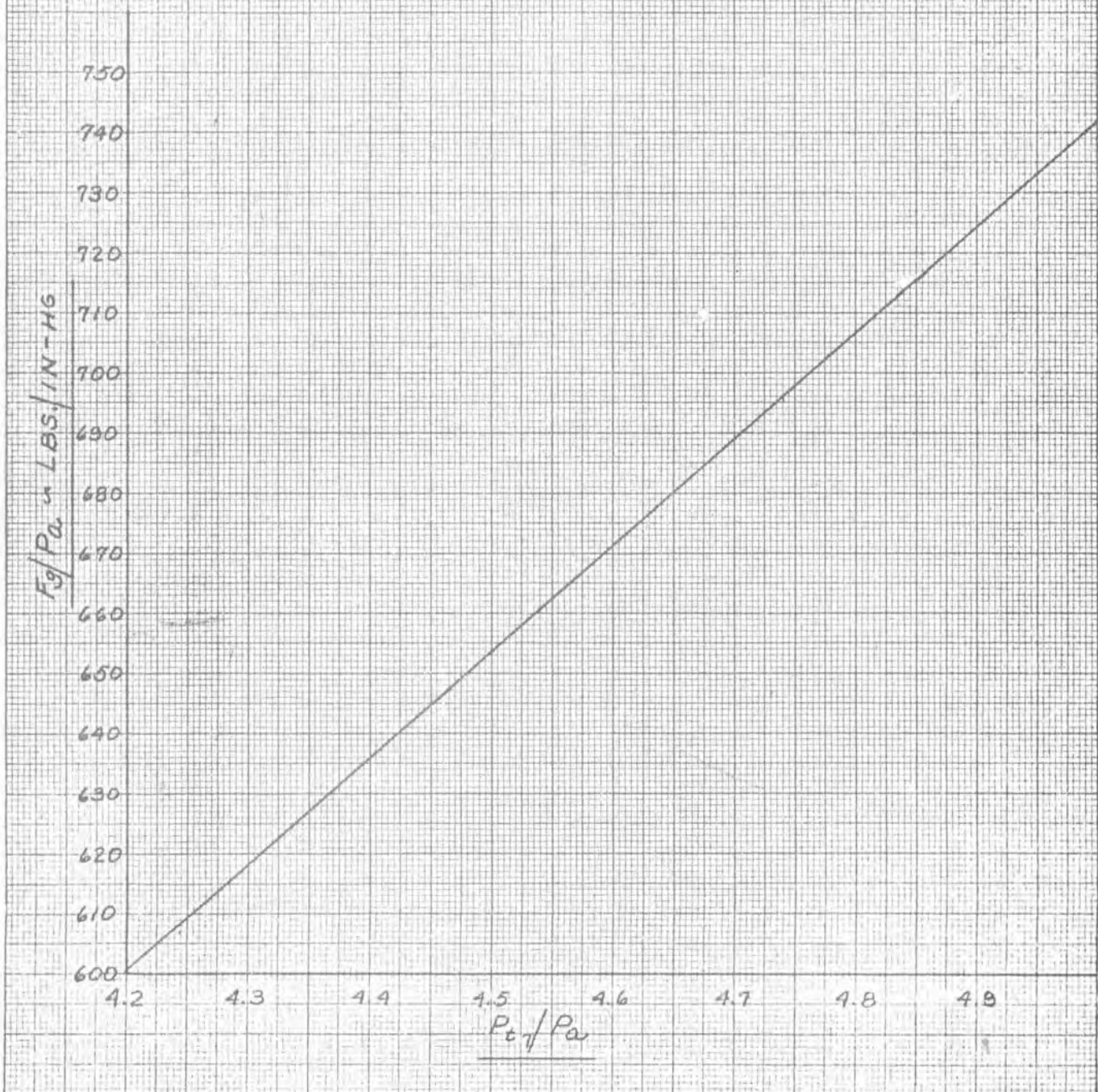
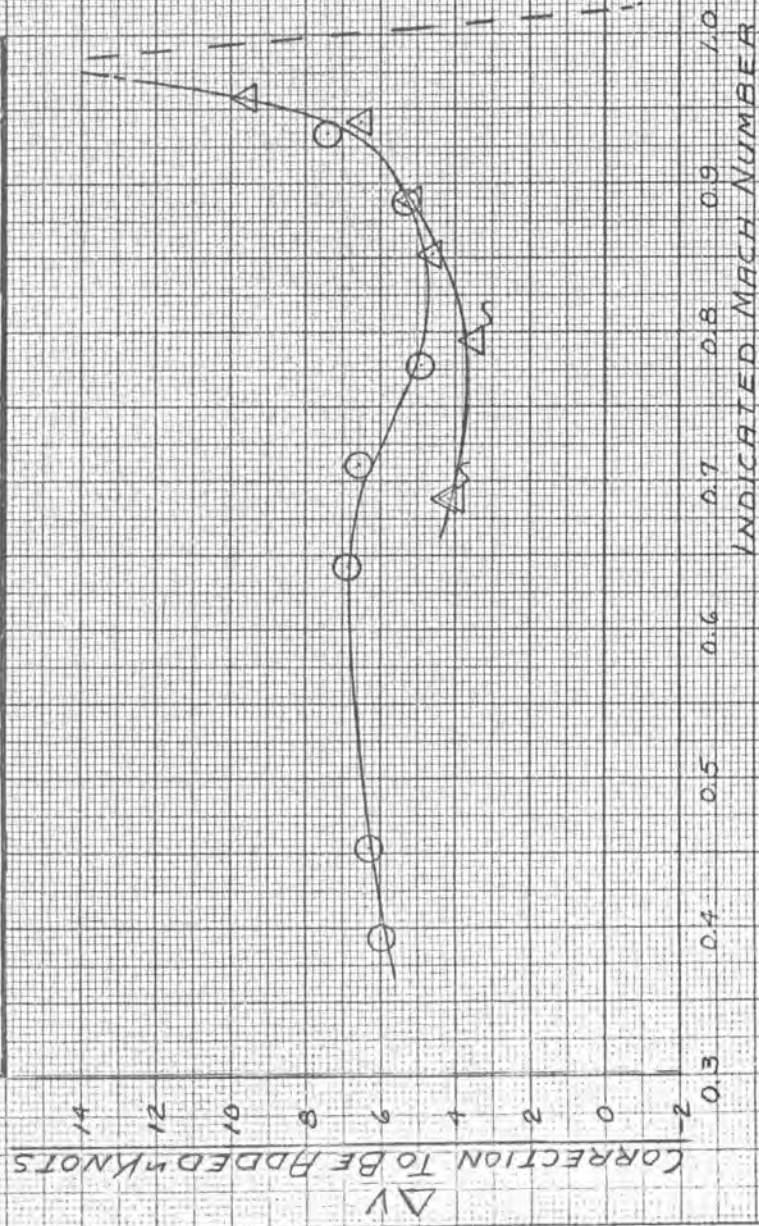
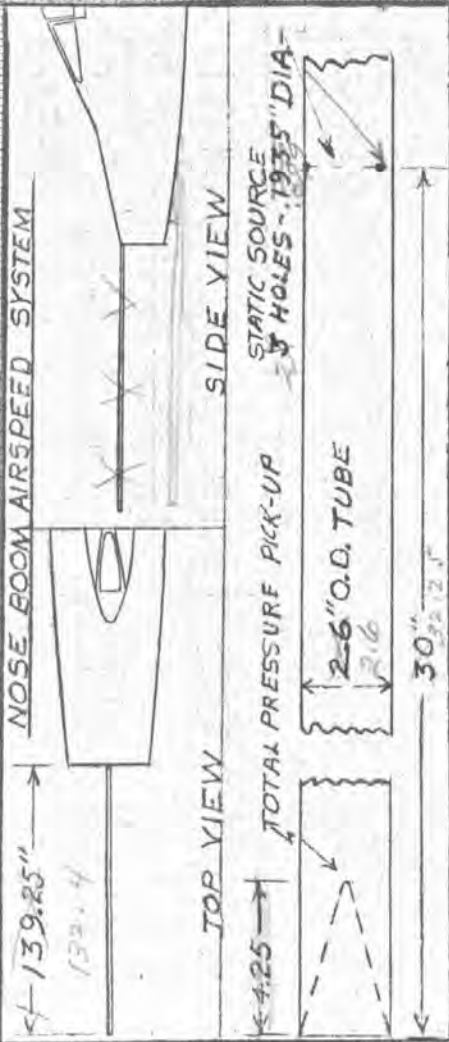


FIGURE No 23

AIR SPEED CALIBRATION
 YF-100A USAF No 52-5754
 GEAR UP

SYM	ALTITUDE	FLY WEIGHT
○	FEET 23000	LBS 23000
△	40000	23000
△	40000	23000
◇	25000	23000

○ TOWER FLY BYS
 △ AIRPLANE FLY BYS
 △ PACE POINTS
 ◇ RADAR TRACKED DIVE



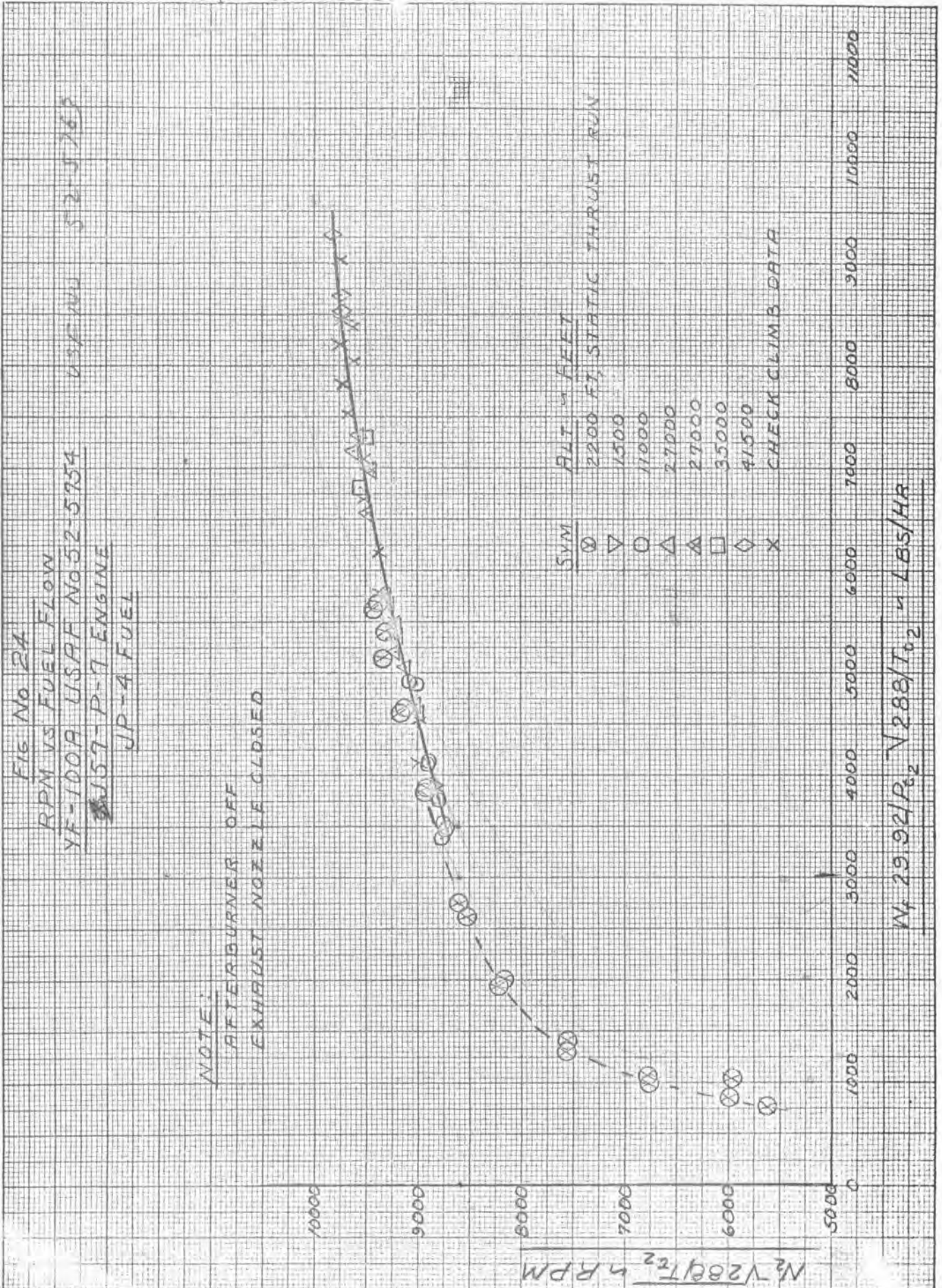
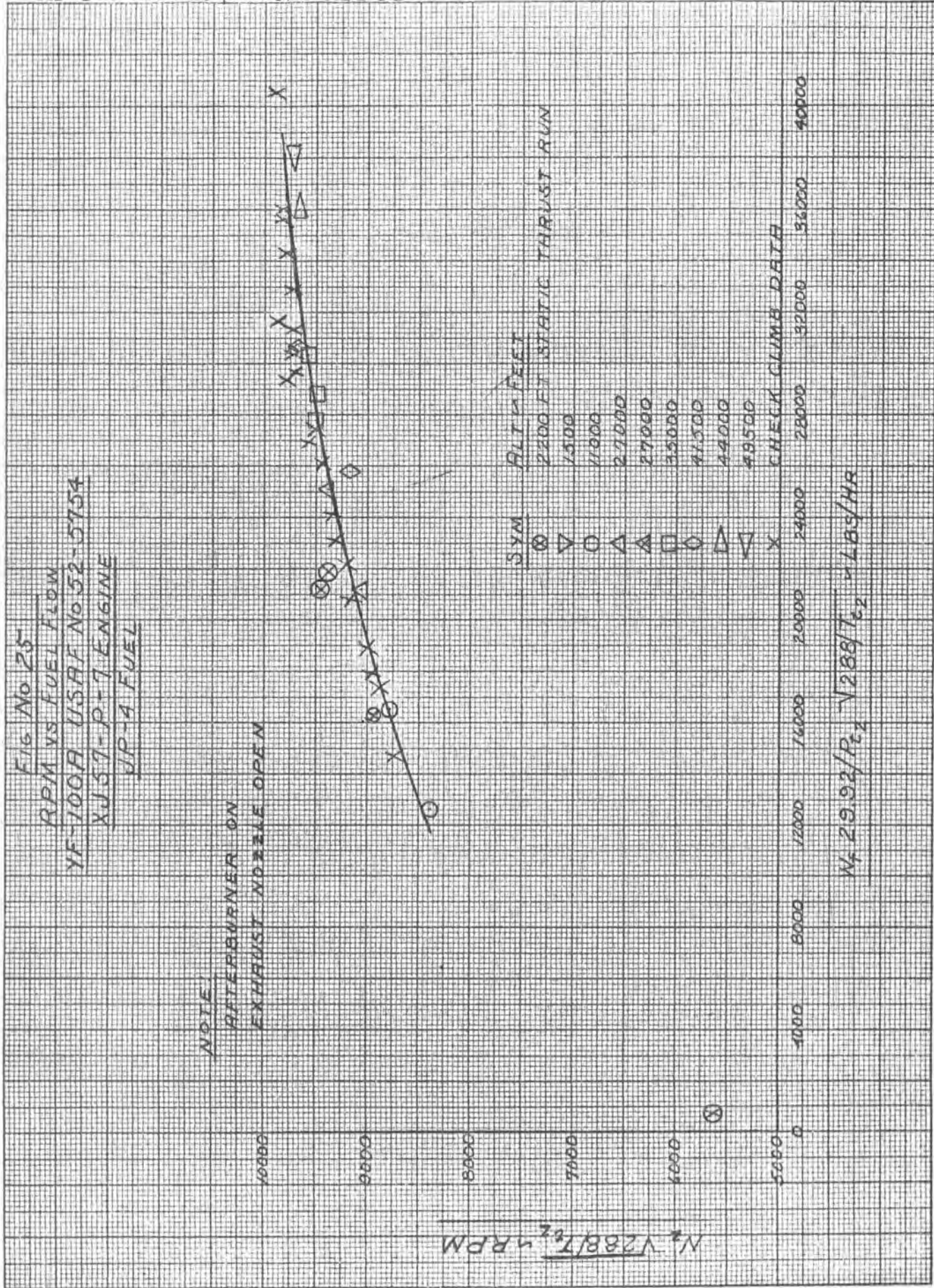


FIG No 25
 RPM VS FUEL FLOW
 YF-100A USAF No 52-5754
 XJS7-P-7 ENGINE
 JP-4 FUEL

NOTE:
 AFTERBURNER ON
 EXHAUST NOZZLE OPEN



$$M_2 = 2.092 / P_{t2} \sqrt{288 / T_{t2}} = 4.05 / HR$$

FIG. NO. 26
 N₂ VS COMPRESSOR INLET TEMPERATURE
 YF-100A - USAF No. 52-5754
 XJ-57-P-7 ENGINE

NOTE: DATA REPRESENTS THE VARIATION
 OF N₂ WITH INLET TEMPERATURE
 FOR A CONSTANT POWER LEVER SETTING

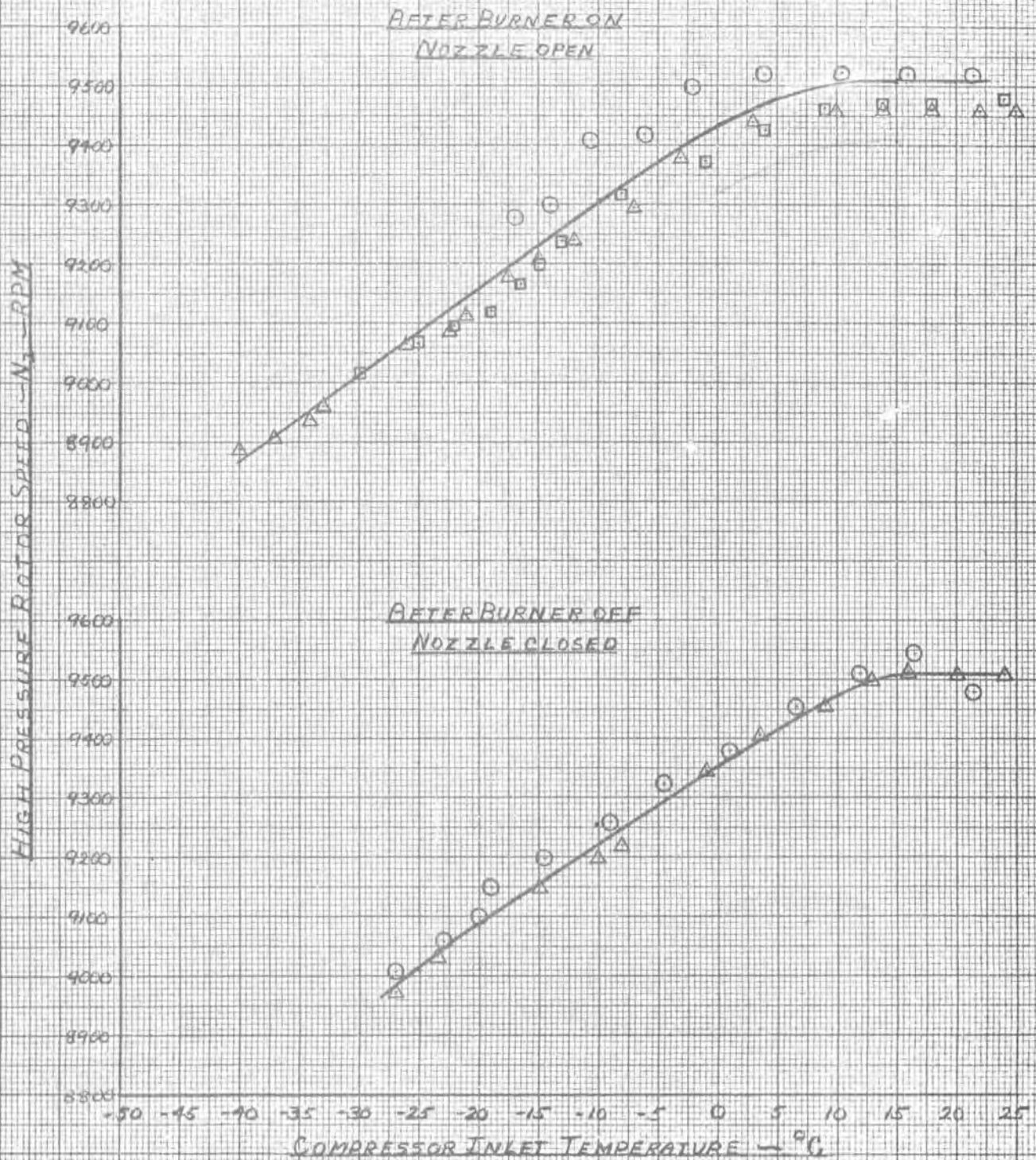


FIG. No. 27
 ESTIMATED VARIATION OF ENGINE AIRFLOW
 WITH LOW ROTOR SPEED
 XJ57-P-7

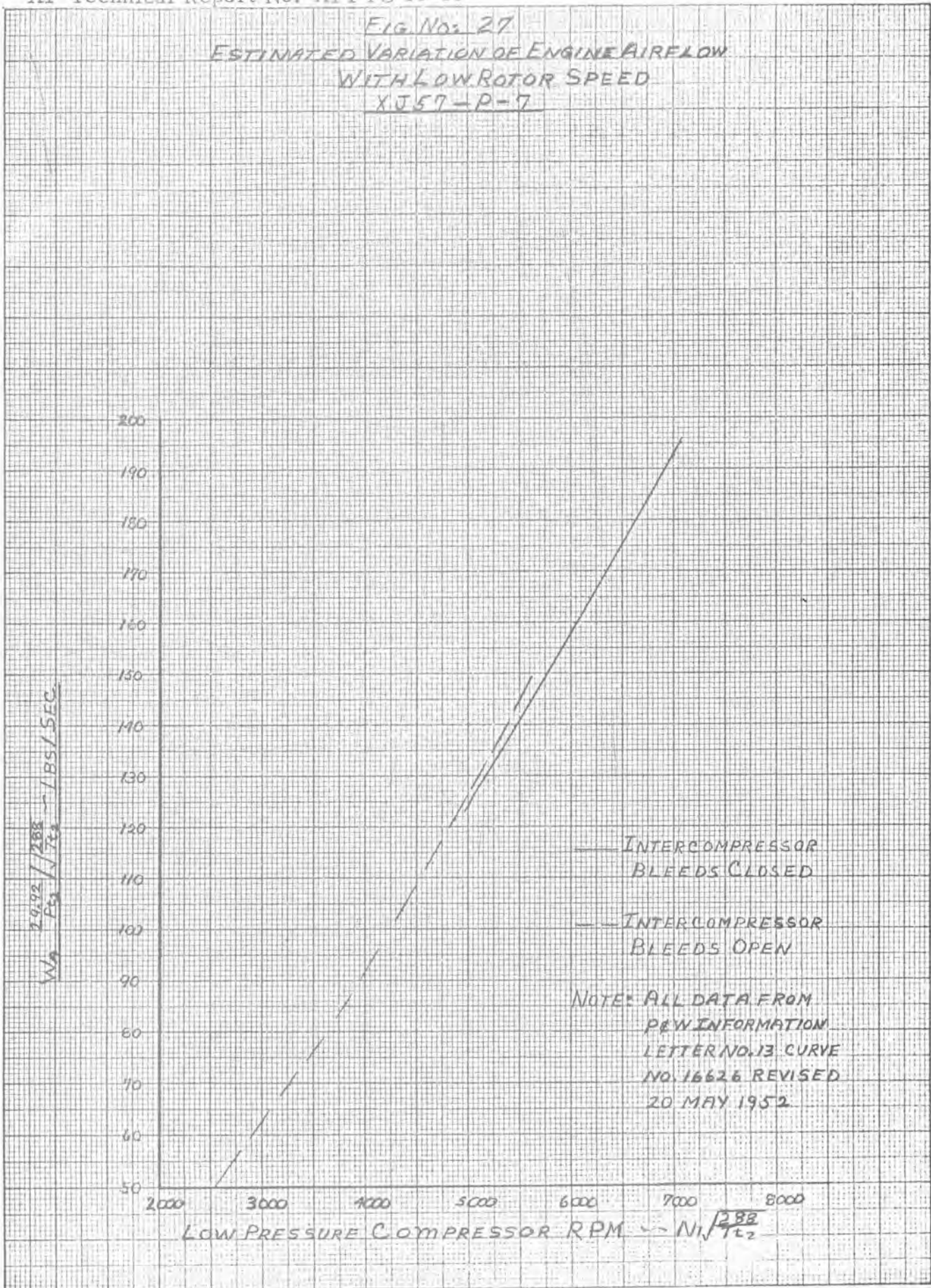


FIG No 28
STABILIZER CONTROL FRICTION
YF-100A USAF No 52-5754
WIND \approx ZERO SFAT \approx 20 °C

NOTE: ARROWS INDICATE DIRECTION
STABILIZER WAS DEFLECTED

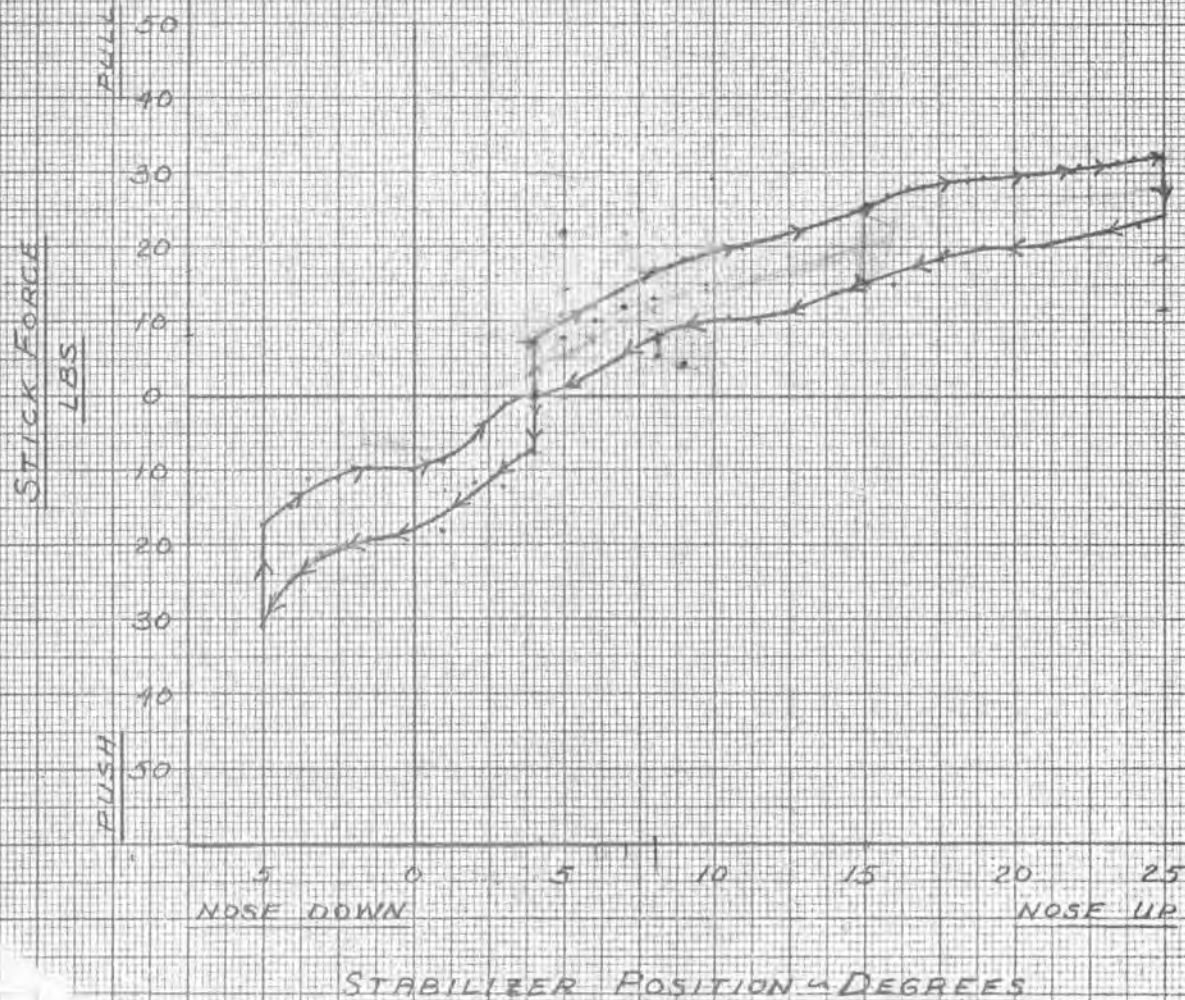


FIG No 29

RUDDER CONTROL FRICTION
YF-100A USAF No 52-5754
WIND \approx ZERO SFHT \approx 20 °C

NOTE: ARROWS INDICATE DIRECTION
RUDDER WAS DEFLECTED

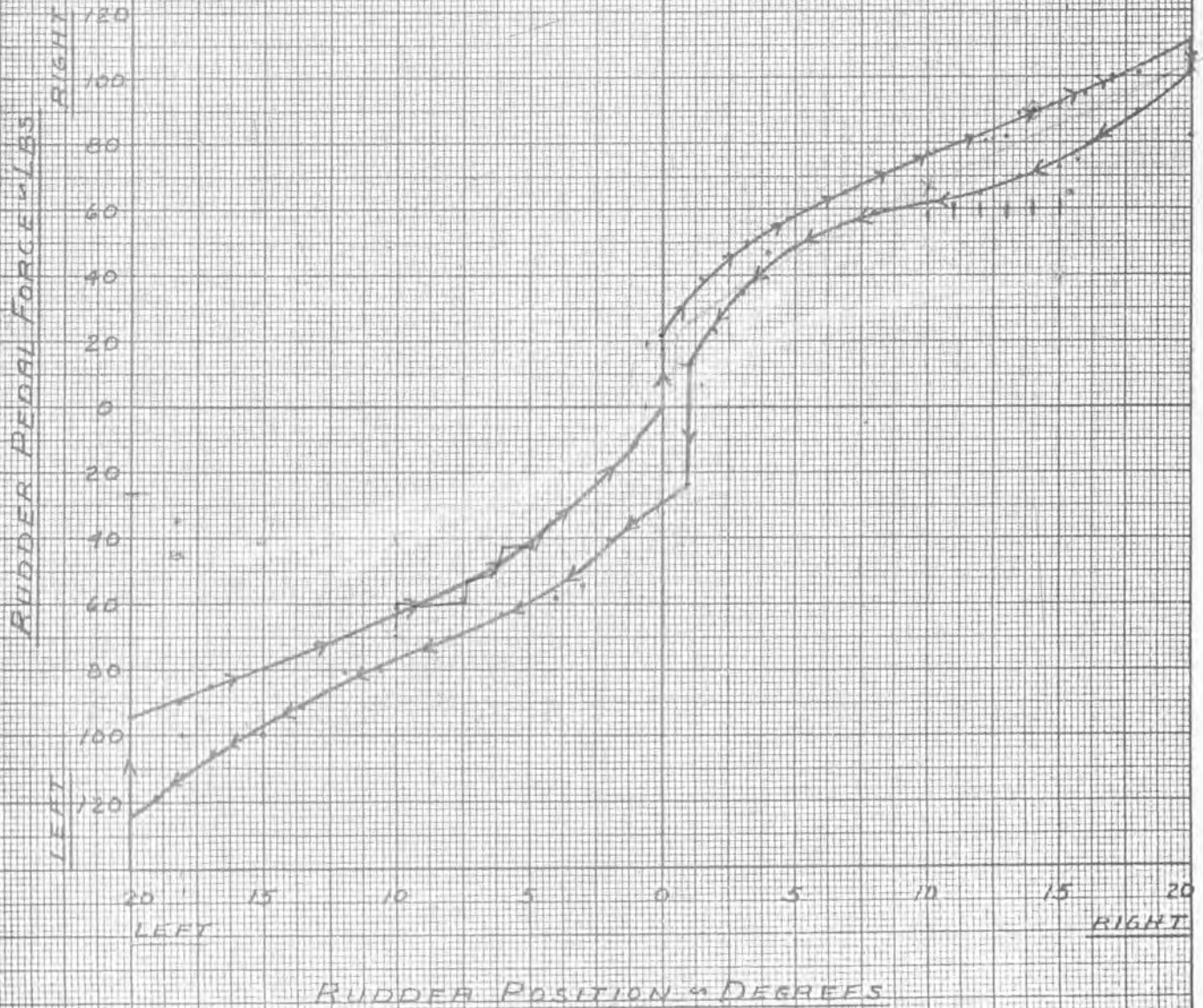


FIG NO 30
AILERON CONTROL FRICTION
YF-100A USAF NO 52-5754
WIND \approx ZERO SEAT \approx 20 °C

NOTE: ARROWS INDICATE DIRECTION
AILERON WAS DEFLECTED

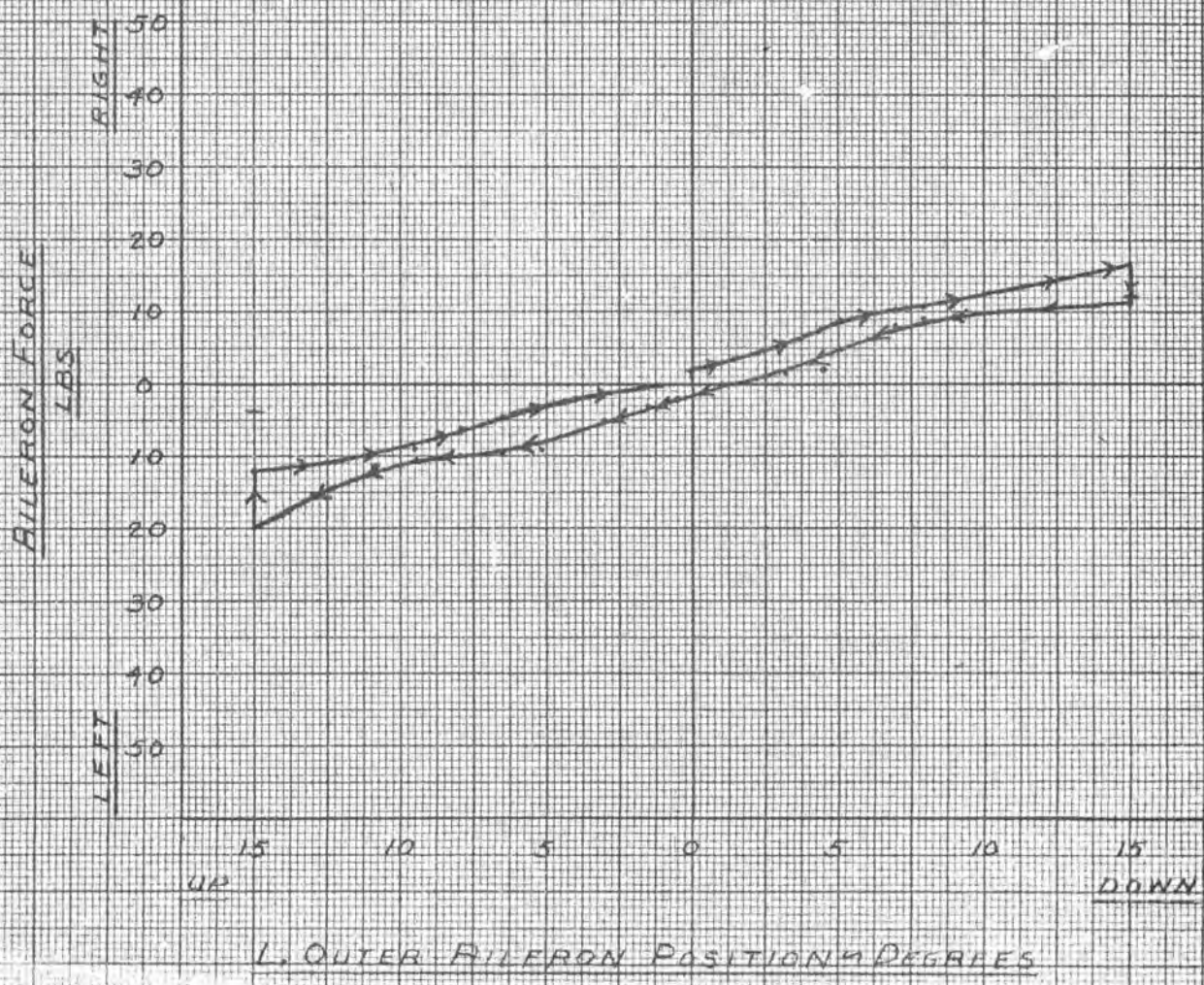


Figure No. 31

LANDING TIME HISTORY

YF-100A, USAF No. 52-5754

Landing Configuration

TRIM CONDITIONS

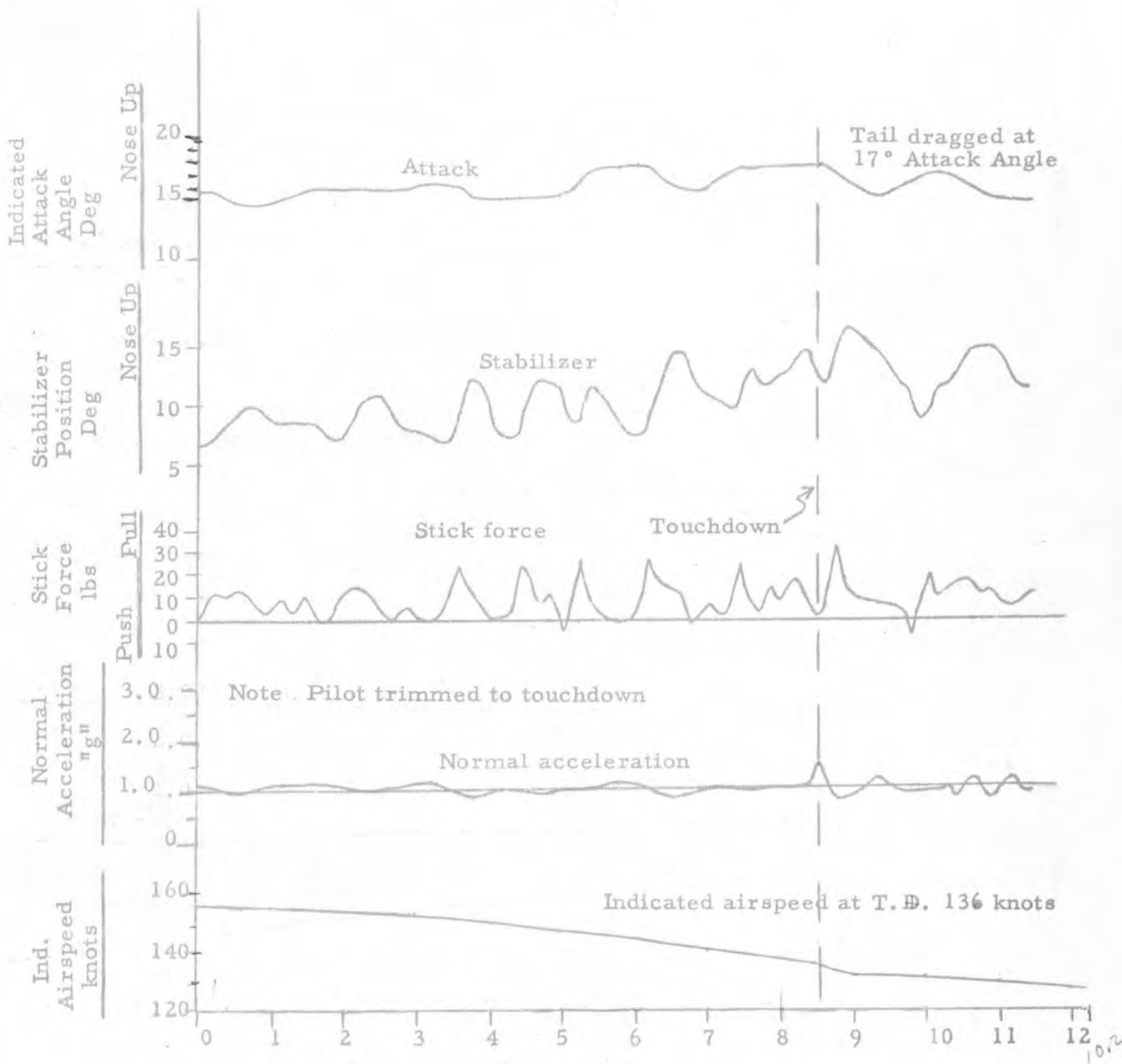
Altitude 2,250 feet

Weight 21,800 lbs

CG 29.5 % MAC

Dive Brake Full Open

No external stores attached



Time - Seconds

APPENDIX I

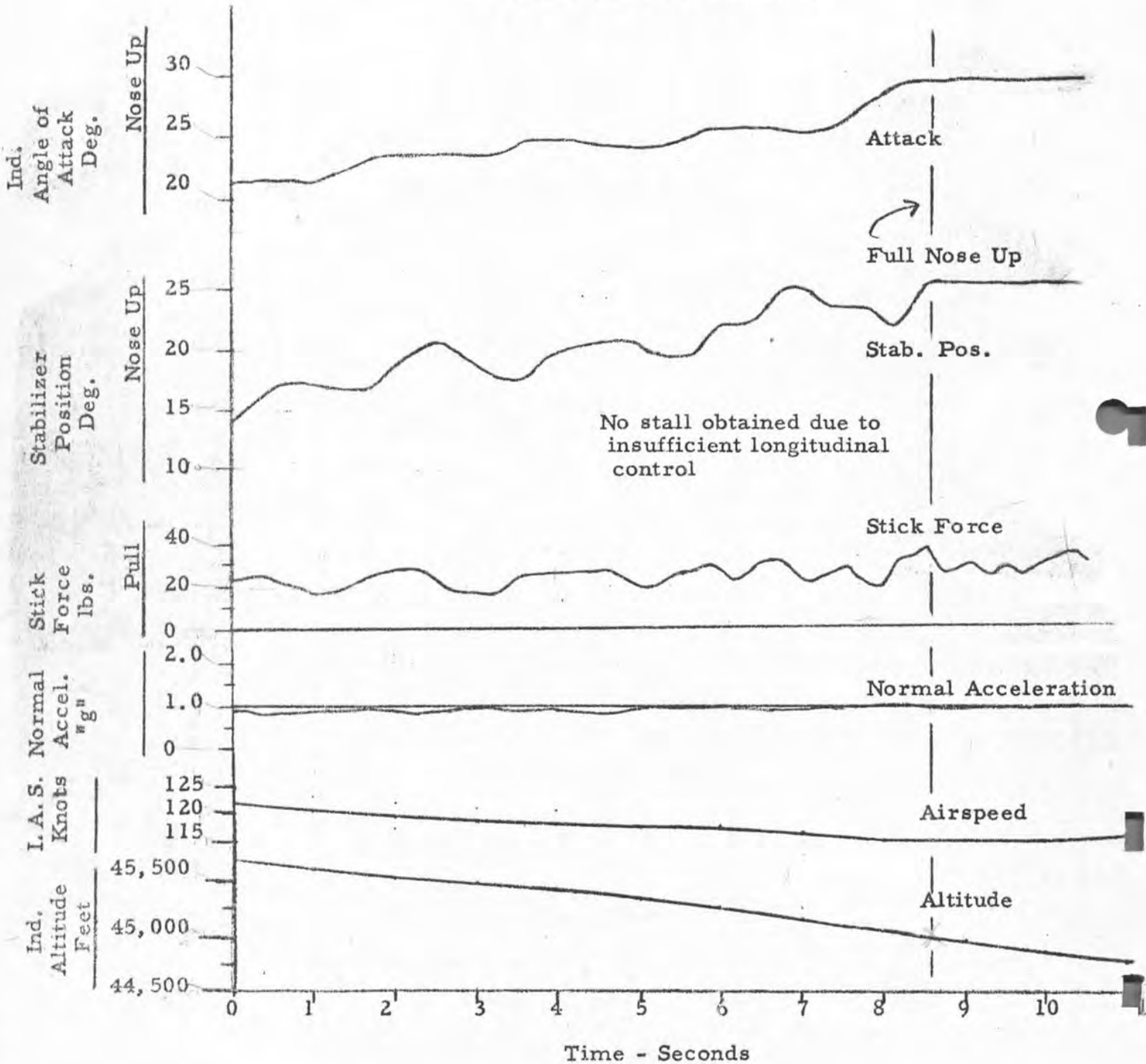
STALL TIME HISTORY

YF-100 A, USAF No. 52-5754
Power Configuration (A/B off)

TRIM CONDITIONS

CAS	220	knots	Altitude	44,800	feet
CG	30	% mac	Weight	21,500	lbs
N ₂	8950	rpm	Stab. Pos.	4.3°	nose up

No external stores attached



Time - Seconds

APPENDIX I

STALL TIME HISTORY

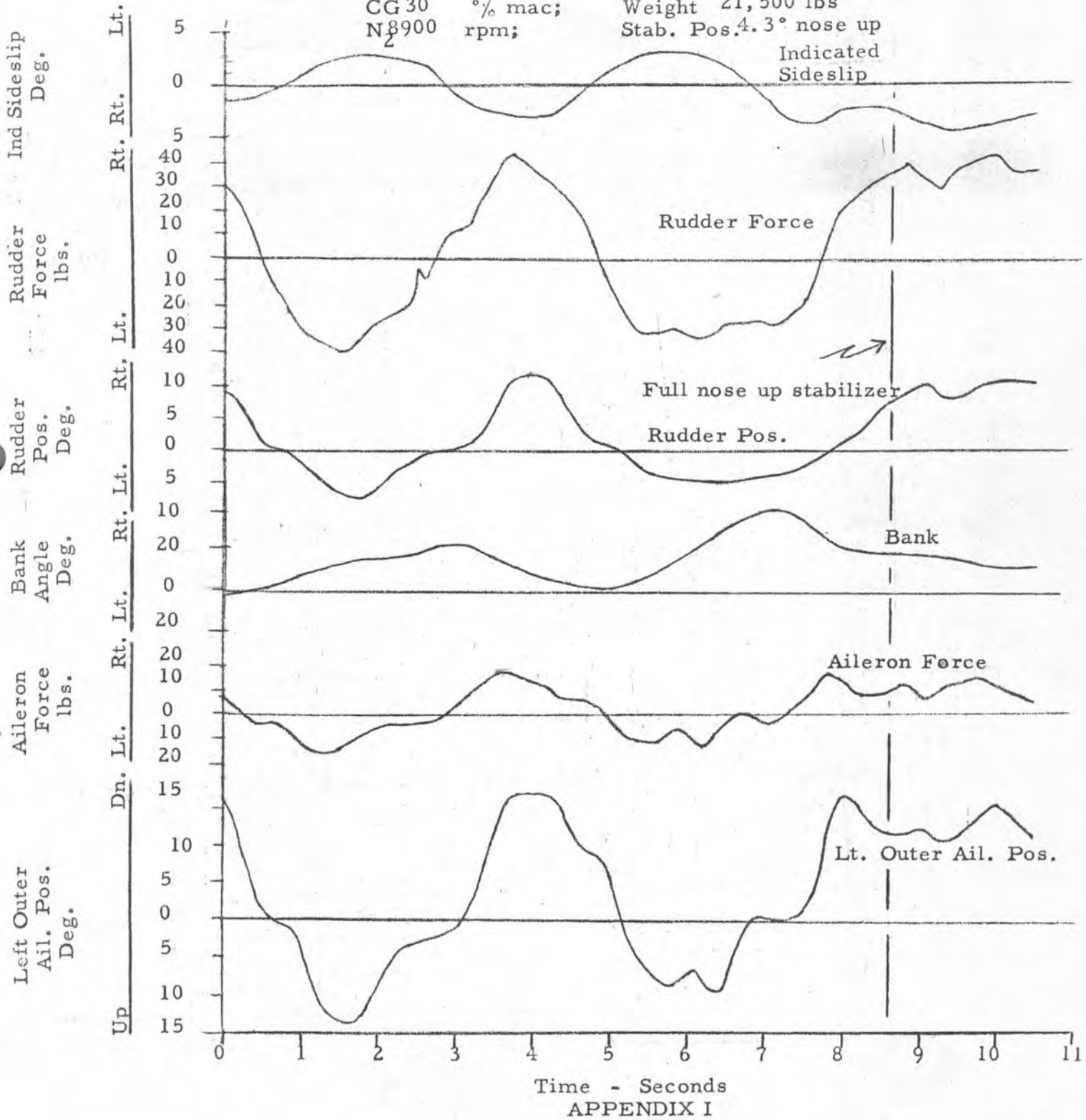
YF-100 A, USAF No. 52-5754

Power Configuration (A/B off)

TRIM CONDITIONS

CAS 220 knots
CG 30 % mac;
N₂ 900 rpm;

Altitude 44,800 feet
Weight 21,500 lbs
Stab. Pos. 4.3° nose up



Time - Seconds

APPENDIX I

Figure No. 33

STALL TIME HISTORY
YF 100A, USAF No. 52-5754
Power Approach Configuration

TRIM CONDITIONS

IAS 1.55.5 knots
CG 30.6 % mac;
N₂ 9100 rpm;

Altitude 8040 feet
Weight 21,100 lbs
Stab. Pos 7.8 nose up

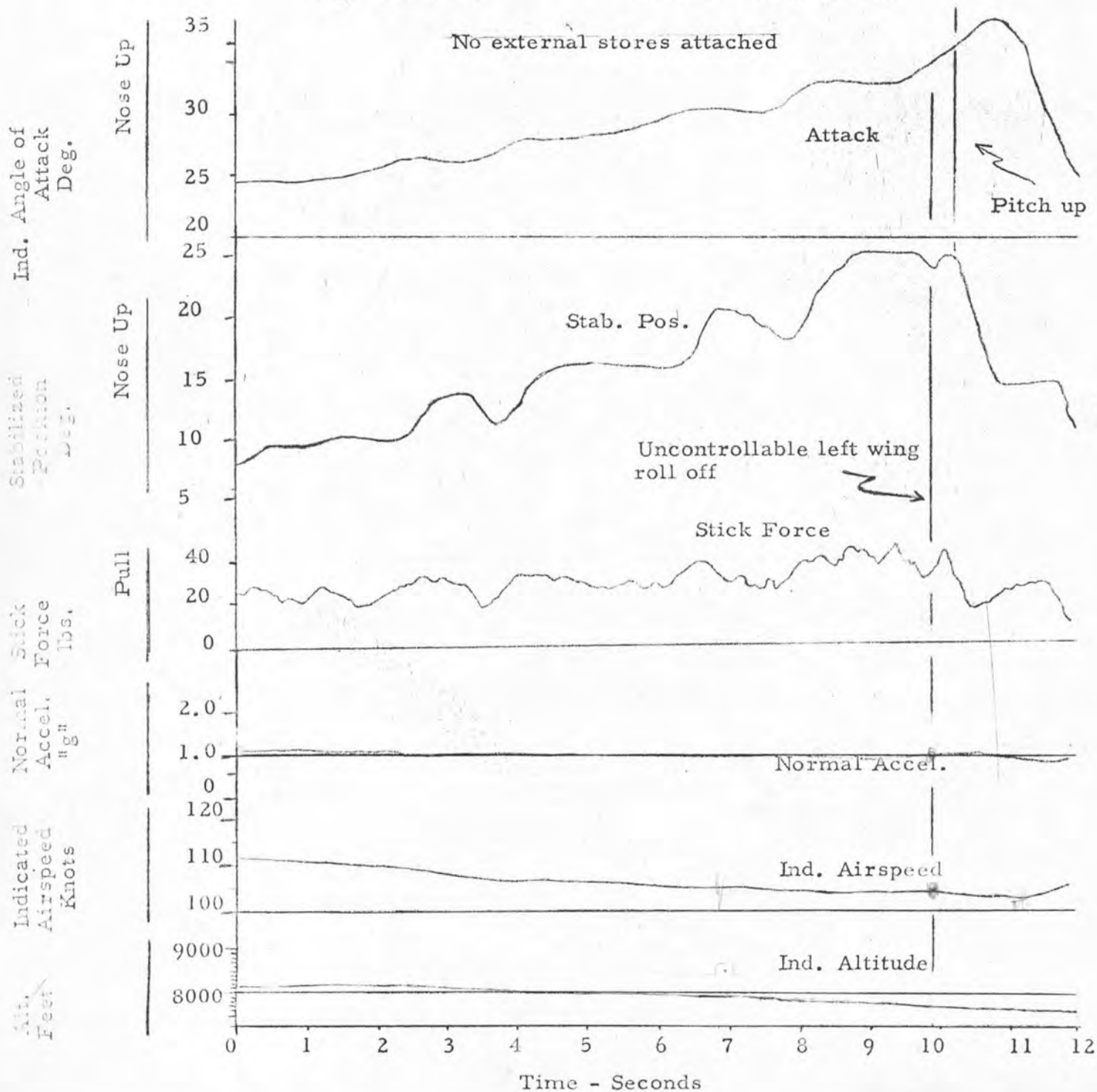


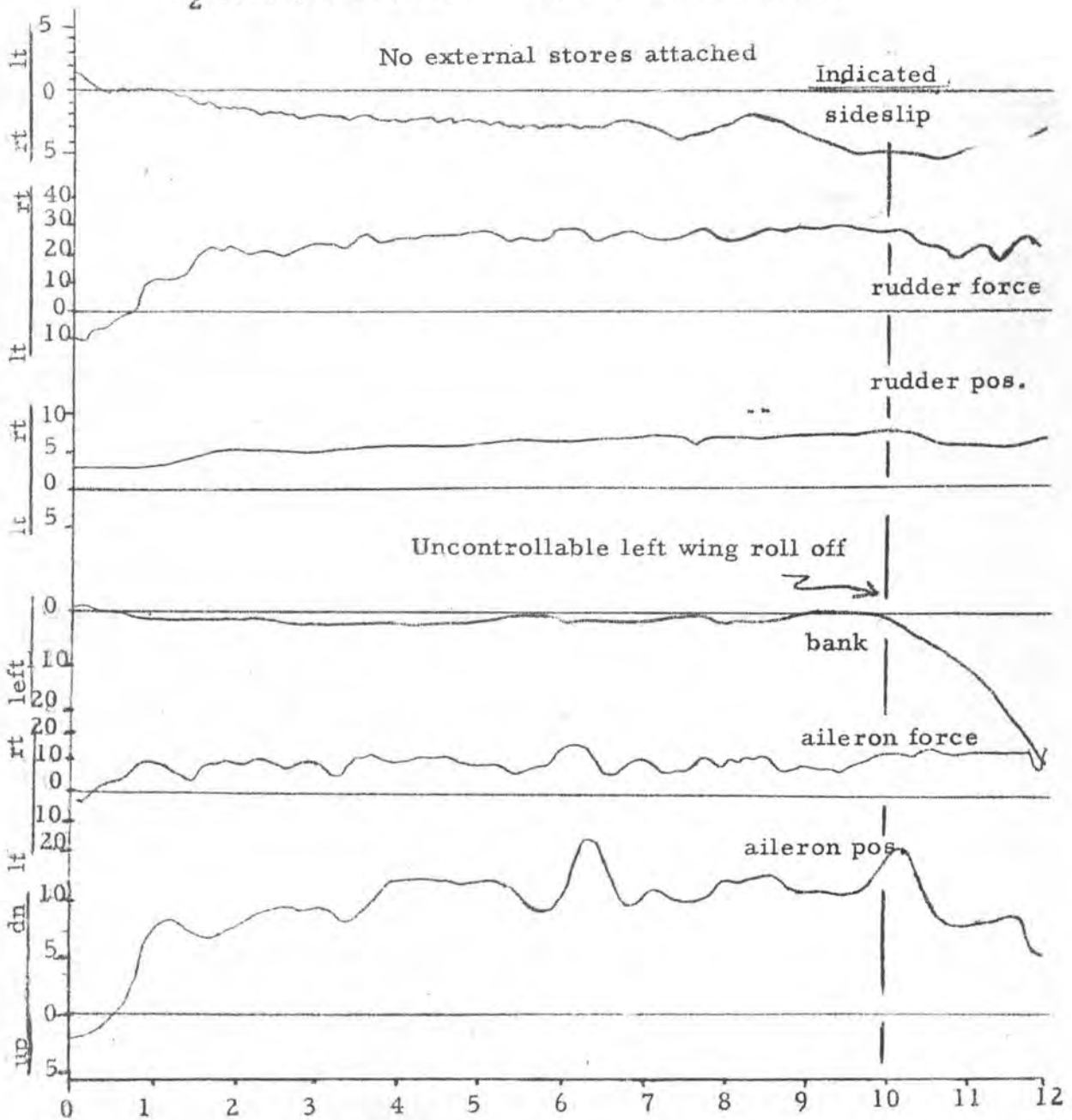
Figure No. 33

STALL TIME HISTORY
YF 100 A, USAF No. 52-5754

Power Approach Configuration

TRIM CONDITIONS

LAS 155.5 knots; Altitude 8040 feet
CG 30.6 %mac; Weight 21000 lbs
N₂ 9100 rpm; Stab. Pos. 7.8° nose up



Time - Seconds

APPENDIX I

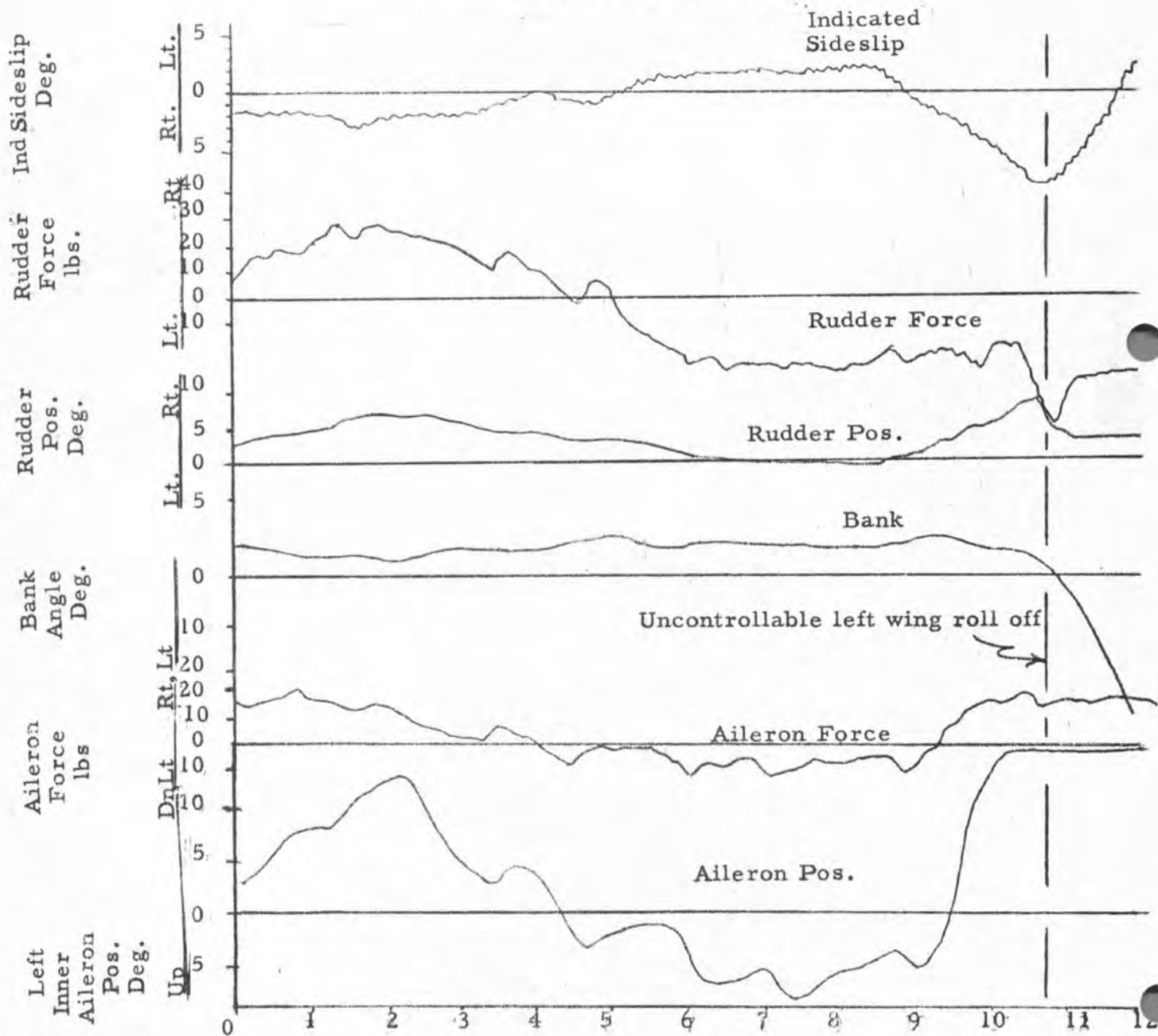
STALL TIME HISTORY

YF 100 A, USAF No. 52-5754

Power Approach Configuration

CAS 157.5 knots Altitude 10,340 feet
 CG 29.9% mac; Weight 21,250 lbs
 N₂ 9100 rpm; Stab. Pos. 7.7° nose up

No external stores attached



Time - Seconds

APPENDIX I

STALL TIME HISTORY
 YF 100A, USAF No. 52-5754
 Powered Approach Configuration
 TRIM CONDITIONS

CAS 157.5 knots
 CG 29.9% mac;
 N₂ 9100rpm;

Altitude 10,340 feet
 Weight 21,250 lbs
 Stab. Pos. 7.7° nose up

No external stores attached

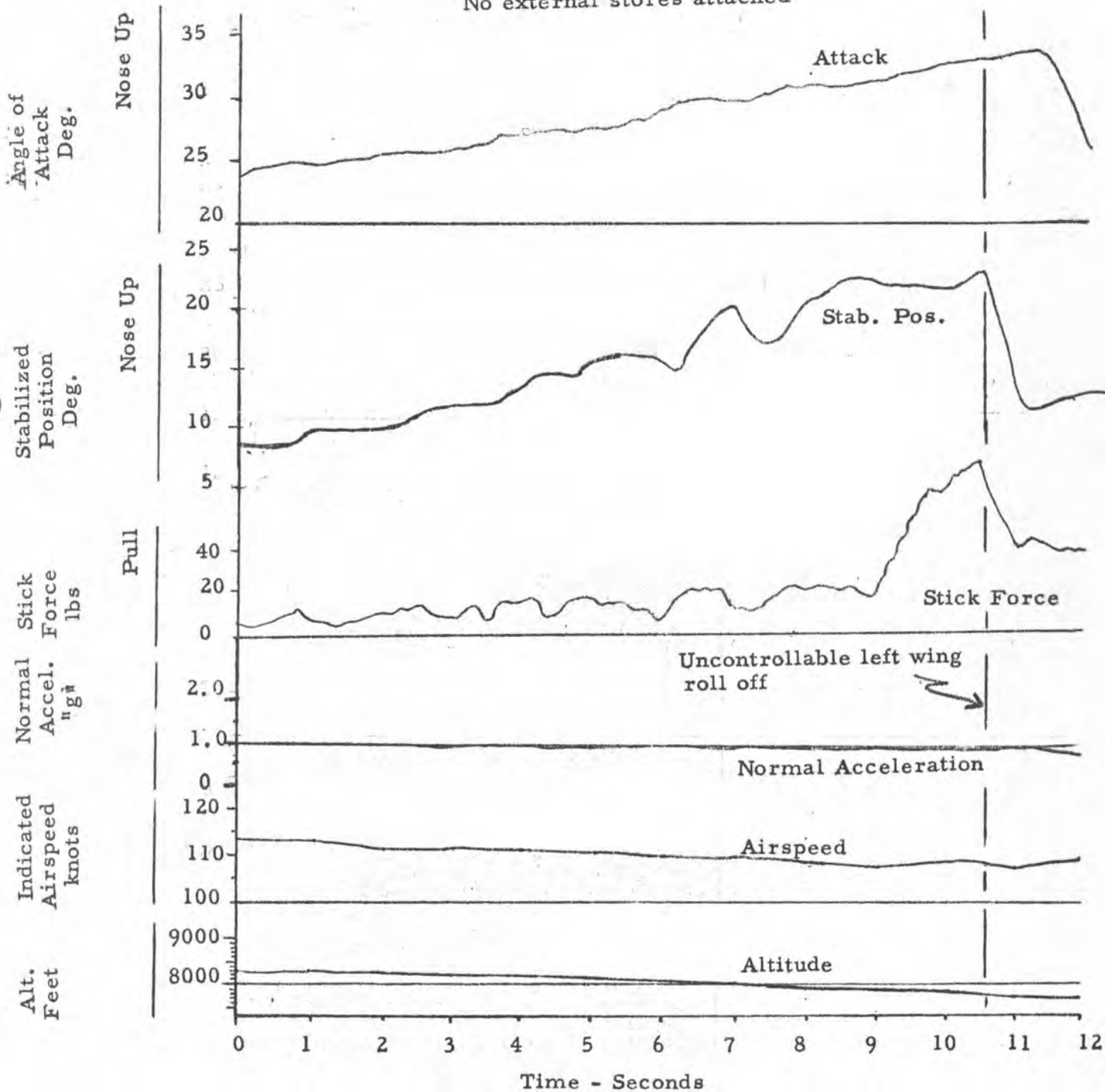


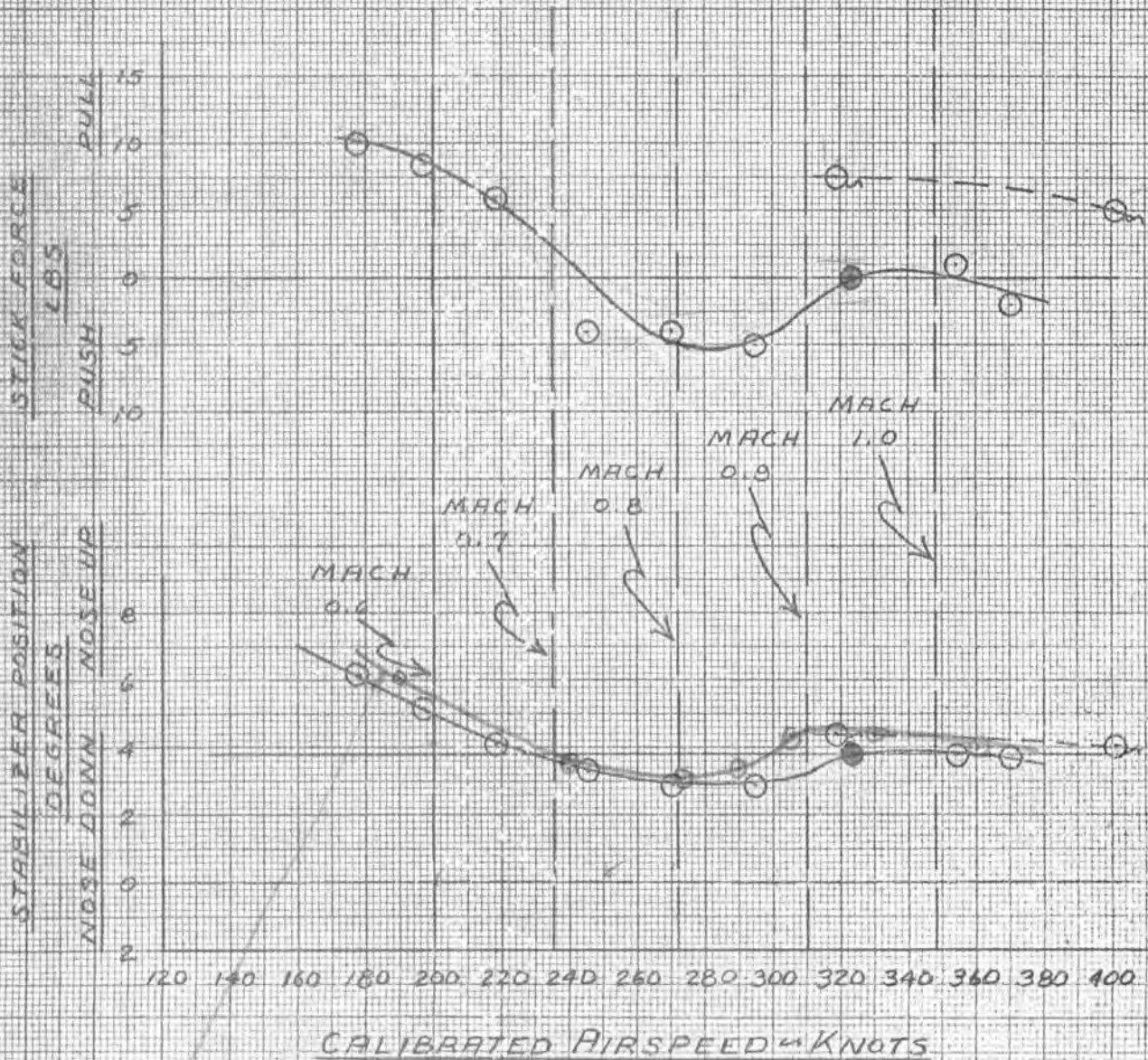
FIG No 35

STATIC LONGITUDINAL STABILITY
 YF-100A USAF No 52-5754
 POWER CONFIGURATION (A/B OFF)
 NO EXTERNAL STORES ATTACHED

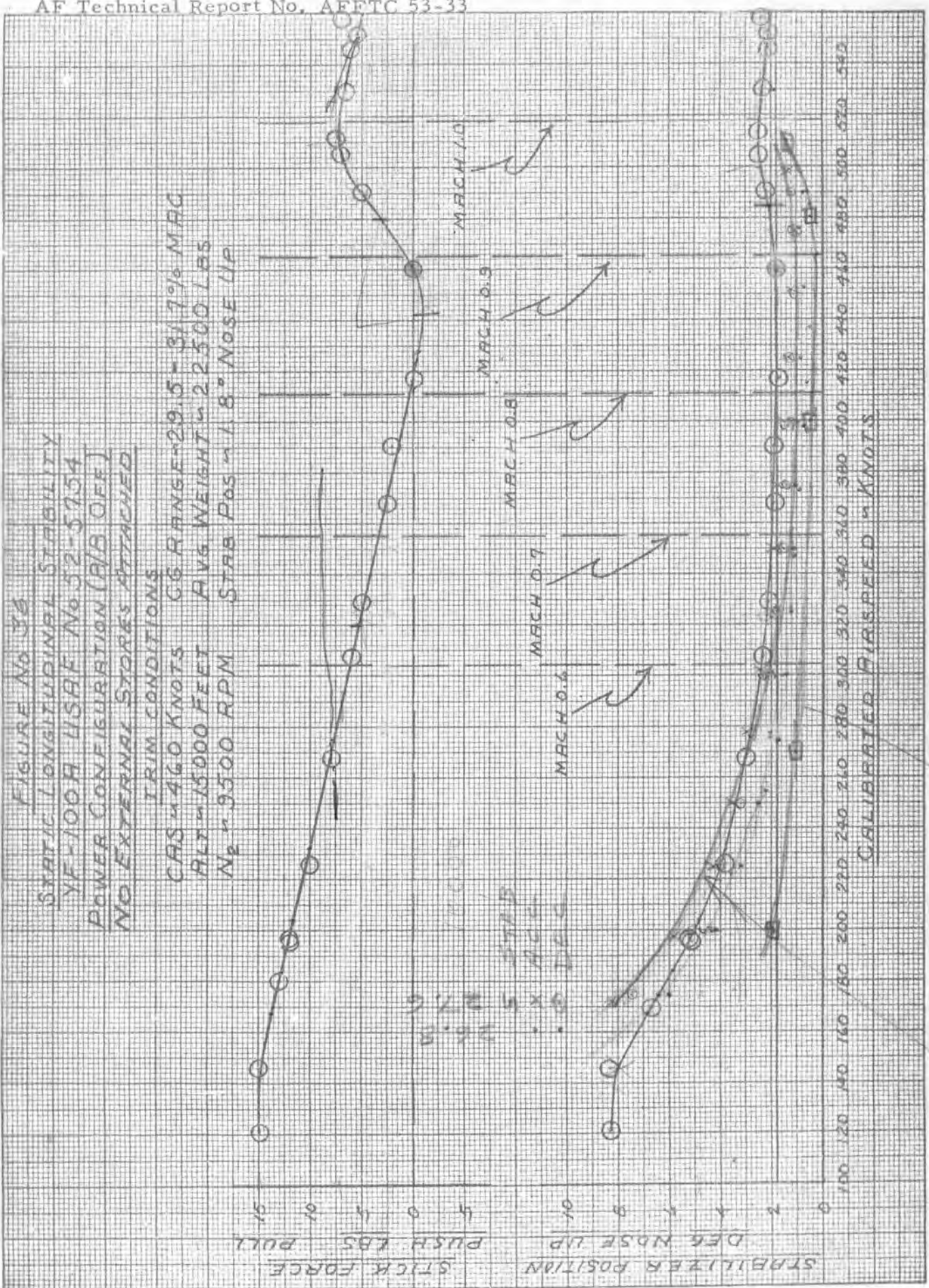
TRIM CONDITIONS

CAS = 323.5 KNOTS CG RANGE 29.5-31.7%MAC
 ALT = 35000 FEET AVG WEIGHT = 22000 LBS
 N₂ = 9240 RPM STAB POS = 3.8° NOSE UP

NOTE * TAILS INDICATE A/B ON



40,000' - fwd.



Next Point 44%
 but slope too low.

35.5% - 10,000'

27% - 10,000'

6-5-7

FIG No 37
STATIC LONGITUDINAL STABILITY
YF-100R USAF No 52-5754
POWER APPROACH CONFIGURATION
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS
 IAS = 159 KNOTS CG RANGE 29.5-31.7% MAC
 ALT = 10000 FEET AVG WEIGHT = 20300 LBS
 N₂ = 9230 RPM STAB POS = 7.7° NOSE UP

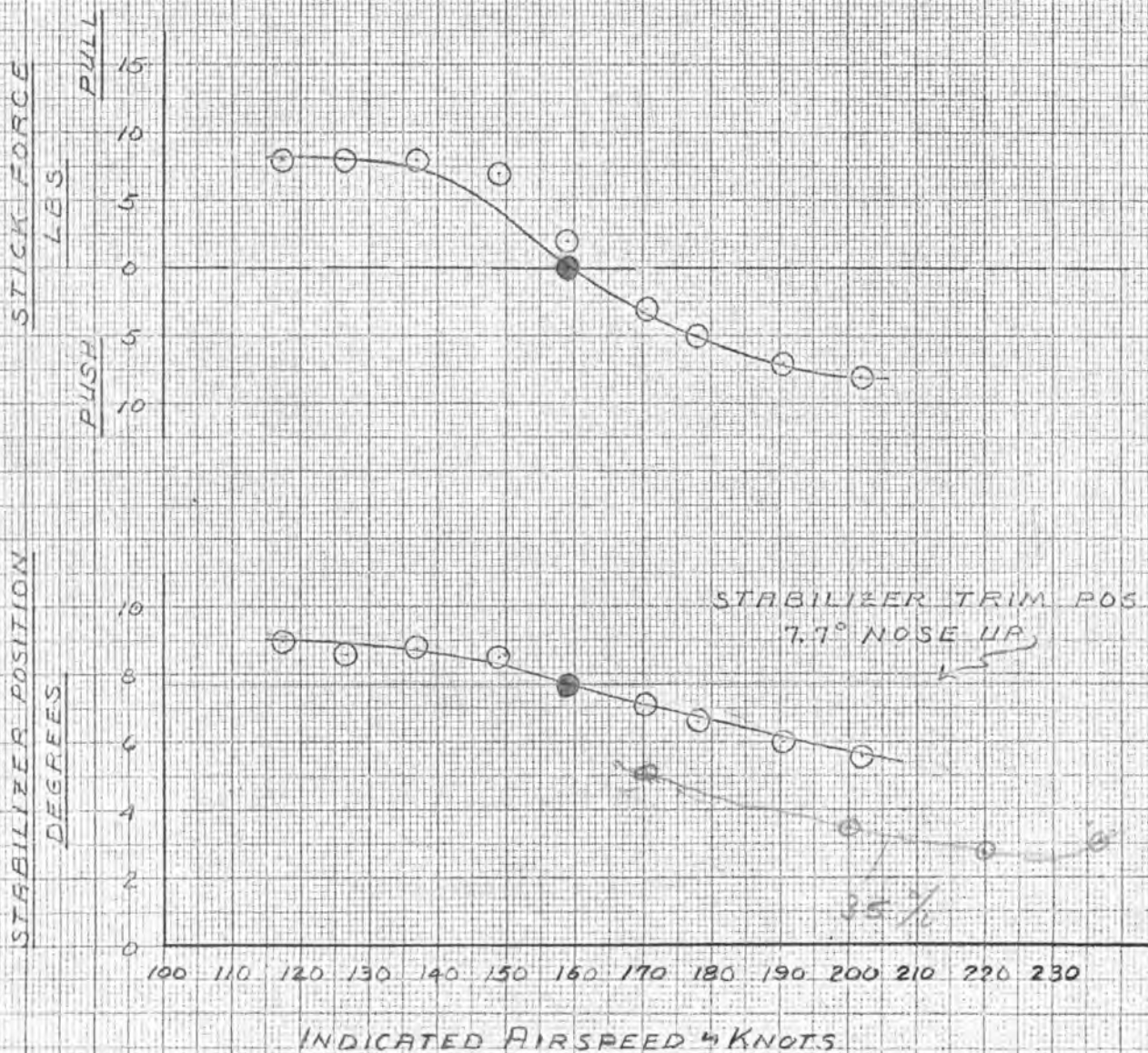


Figure No. 38

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 31.0 o/o MAC Weight 22,800 lbs
 Ave N₂ 9070 RPM Stabilizer 5.7° N.U.

No external stores attached

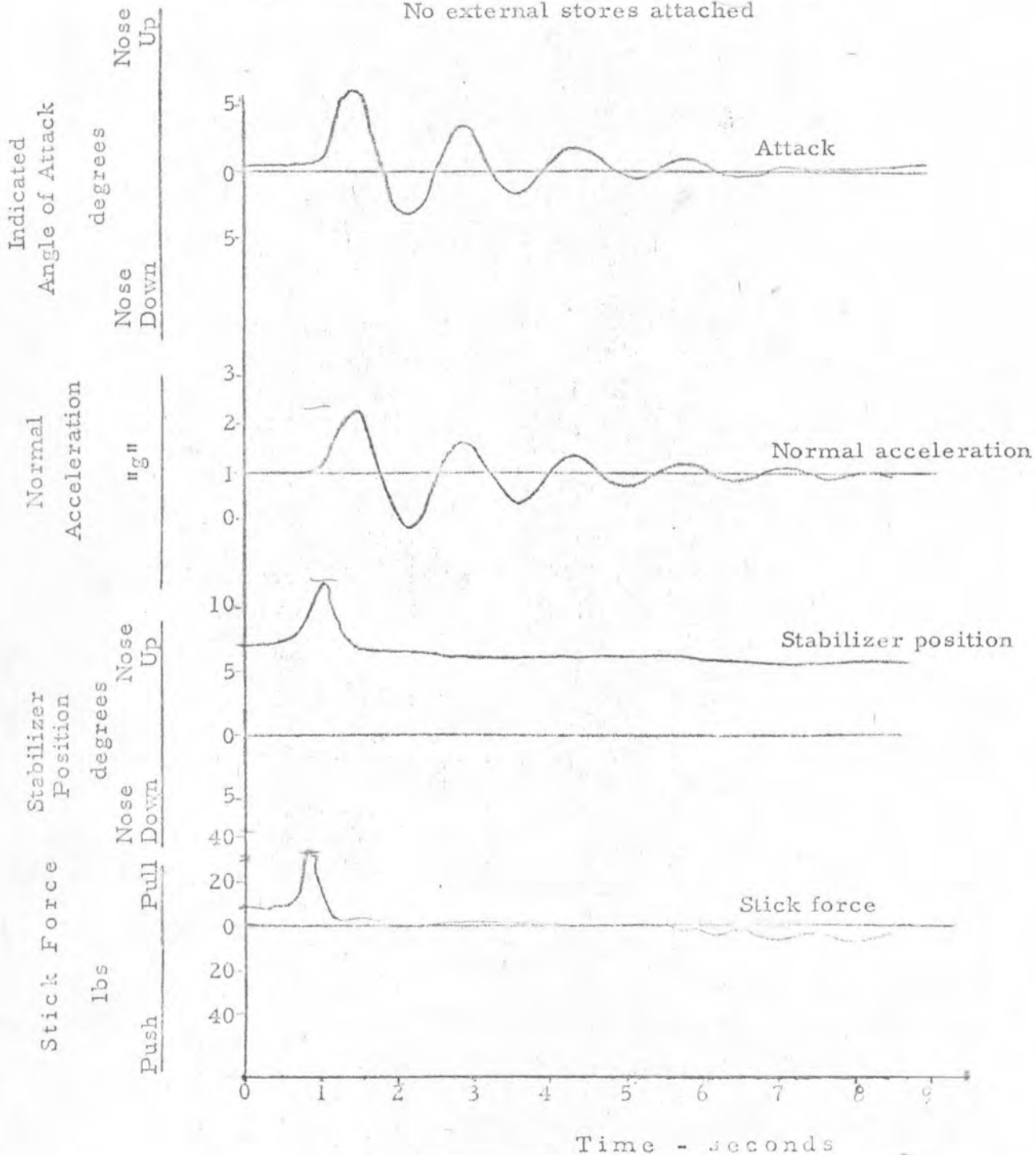


Figure No. 39

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 31.0 % MAC Weight 22,800 lbs
 Ave N₂ 9070 RPM Stabilizer 5.7° N.U.

No external stores attached

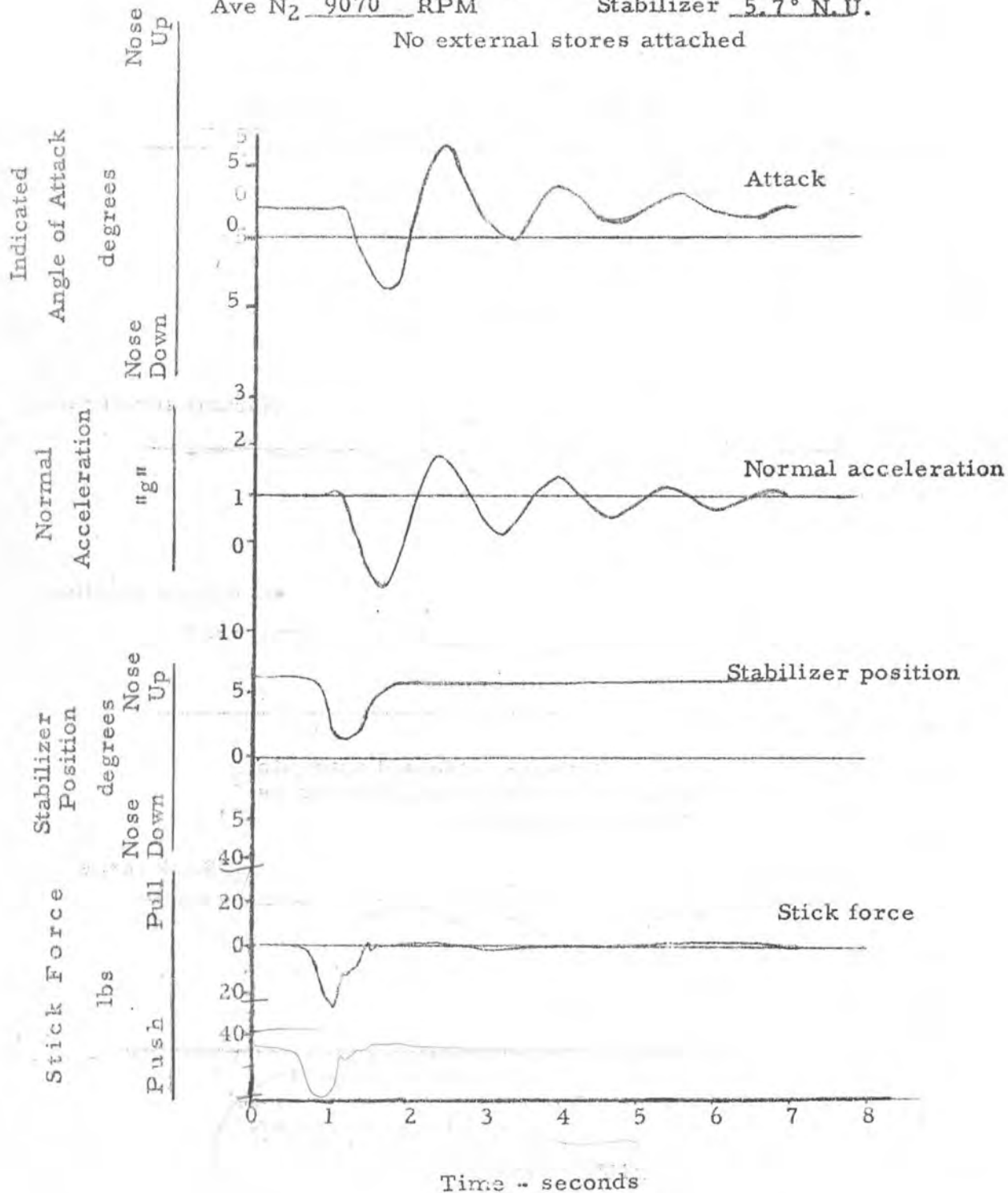


Figure No. 40

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 31.0 % MAC Weight 22,800 lbs
 Ave N₂ 9070 RPM Stabilizer 5.7° N.U.

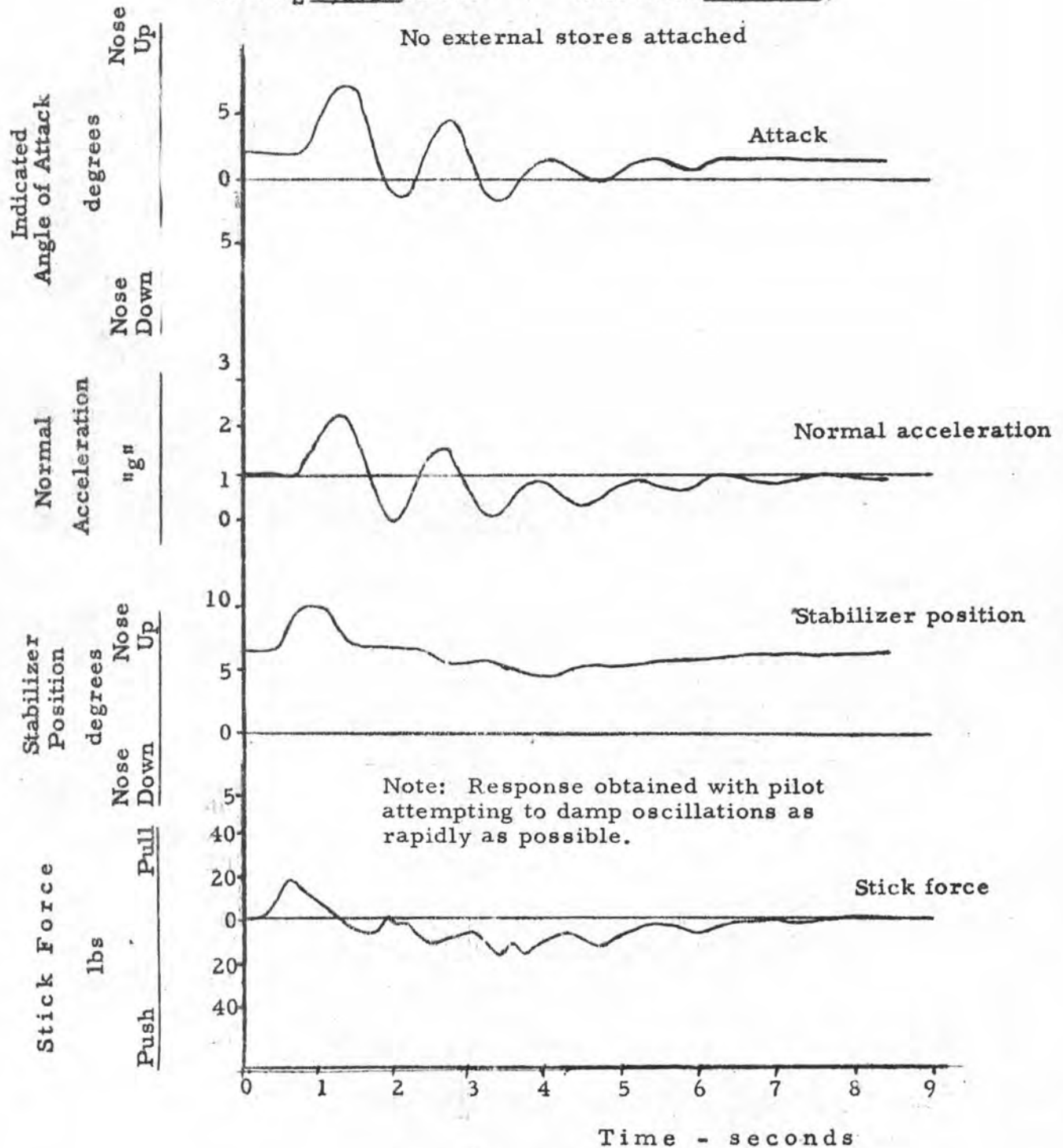


Figure No. 41

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 31.0 % MAC Weight 22,800 lbs
 Ave N₂ 9070 RPM Stabilizer 5.7° N.U.

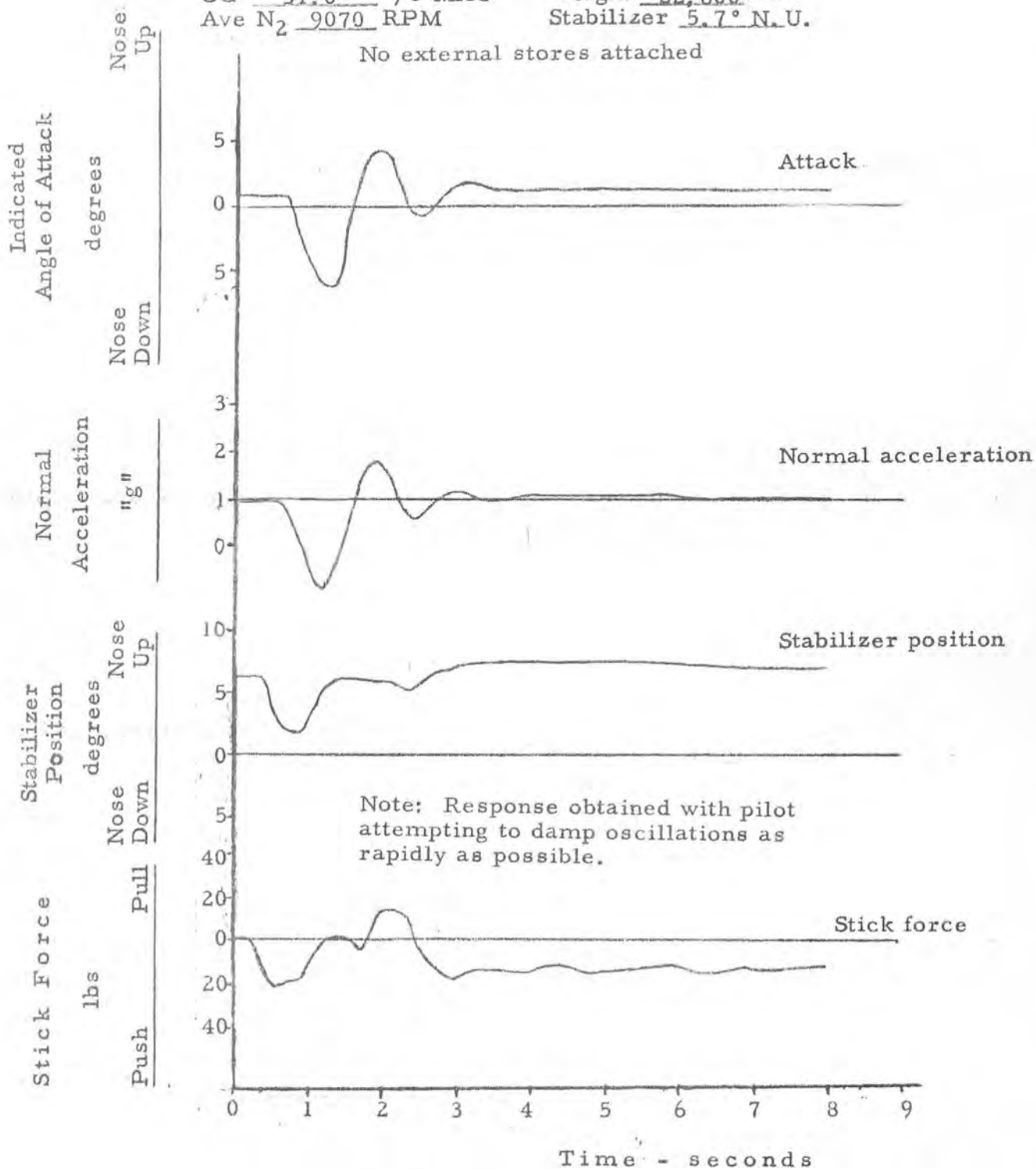


Figure No. 42

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS	<u>475</u>	knots	Altitude	<u>10,500</u>	feet
CG	<u>31.2</u>	% MAC	Weight	<u>23,600</u>	lbs
Ave N ₂	<u>9550</u>	RPM	Stabilizer	<u>1.7° N.U.</u>	

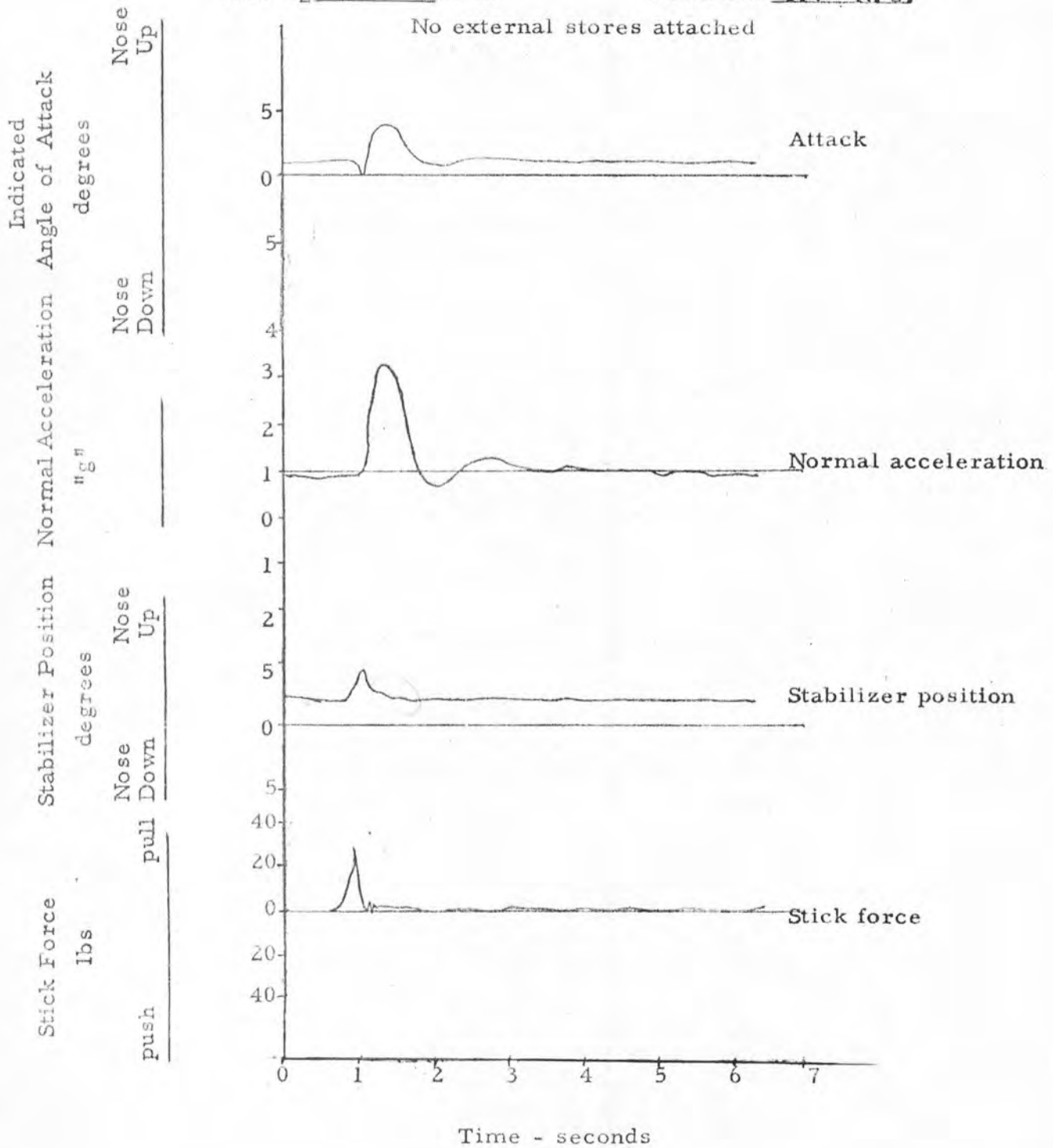


Figure No. 43

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 31.2 o/o MAC Weight 23,600 lbs
 Ave N₂ 9550 RPM Stabilizer 1.7° N.U.

No external stores attached

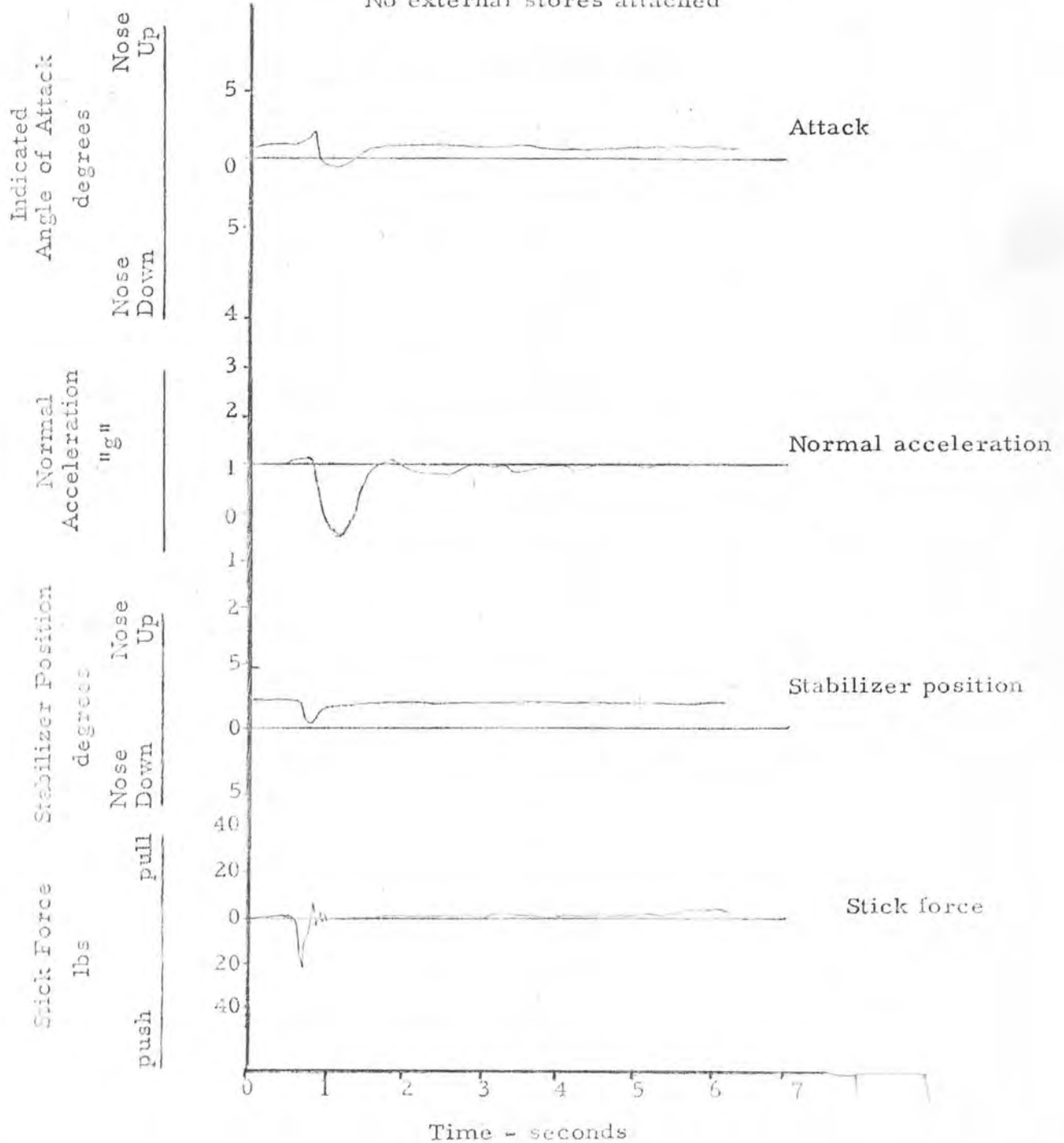


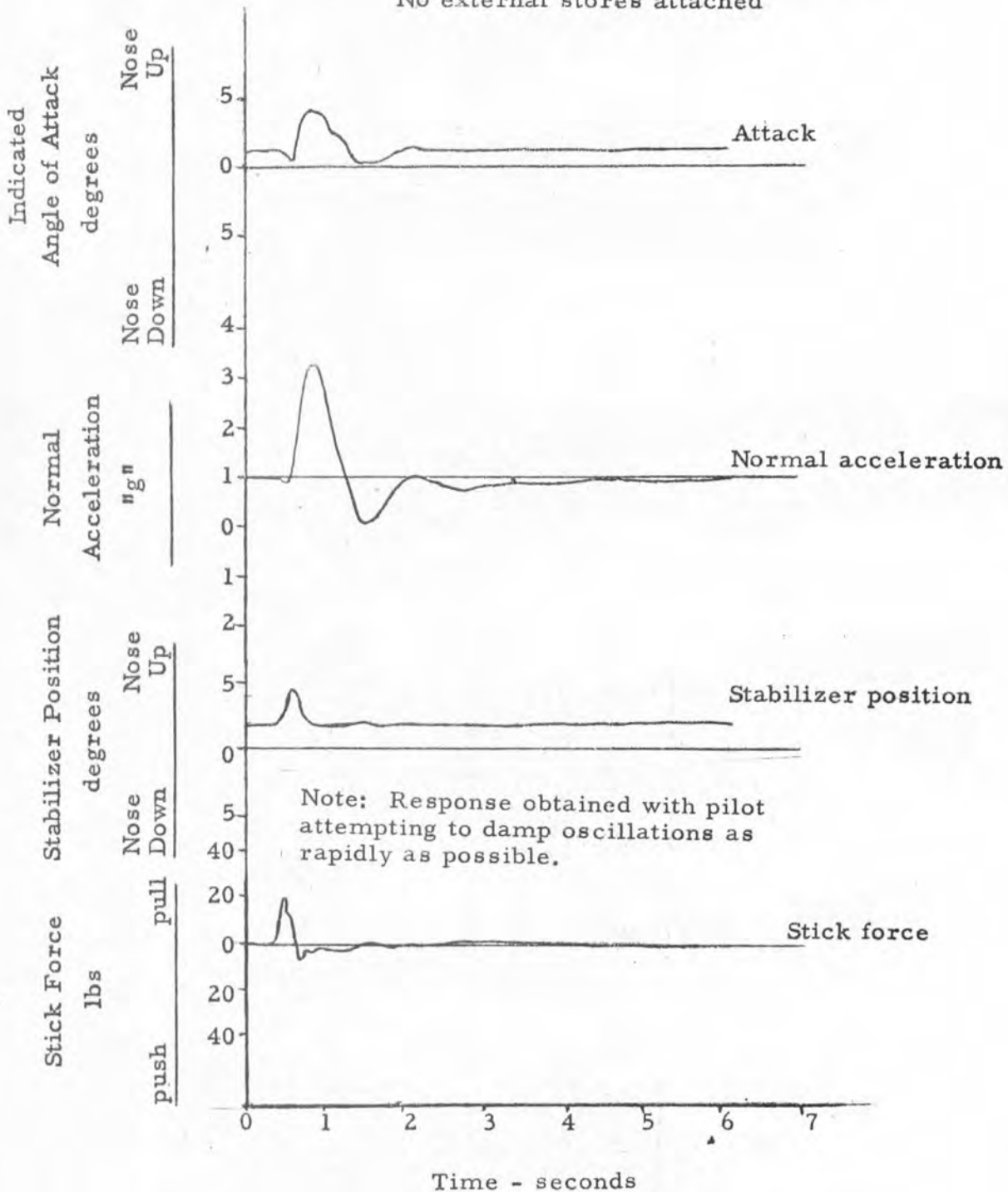
Figure No. 44

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B off)

TRIM CONDITIONS

CAS	<u>475</u>	knots	Altitude	<u>10,500</u>	feet
CG	<u>31.2</u>	% MAC	Weight	<u>23,600</u>	lbs
Ave N ₂	<u>9550</u>	RPM	Stabilizer	<u>1.7°</u>	N.U.

No external stores attached



Time - seconds

APPENDIX I

Figure No. 45

DYNAMIC LONGITUDINAL STABILITY
 Power Configuration (A/B off)
 YF-100A, USAF No. 52-5754

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 31.2 % MAC Weight 23,600 lbs
 Ave N₂ 9550 RPM Stabilizer 1.7° N. U.

No external stores attached

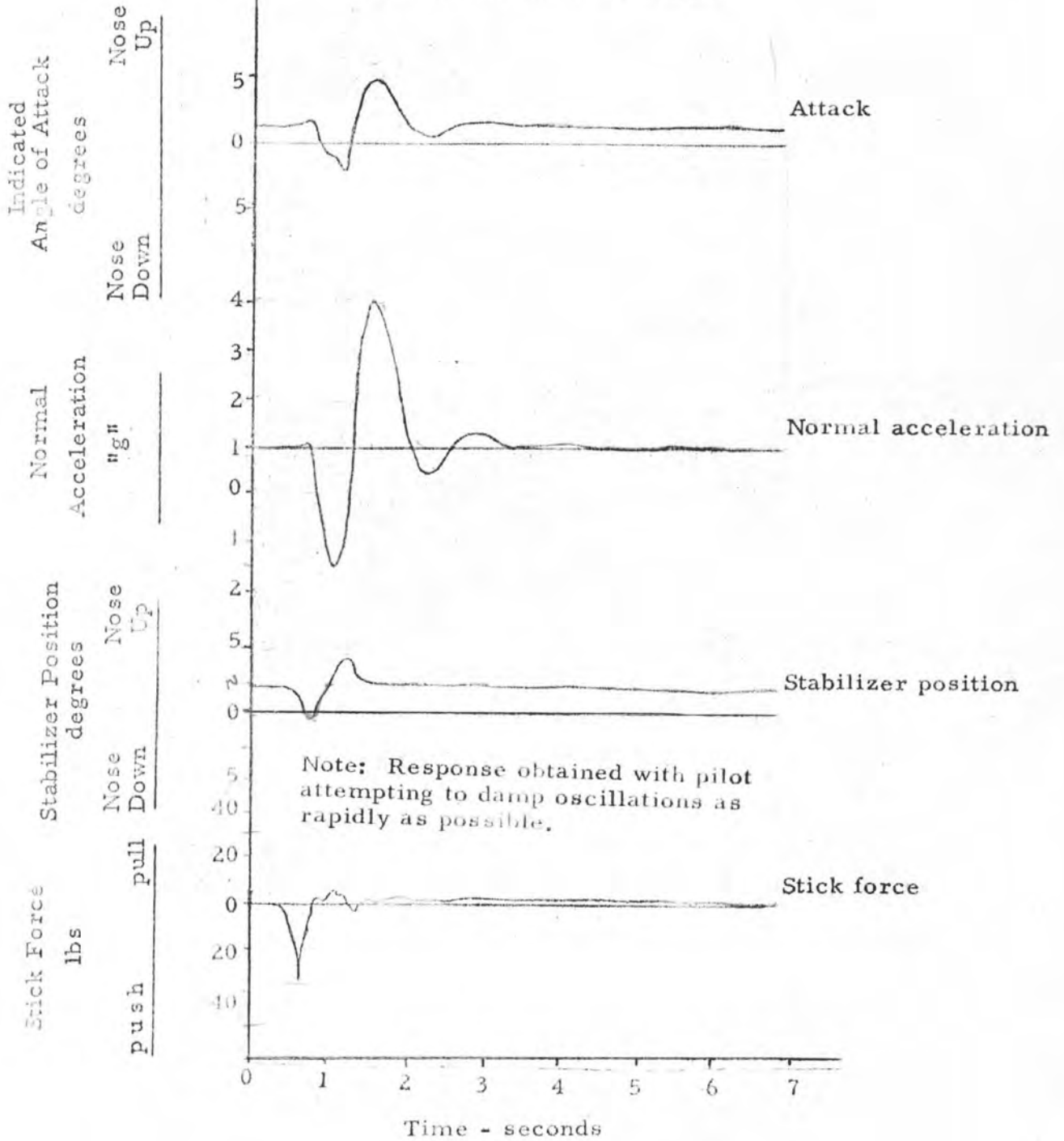
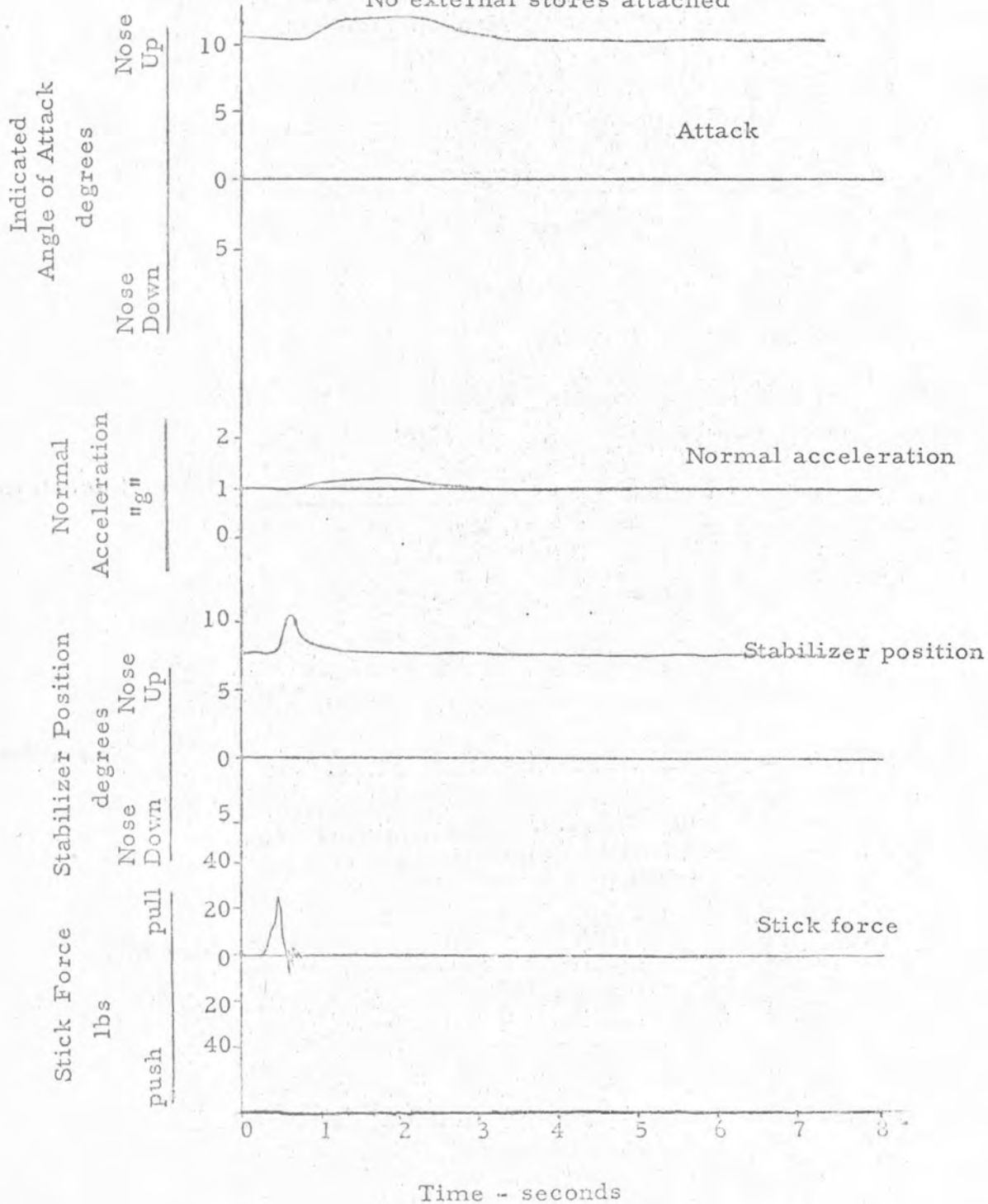


Figure No. 46

DYNAMIC LONGITUDINAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Approach Configuration Controls Free
 TRIM CONDITIONS

CAS 164 knots Altitude 9,900 feet
 CG 30.1 % MAC Weight 21,450 lbs
 Ave N₂ 9240 RPM Stabilizer 7.7° N.U.
 No external stores attached



Time - seconds

APPENDIX I

Figure No. 47

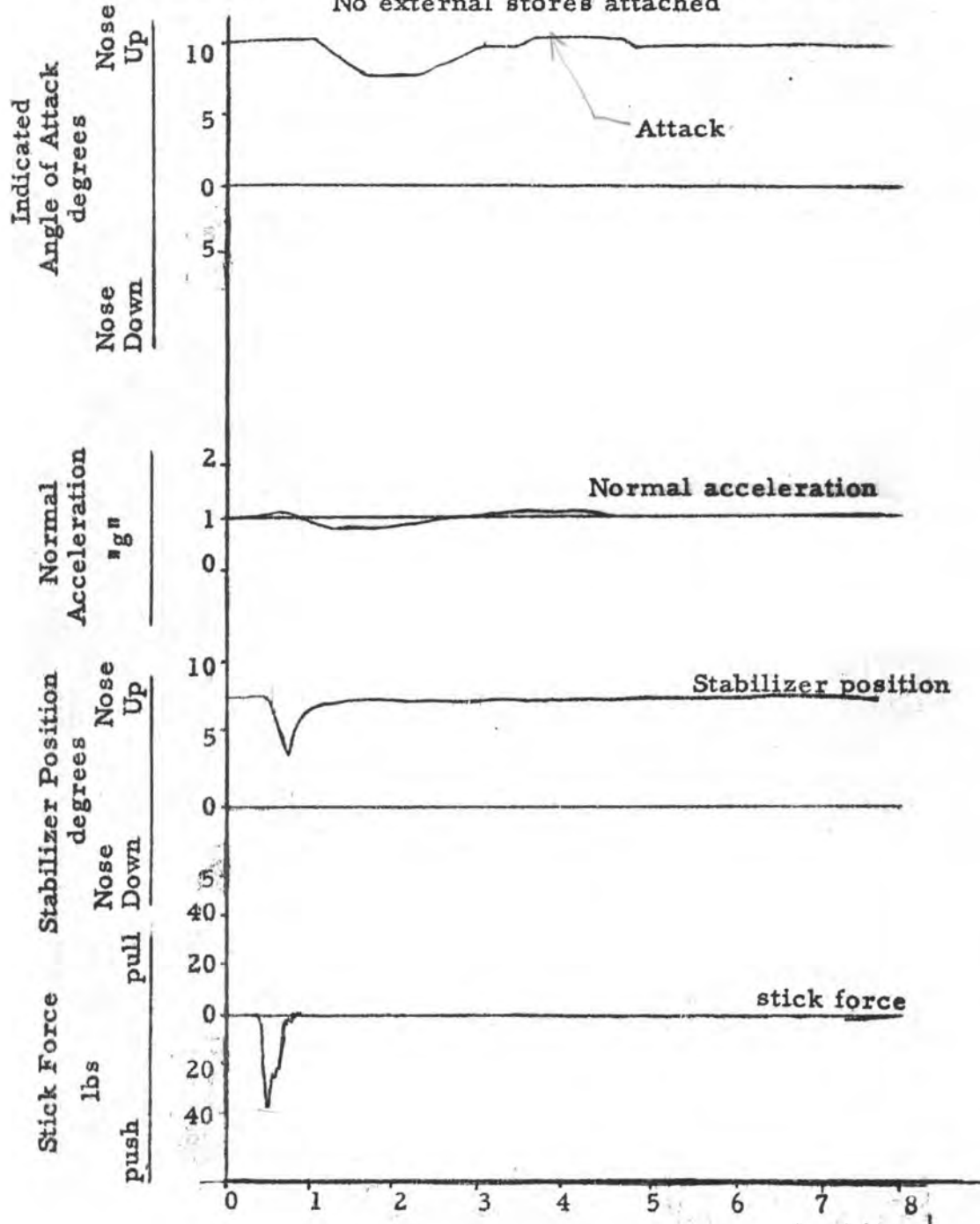
DYNAMIC LONGITUDINAL STABILITY
YF-100A, USAF No. 52-5754

Power Approach Configuration, Controls Free

CRUISE LANDING
TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
CG 30.1 % MAC Weight 21,450 lbs
Ave N₂ 9240 RPM Stabilizer 7.7° N.U.

No external stores attached



Time - seconds

APPENDIX I

FIG No 48

MANUEVERING FLIGHT CHARACTERISTICS
 YF-100A USAF No 52-5754
 POWER CONFIGURATION (A/B ON)
 NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS			
MACH	262 KNOTS;	CG RANGE	30.5 % MAC
ALT	45800 FEET;	AVG WEIGHT	25400 LBS
N ₂	3100 RPM;	STAB POS	5.6 ° NOSE UP

NOTE: ABRUPT PULL-UP DATA INDICATE 3.2g
 AT 27 POUNDS STICK FORCE FOR 13° NOSE UP
 STABILIZER

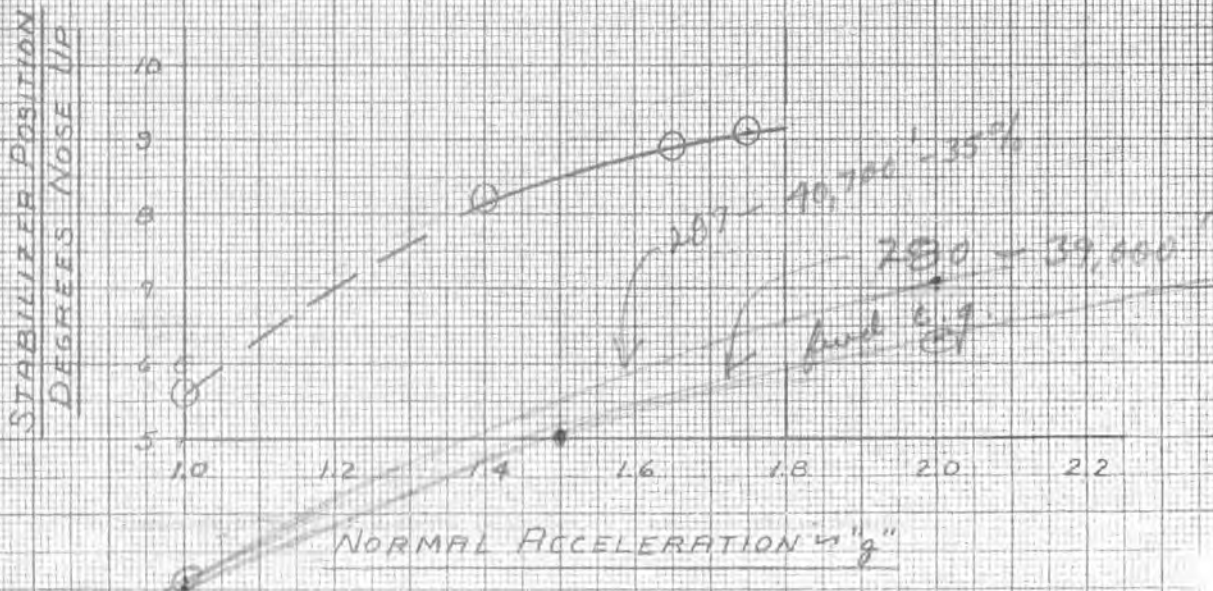
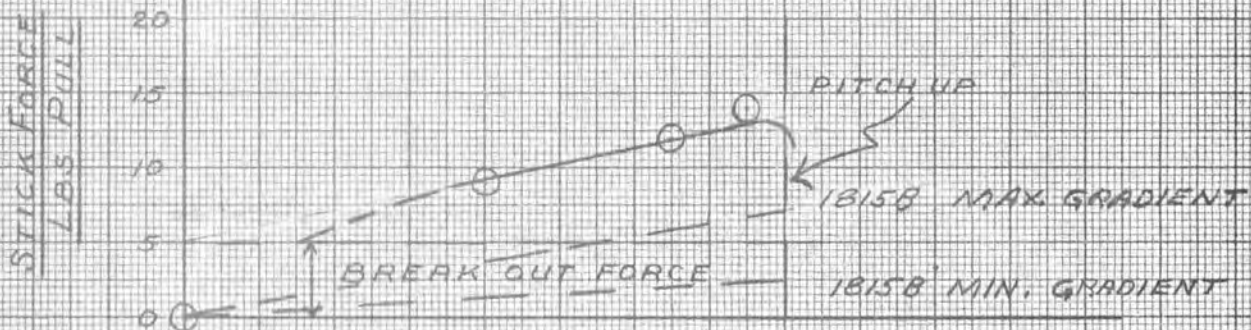


FIG NO 49
MANEUVERING FLIGHT CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION (A/B ON)
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CBS	271	KNOTS	CG RANGE	29.9	% MPC
ALT	4000	FEET	AVG WEIGHT	22000	LBS
N ₂	9055	RPM	STAB POS	5.3	° NOSE UP

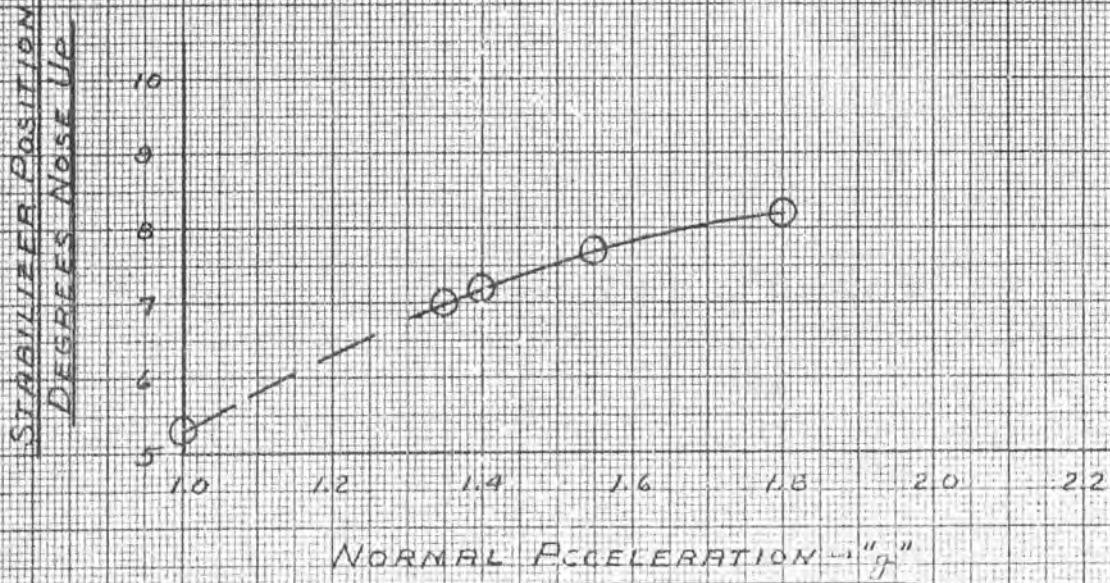
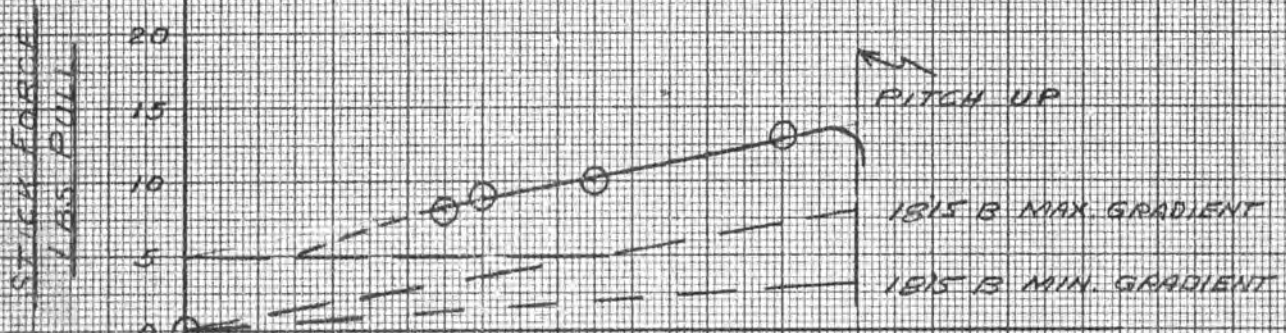


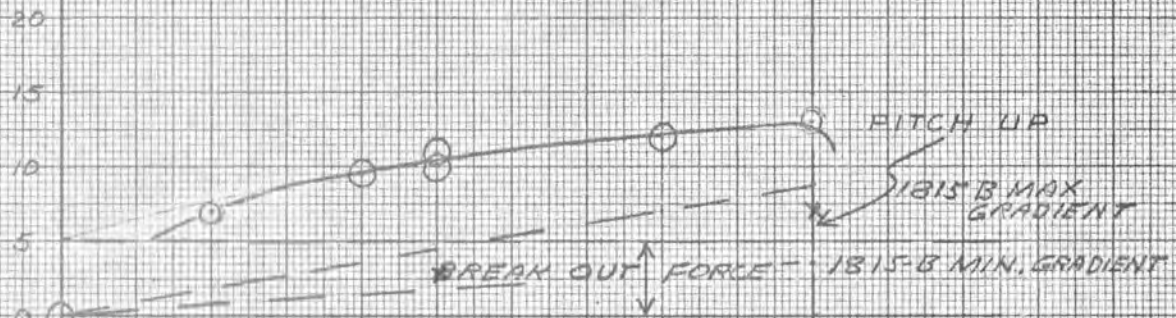
FIG No 50

MANUEVERING FLIGHT CHARACTERISTICS
 YF-100A USAF No 52-5754
 CRUISE CONFIGURATION (A/B OFF)
 NO EXTERNAL STORES ATTACHED

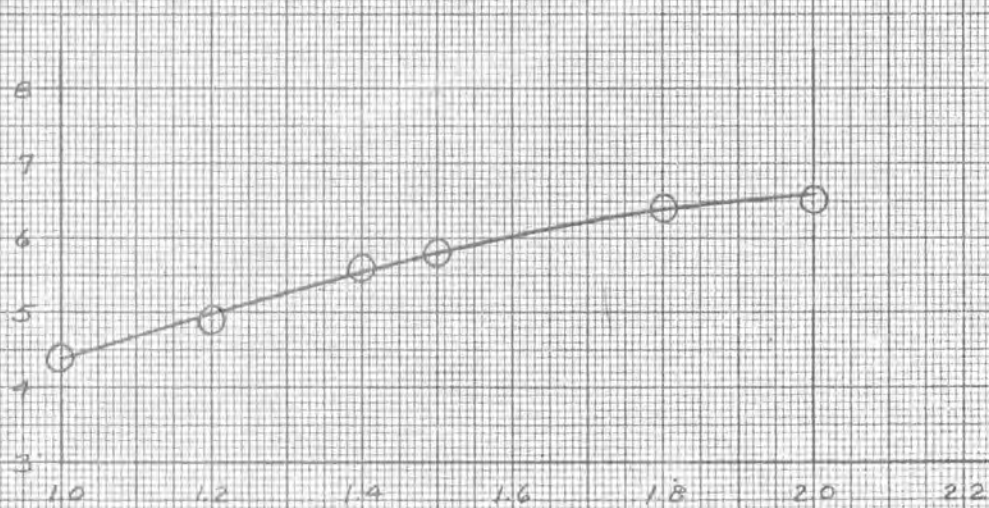
TRIM CONDITIONS

CAS	254 KNOTS	CG RANGE	29.8	% MAC
ALT	45900 FEET	AVG WEIGHT	21670	LBS
N ₂	9000 RPM	STAB POS	4.4	° NOSE UP

STICK FORCE
LBS PULL



STABILIZER POSITION
DEGREES NOSE UP



NORMAL ACCELERATION "g"

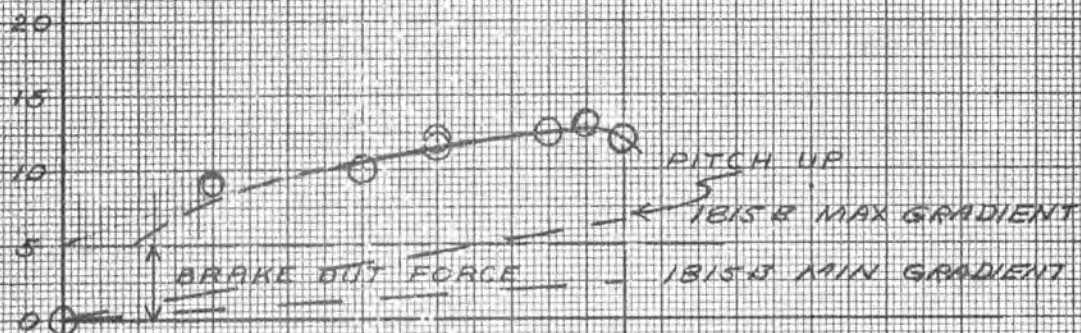
FIG No 51

MANEUVERING FLIGHT CHARACTERISTICS
 YF-100A USAF No 52-5754
 CRUISE CONFIGURATION (A/B OFF)
 NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS	237 KNOTS	CG RANGE	30.35	% _D MAC
ALT	45700 FEET	AVG WEIGHT	21300	LBS
N ₂	8900 RPM	STAB POS	3.8	° NOSE UP

STICK FORCE
 LBS PULL



STABILIZER POSITION
 DEGREES NOSE UP

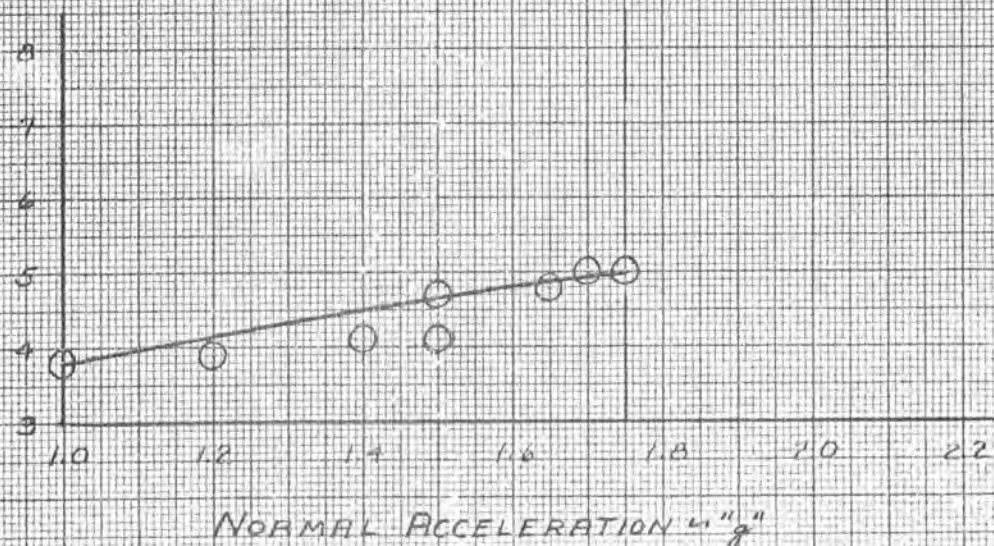


FIG No 52

MANEUVERING FLIGHT CHARACTERISTICS
 YF-100A USAF No 52-5754
 POWER CONFIGURATION (A/B OFF)
 NO EXTERNAL STORES ATTACHED

ME, 29 TRIM CONDITIONS

CAS 484 KNOTS; CG RANGE 29.5 TO 31.7% MAC

ALT 11950 FEET; AVG WEIGHT 23000 LBS

N₂ 9535 RPM; STAB POS. 1.8 ° NOSE UP

○ MANEUVERING FLIGHT

* ABRUPT PULL UPS

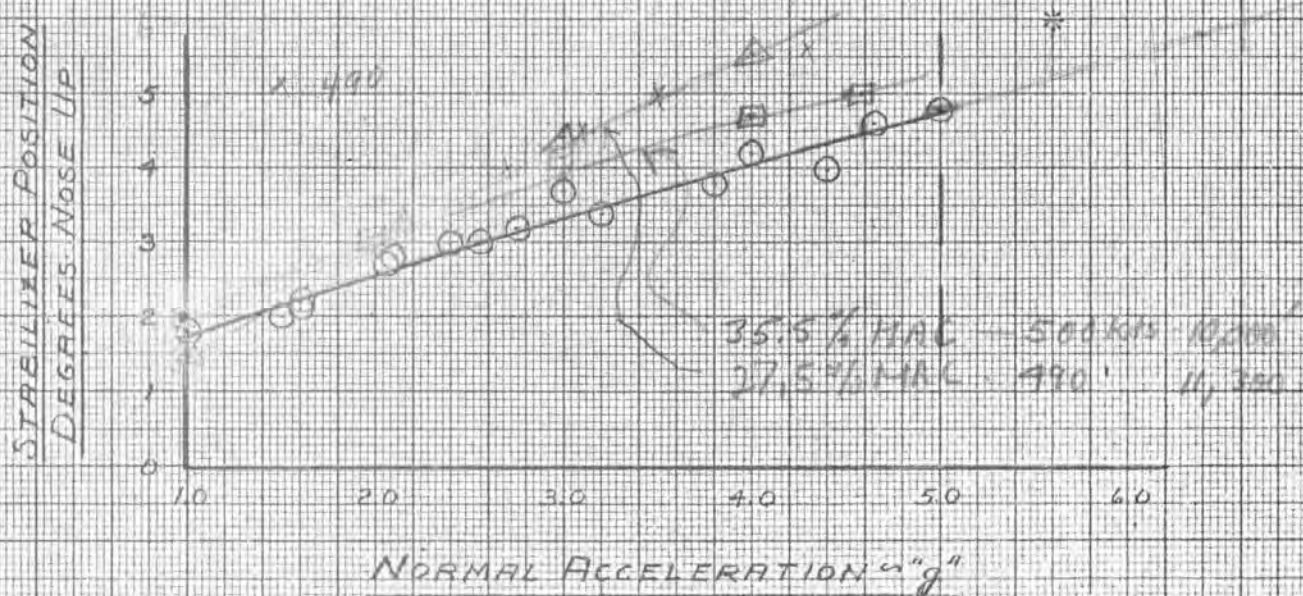
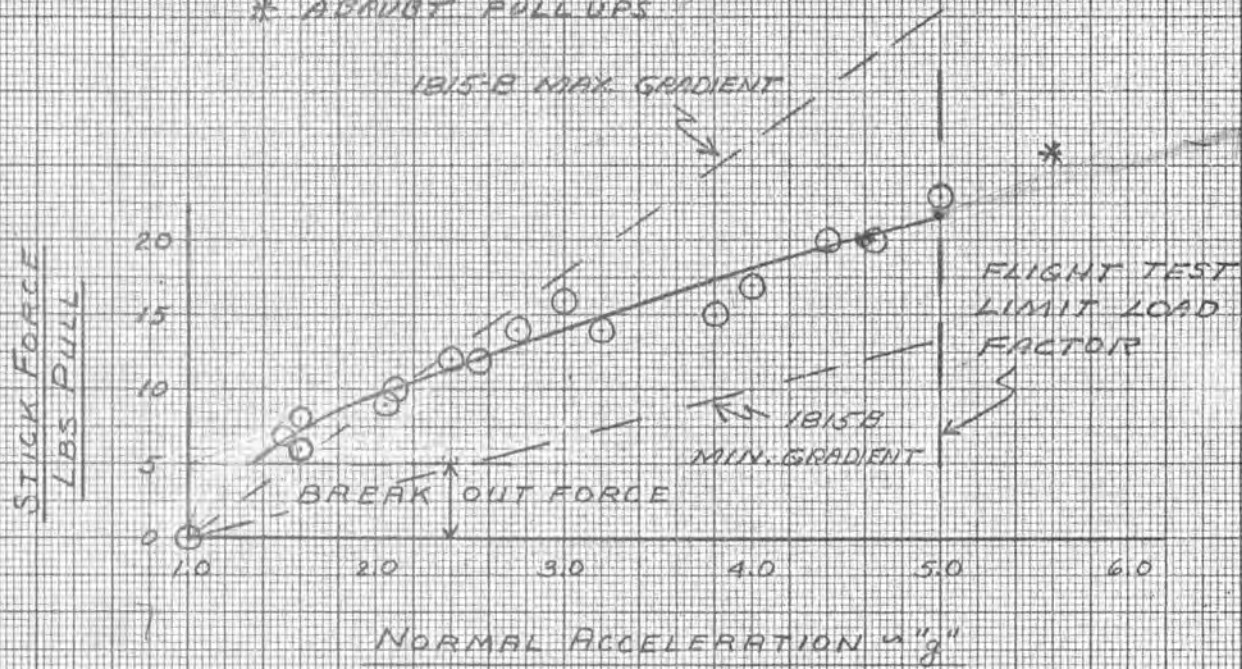


Fig No 53

MANUVERING FLIGHT CHARACTERISTICS
YF-100A USAF No 52-5754
POWER APPROACH CONFIGURATION (A/B OFF)
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

IAS	159	KNOTS	CG RANGE	31	%MRC
ALT	11190	FEET	AVG WEIGHT	21900	LBS
N ₂	9220	RPM	STAB POS	7.9	°NOSE UP

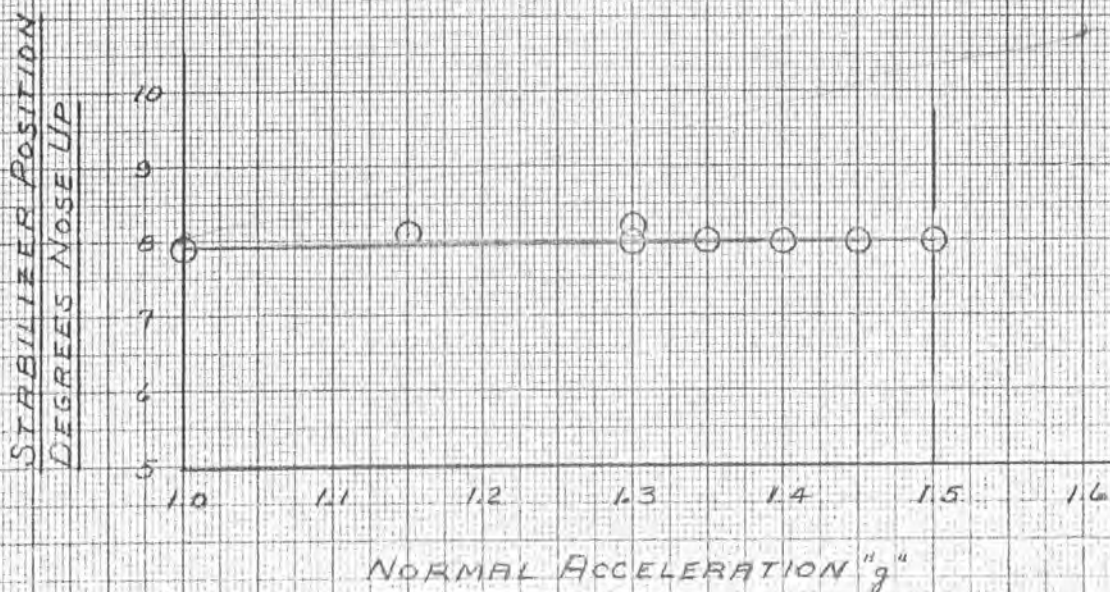
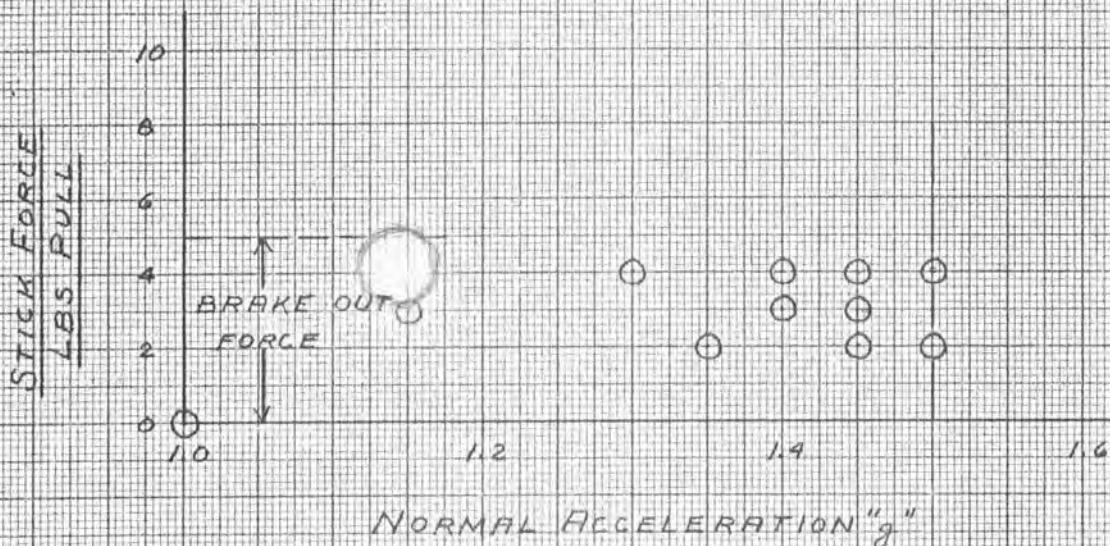
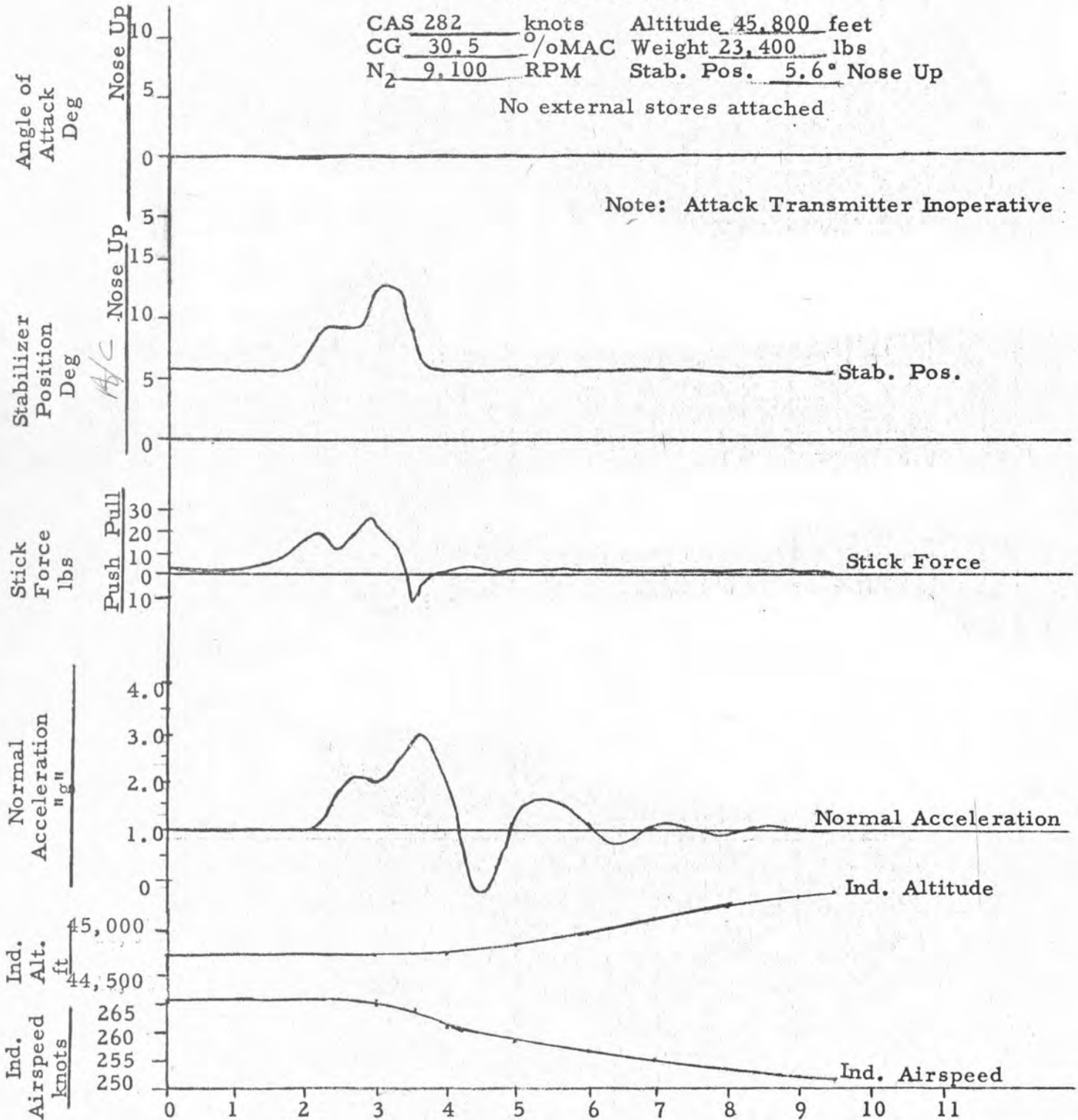


Figure No. 54
TIME HISTORY OF AN ABRUPT PULL-UP
 YF-100A, USAF NO. 52-5754
 Power Configuration (A/B On)
 TRIM CONDITIONS

CAS 282 knots Altitude 45,800 feet
 CG 30.5 %MAC Weight 23,400 lbs
 N₂ 9,100 RPM Stab. Pos. 5.6° Nose Up

No external stores attached

Note: Attack Transmitter Inoperative



Time - Seconds

APPENDIX I

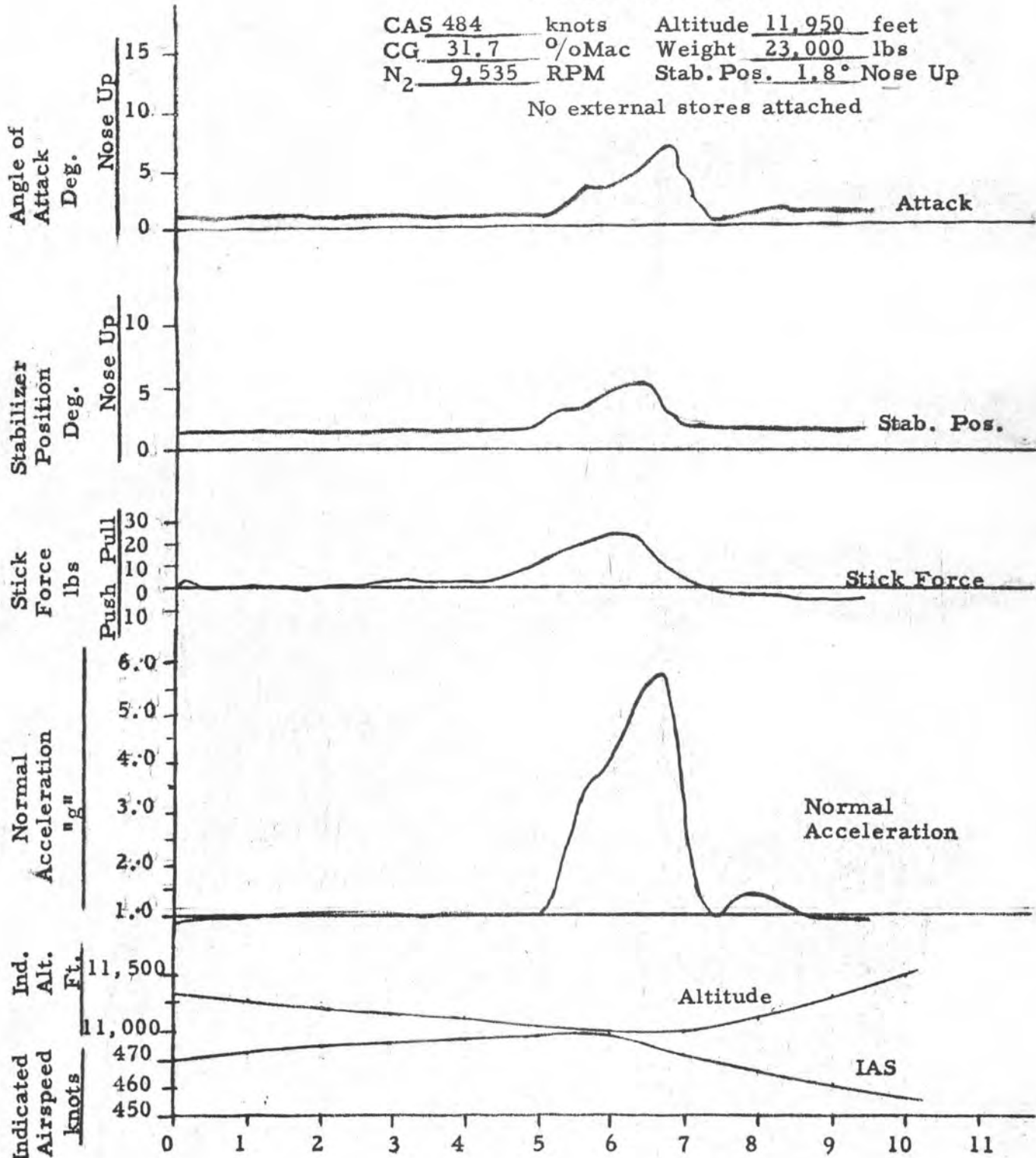
Figure No. 55

TIME HISTORY OF AN ABRUPT PULL-UP
 YF-100A, USAF No. 52-5754
 Power Configuration(A/B Off)

TRIM CONDITIONS

CAS 484 knots Altitude 11,950 feet
 CG 31.7 %oMac Weight 23,000 lbs
 N₂ 9,535 RPM Stab. Pos. 1.8° Nose Up

No external stores attached

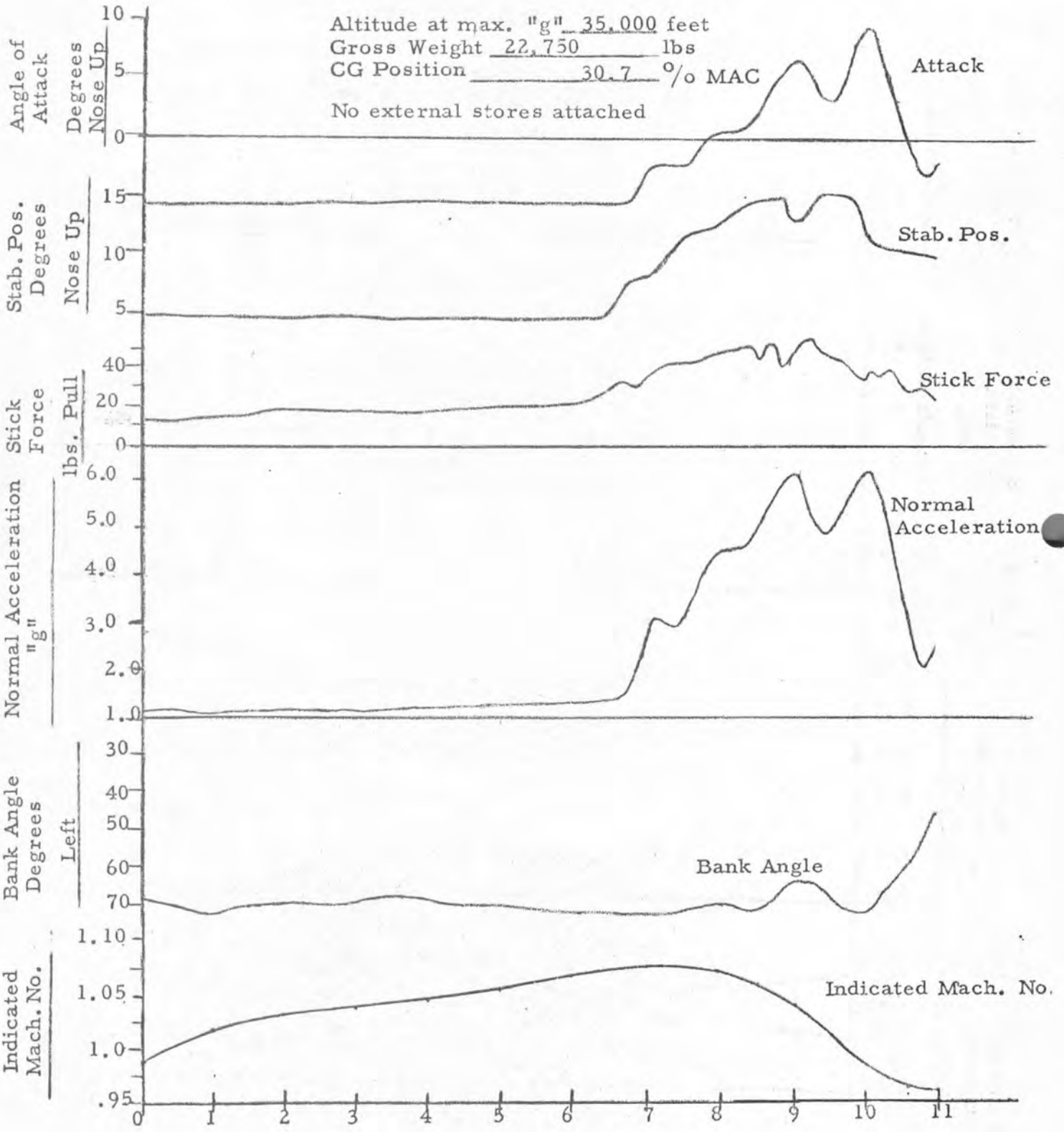


Time - Seconds

APPENDIX I

Figure No. 56

TIME HISTORY OF LIMIT "g" DIVING TURN
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On)



Time - Seconds
 APPENDIX I

Figure No. 57

TIME HISTORY OF LIMIT "g" DIVING TURN

YF-100A, USAF No. 52-5754

Power Configuration (A/B On)

Altitude at max. "g" 35,000 feet

Gross Weight 23,000 lbs

CG Position 31 % MAC

No external stores attached

Attack

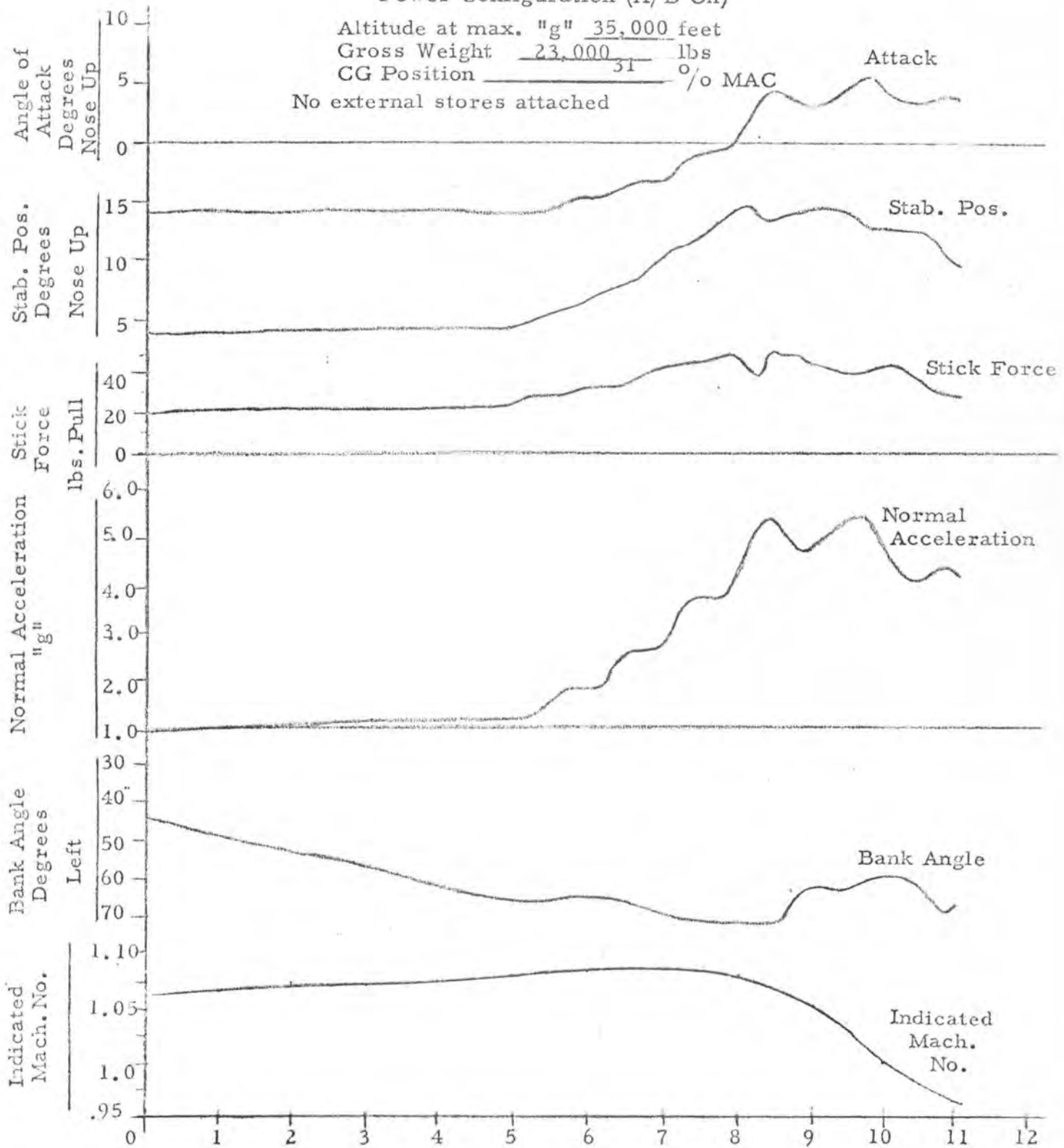
Stab. Pos.

Stick Force

Normal Acceleration

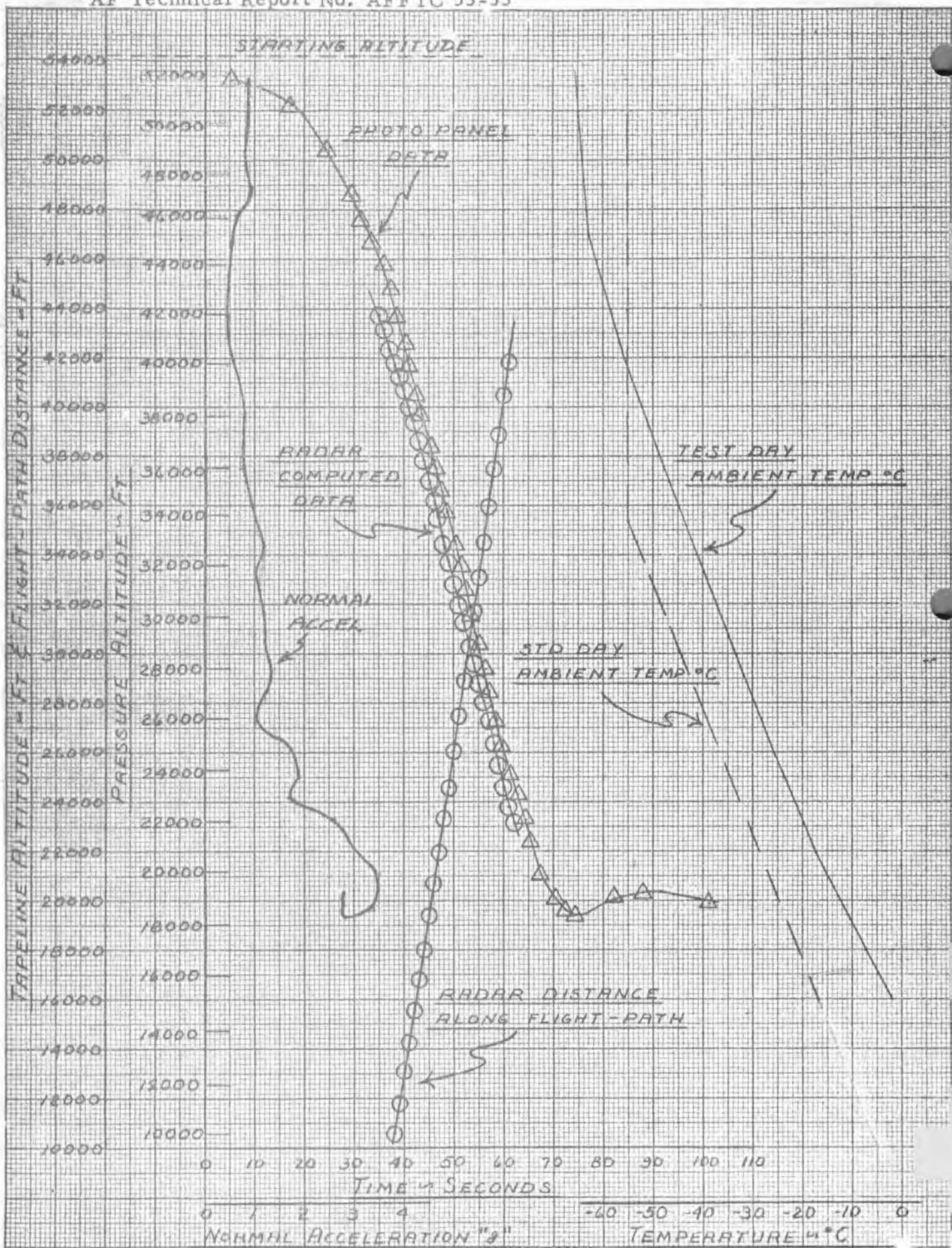
Bank Angle

Indicated Mach. No.



Time - Seconds

APPENDIX I



APPENDIX I

FIGURE NO. 59

STABILITY & CONTROL OF A HIGH SPEED DIVE

YF-100A USAF No 52-5754

POWER CONFIGURATION (A/B ON)

NO EXTERNAL STORES ATTACHED

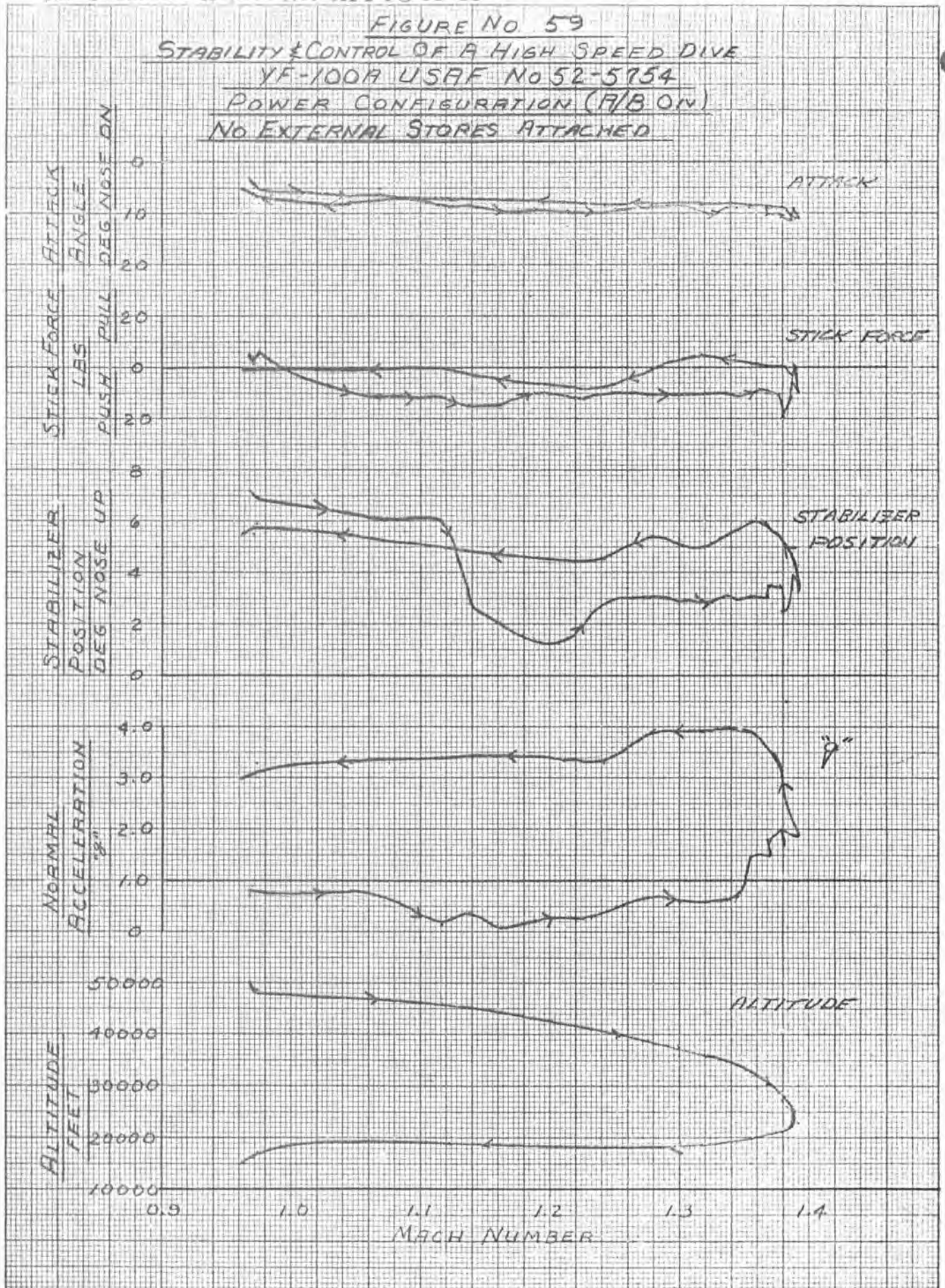
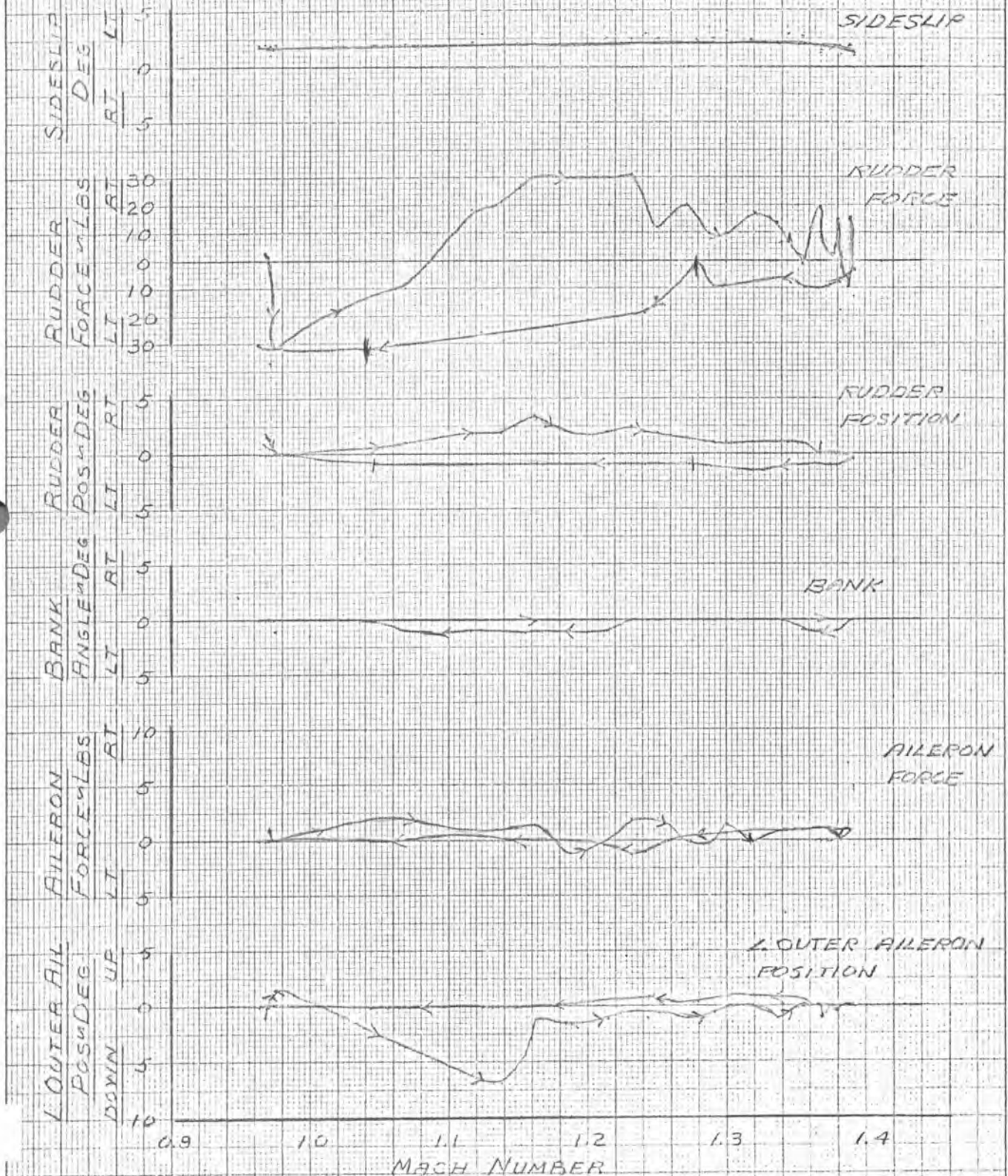


FIGURE NO. 59
TRIM CONDITIONS

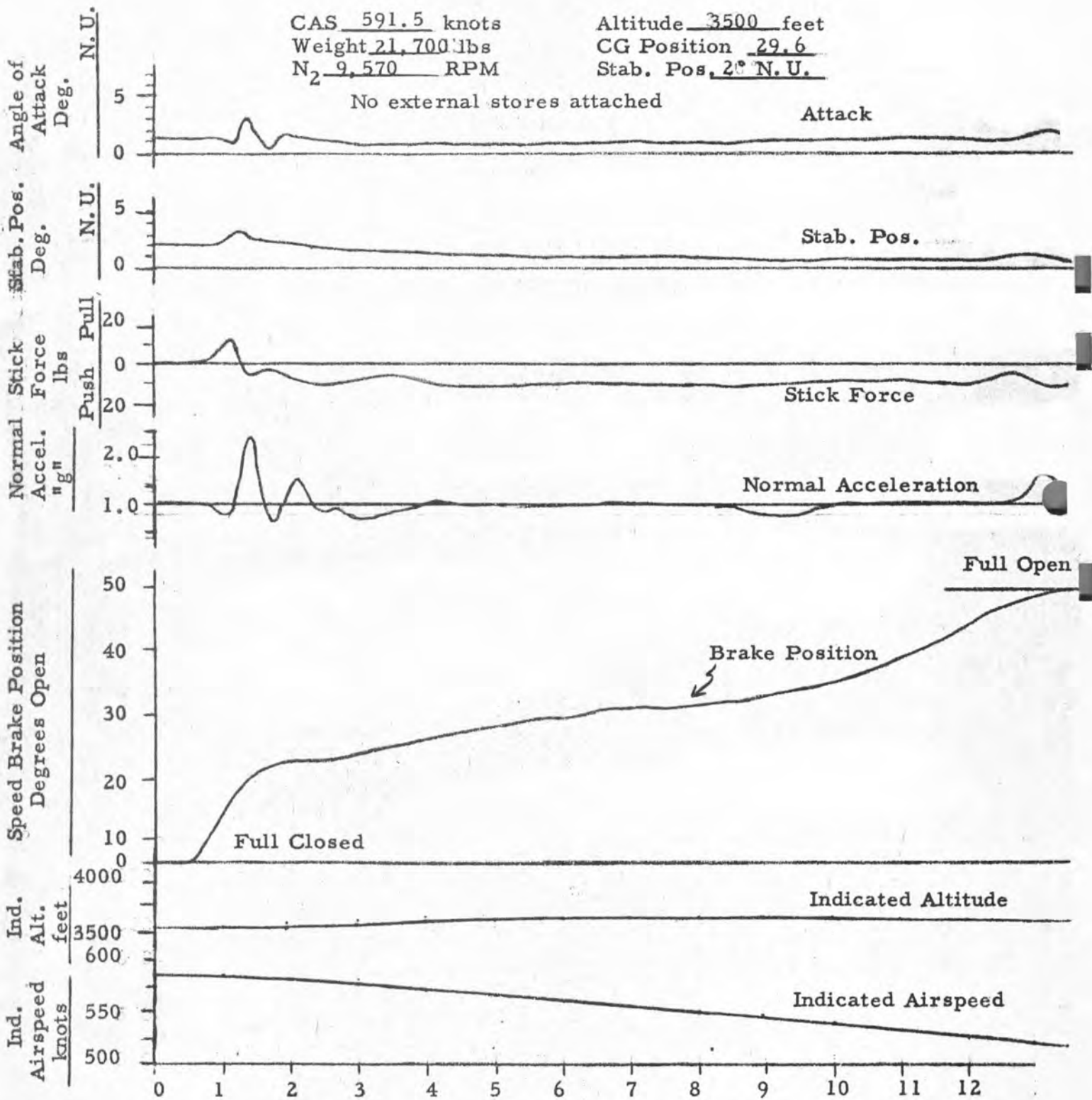
IAS 222 KNOTS CG 29.62 MAC RUDDER POS 1.3° RIGHT
 ALTITUDE 51200 FT N₂ 8070 RPM L. OUTER AIL. POS 0.8° DN
 WEIGHT 21700 LBS STAB POS 6.8° NOSE UP



TIME HISTORY OF SPEED BRAKE OPENING
YF-100A, USAF No. 52-5754
Power Configuration
TRIM CONDITIONS

CAS 591.5 knots Altitude 3500 feet
Weight 21,700 lbs CG Position 29.6
N₂ 9,570 RPM Stab. Pos. 2° N.U.

No external stores attached



Time - Seconds

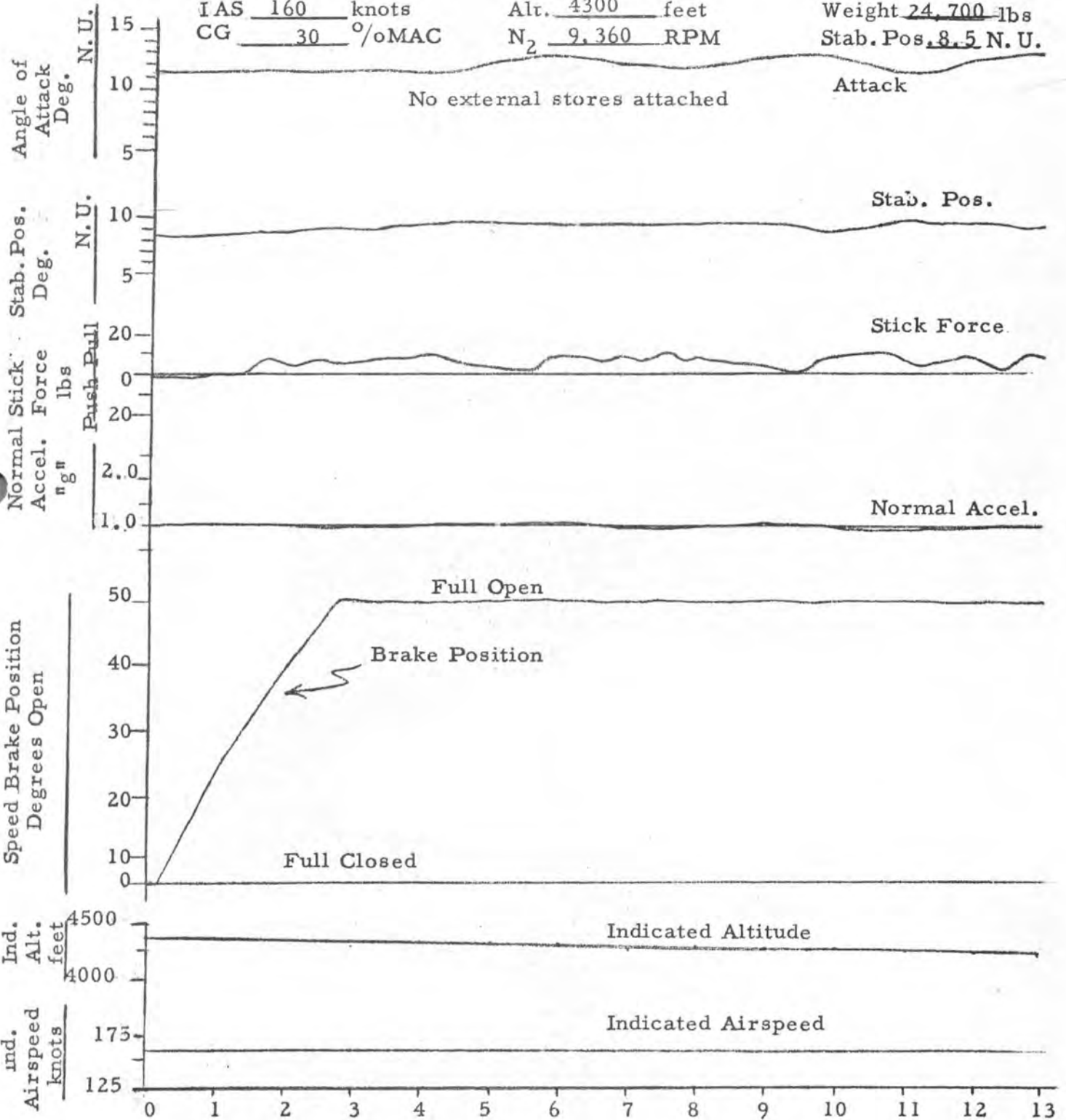
APPENDIX I

TIME HISTORY OF SPEED BRAKE OPENING

YF-100 A, USAF No. 52-5754
 Power Approach Configuration

TRIM CONDITIONS

IAS 160 knots Alt. 4300 feet Weight 24,700 lbs
 CG 30 %MAC N₂ 9,360 RPM Stab. Pos. 8.5 N. U.



Time - Seconds

APPENDIX I

FIG No 62

SIDESLIP CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION (A/B ON)
NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

CAS 281 KNOTS RUDDER POS 0°
 ALT 45970 FEET L. OUTER AIL POS .5° UP
 N₂ 9070 RPM STAB POS 5.5° NOSE UP
 GROSS WEIGHT 21700 LBS

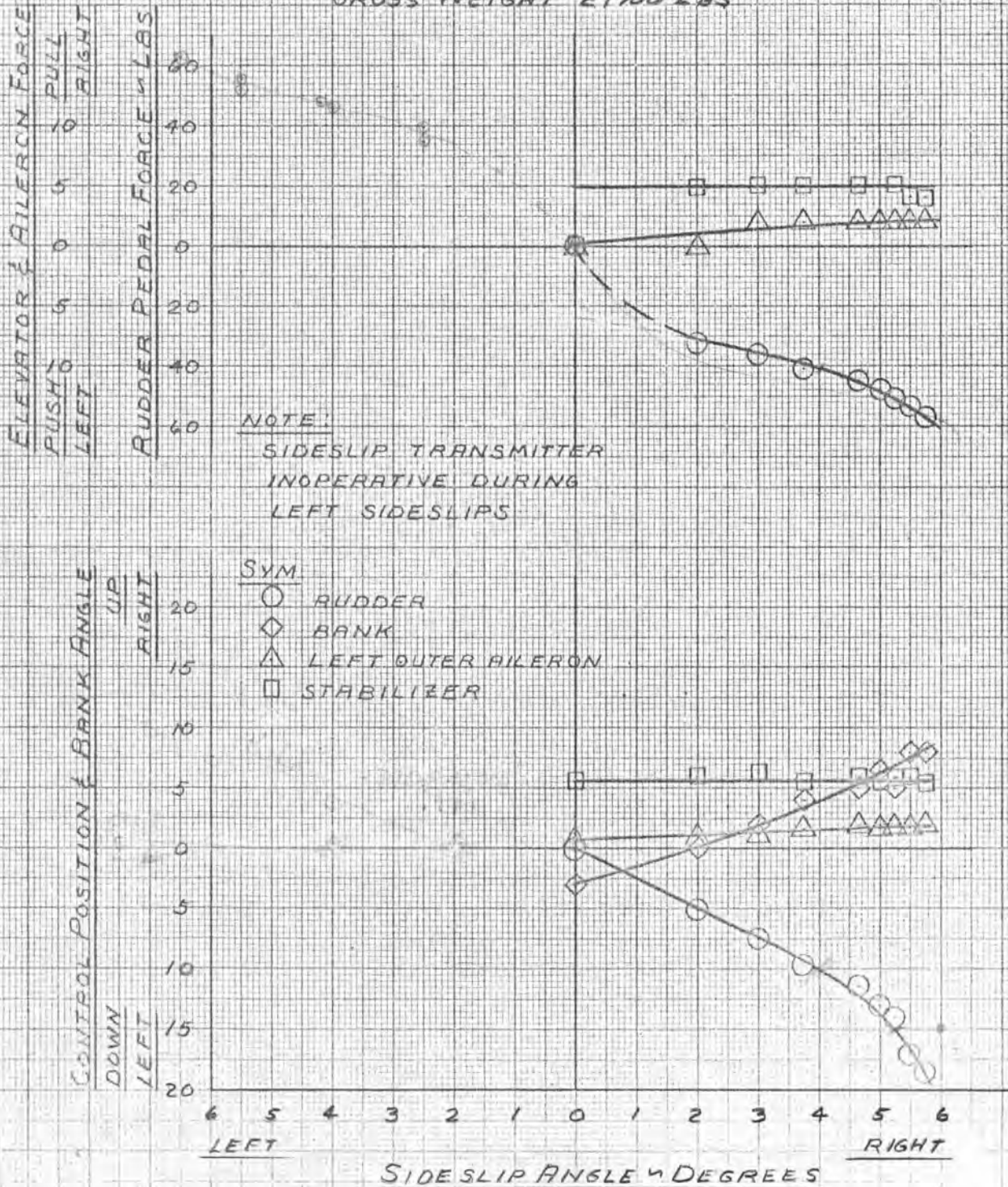


FIG No 63

SIDESLIP CHARACTERISTICS
 YF-100A USAF No 52-5754
 POWER APPROACH CONFIGURATION
 NO EXTERNAL STORES ATTACHED

TRIM CONDITIONS

IAS 160 KNOTS RUDDER POS 1.5° RIGHT
 ALT 11620 FEET L OUTER AIL POS 0°
 N₂ 9440 RPM STAB POS 8.5° NOSE UP
 GROSS WEIGHT 23500 LBS

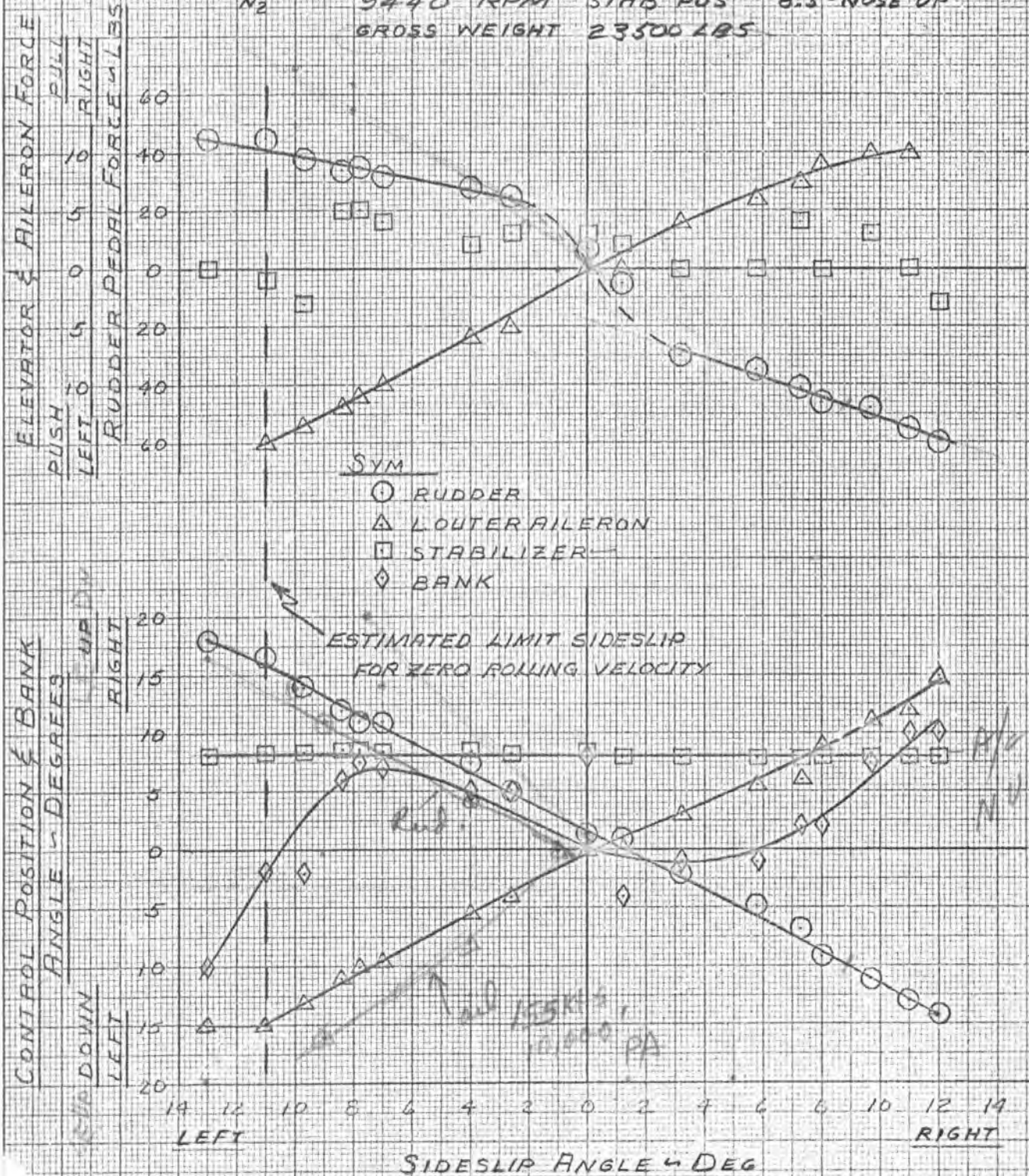


Figure No. 64

DYNAMIC DIRECTIONAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.8 % MAC Weight 22,200 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 degrees
 L. Aileron Position 0.4 degrees Up

No external stores attached

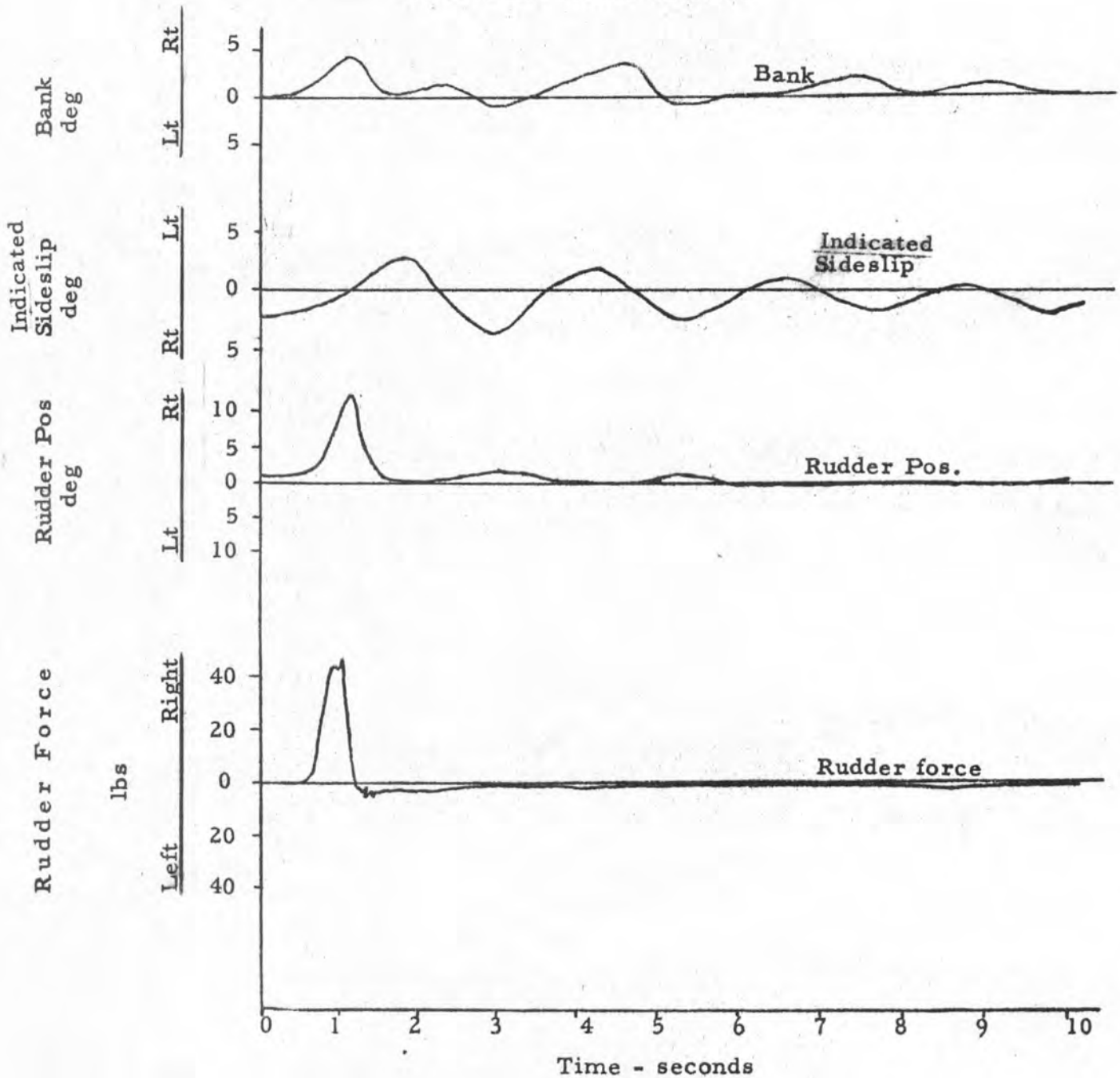


Figure No. 65

DYNAMIC DIRECTIONAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.8 % MAC Weight 22,200 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 degrees
 L. Aileron Position 0.4 degrees Up

No external stores attached

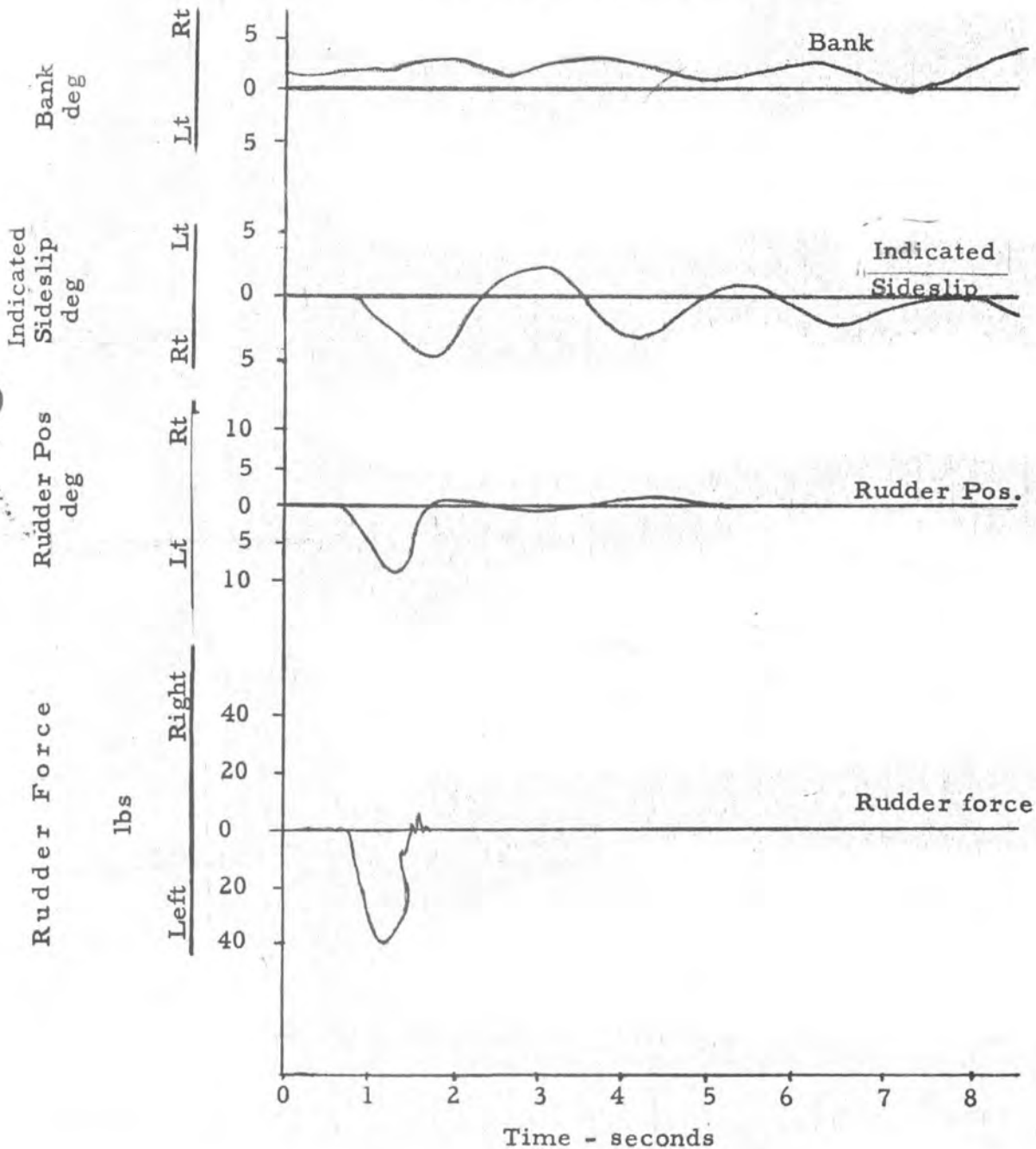


Figure No. 66

DYNAMIC DIRECTIONAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.8 % MAC Weight 22,200 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 degrees
 L. Aileron Position 0.4 degrees Up
 No external stores attached

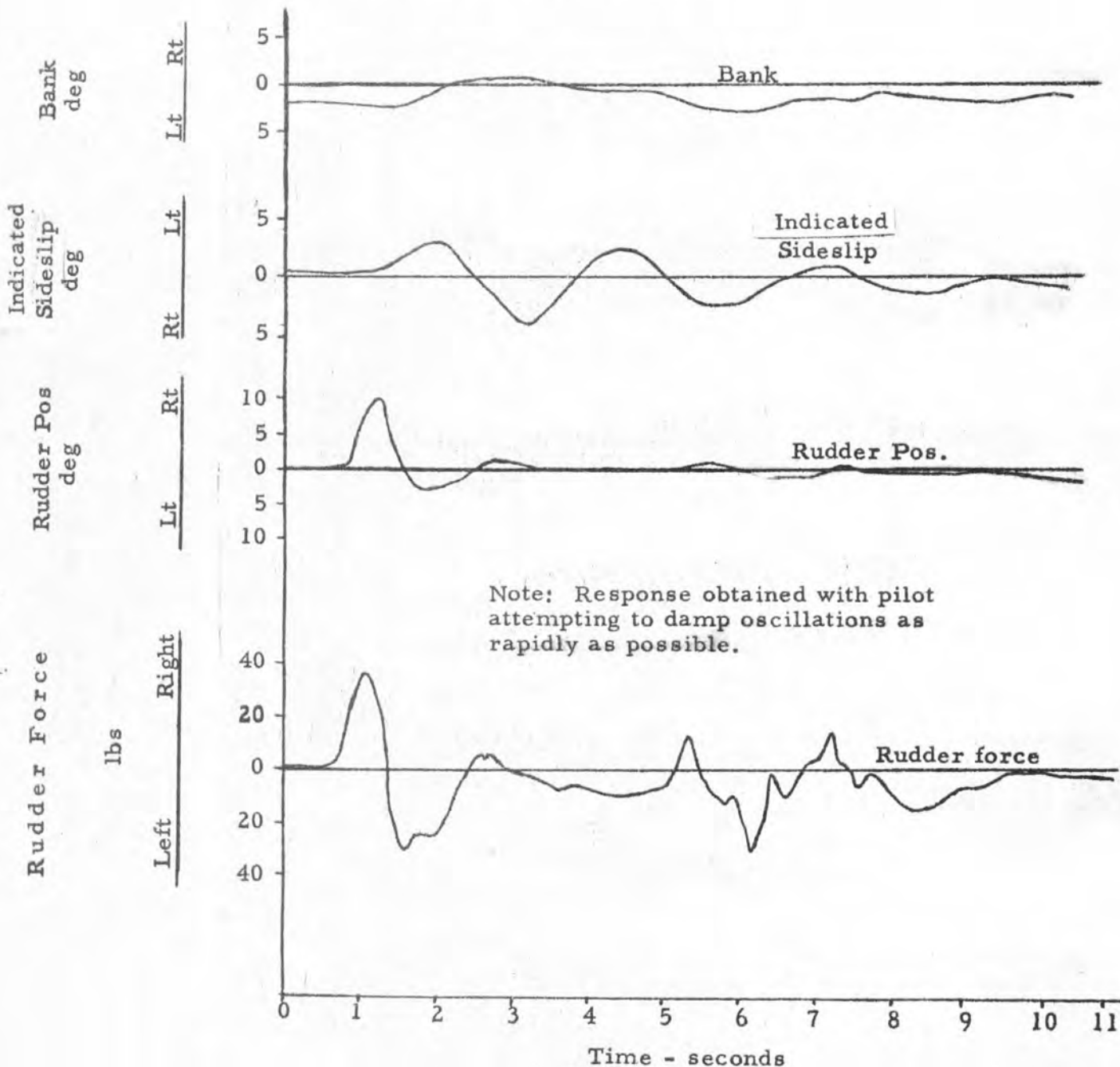


Figure No. 67

DYNAMIC DIRECTIONAL STABILITY
 YF-100A, USAF No. 52-5754.
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.8 o/o MAC Weight 22,200 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 degrees
 L. Aileron Position 0.4 degrees Up
 No external stores attached

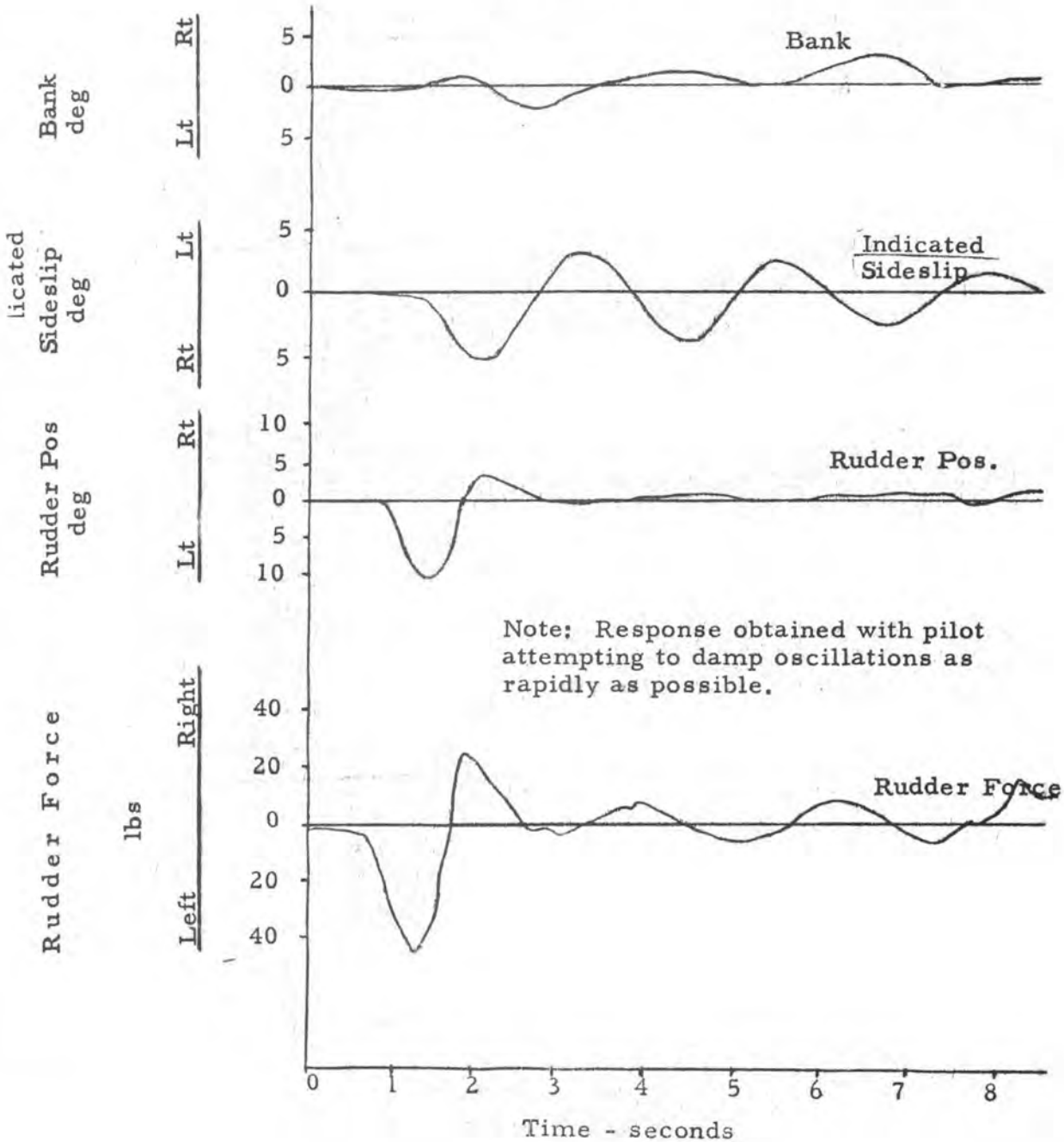
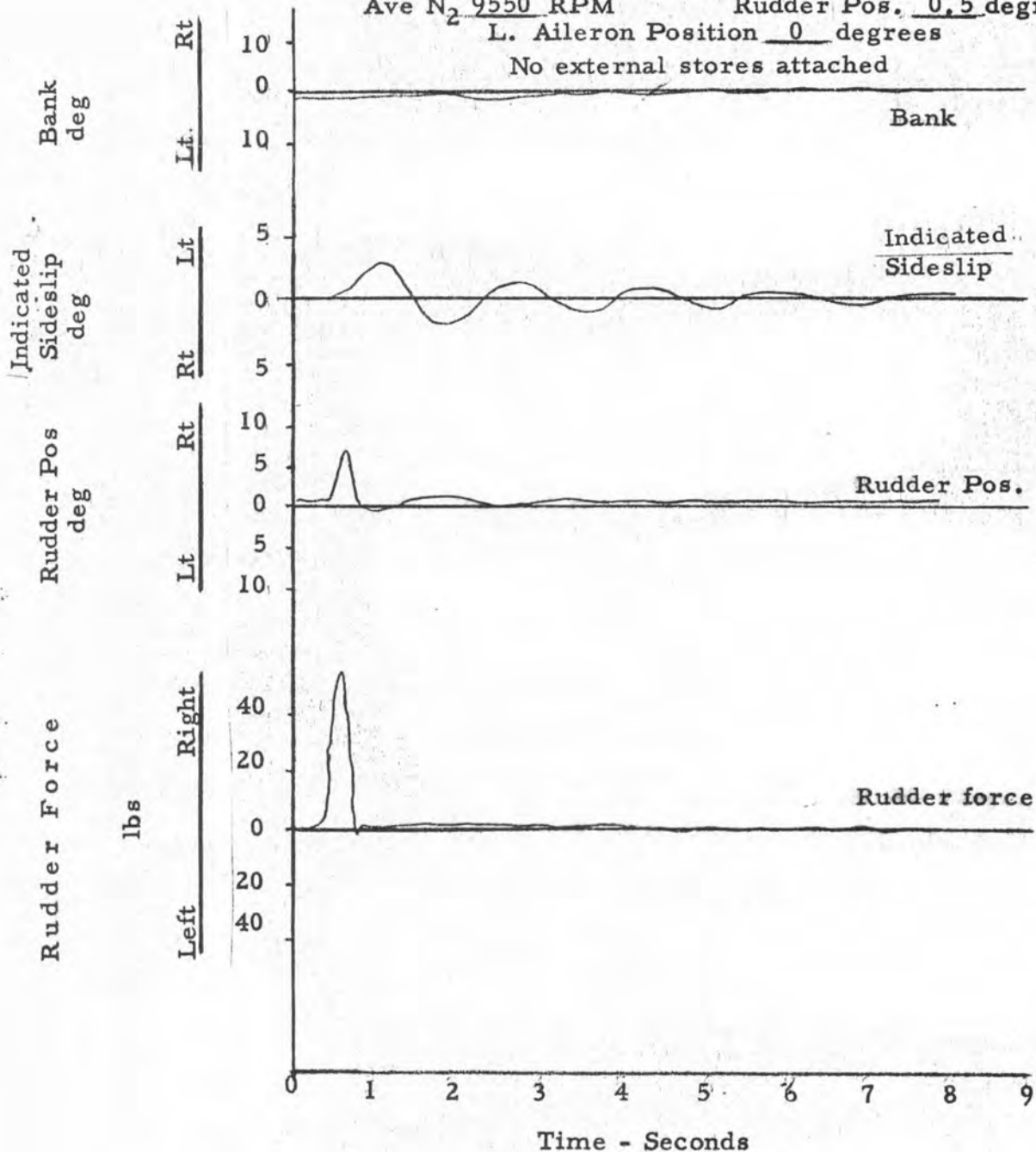


Figure No. 68

DYNAMIC DIRECTIONAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 31.1 % MAC Weight 22,900 lbs
 Ave N₂ 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Position 0 degrees
 No external stores attached



Time - Seconds

APPENDIX I

Figure No. 69

DYNAMIC DIRECTIONAL STABILITY
 YF-100 A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots Altitude 10,000 feet
 CG 31.1 % MAC Weight 22,900 lbs
 Ave N_2 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Pos. 0 degrees

No external stores attached

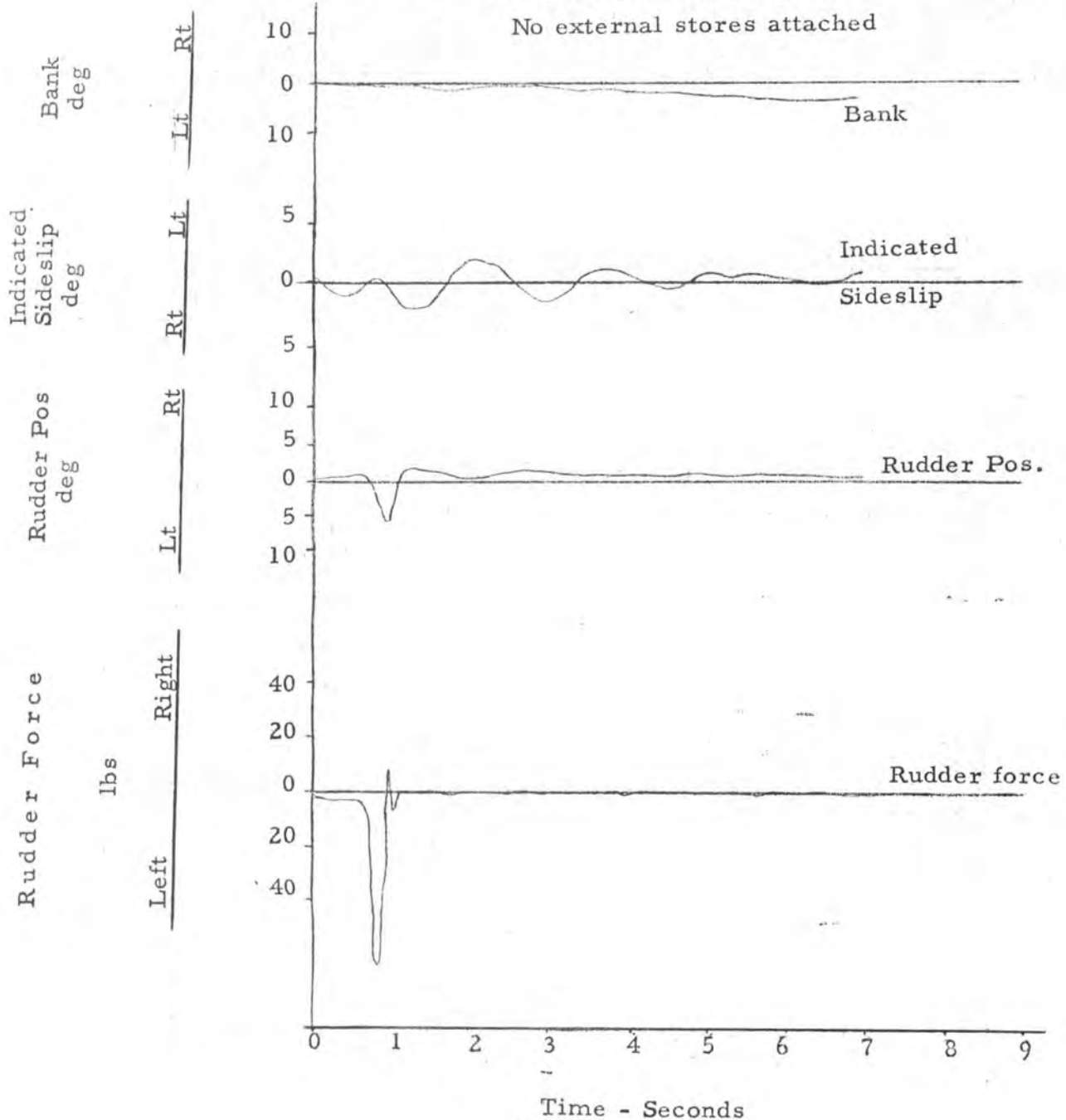


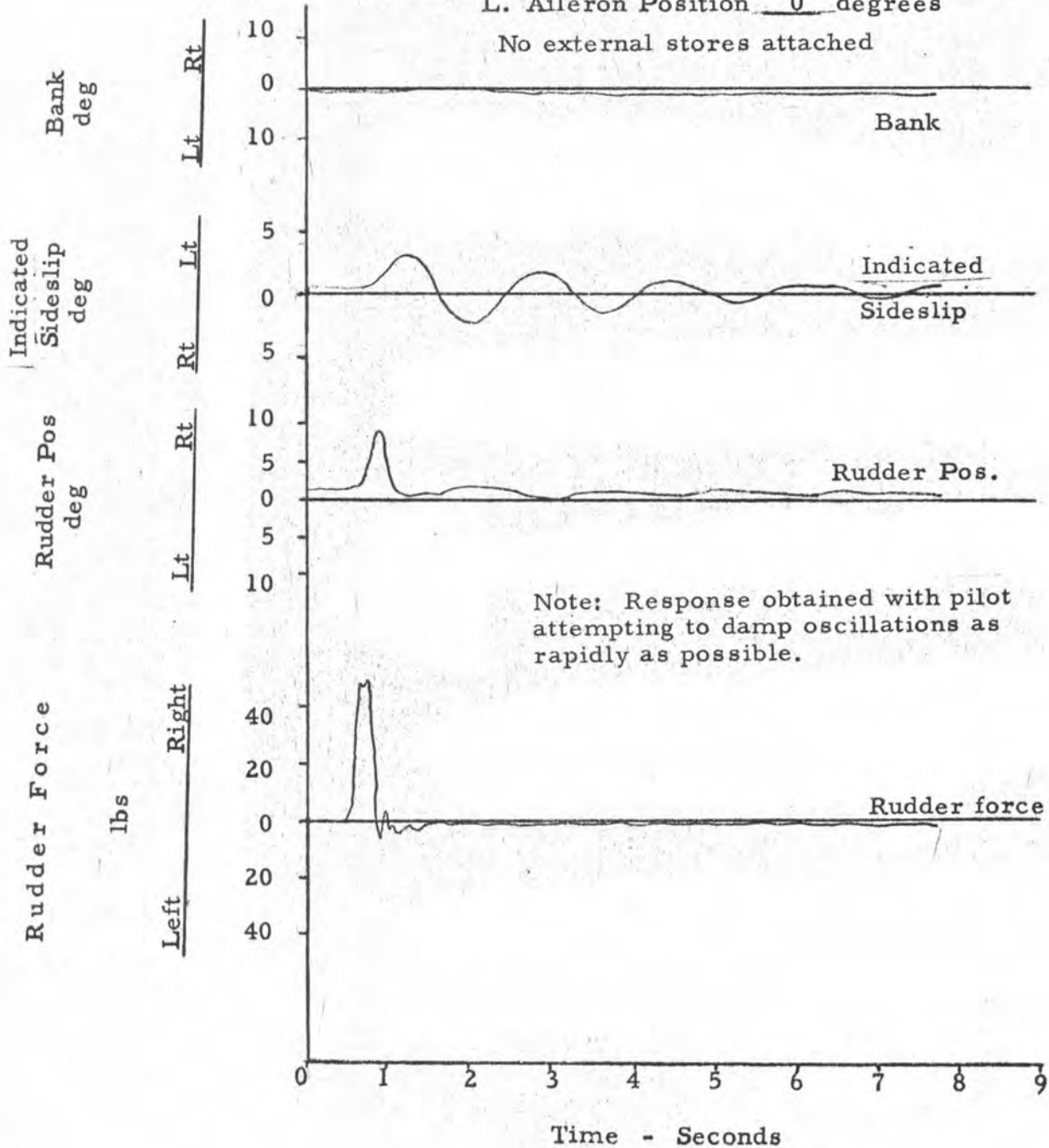
Figure No. 70

DYNAMIC DIRECTIONAL STABILITY
 YF-100 A, USAF No. 52-5754
 Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 31.1 % MAC Weight 22,900 lbs
 Ave N₂ 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Position 0 degrees

No external stores attached



Note: Response obtained with pilot attempting to damp oscillations as rapidly as possible.

Figure No. 71

DYNAMIC DIRECTIONAL STABILITY

YF-100 A₂ USAF No. 52-5754

Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 31.1 /o MAC Weight 22,900 lbs
 Ave N₂ 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Position 0 degrees

No external stores attached

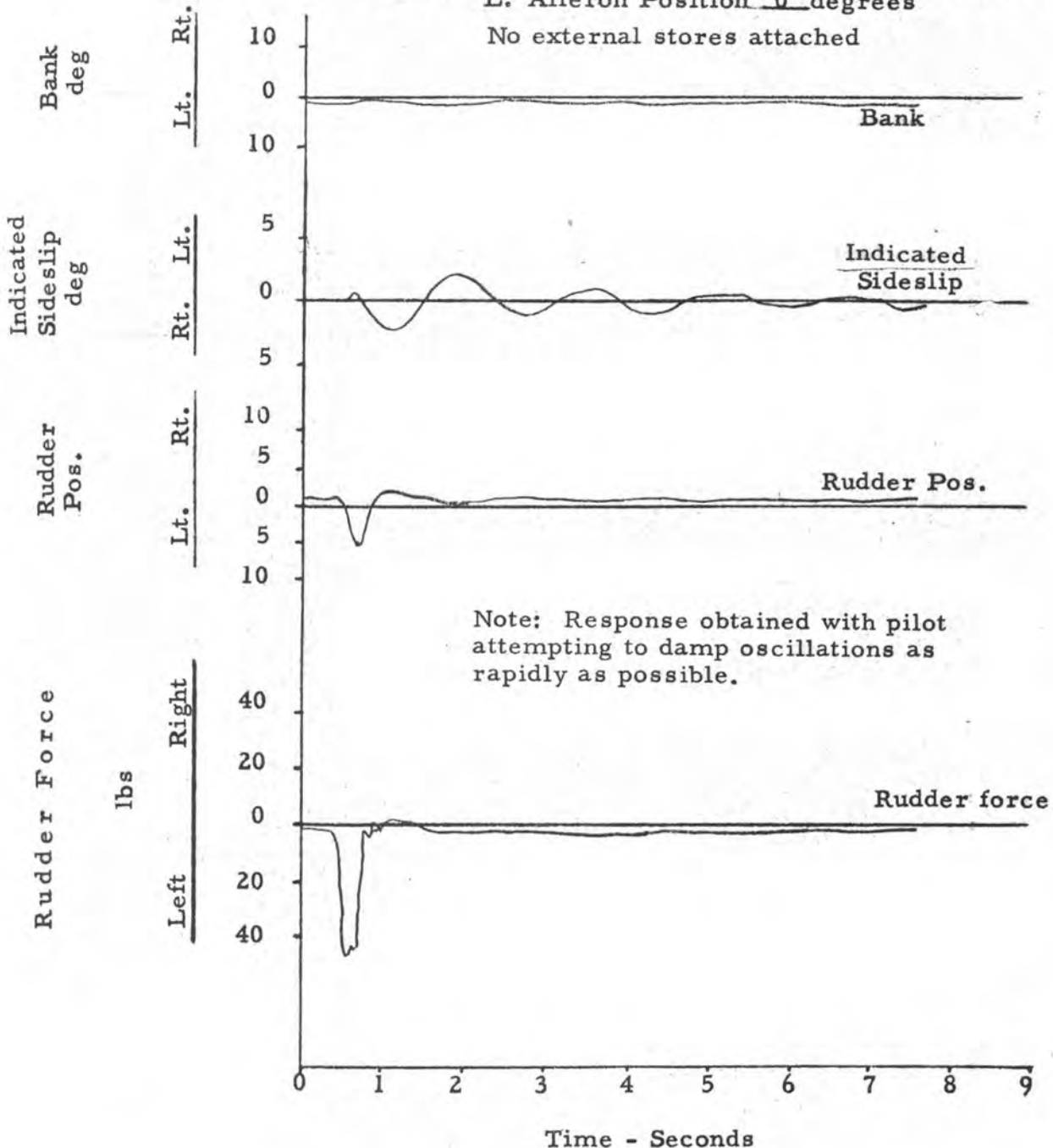


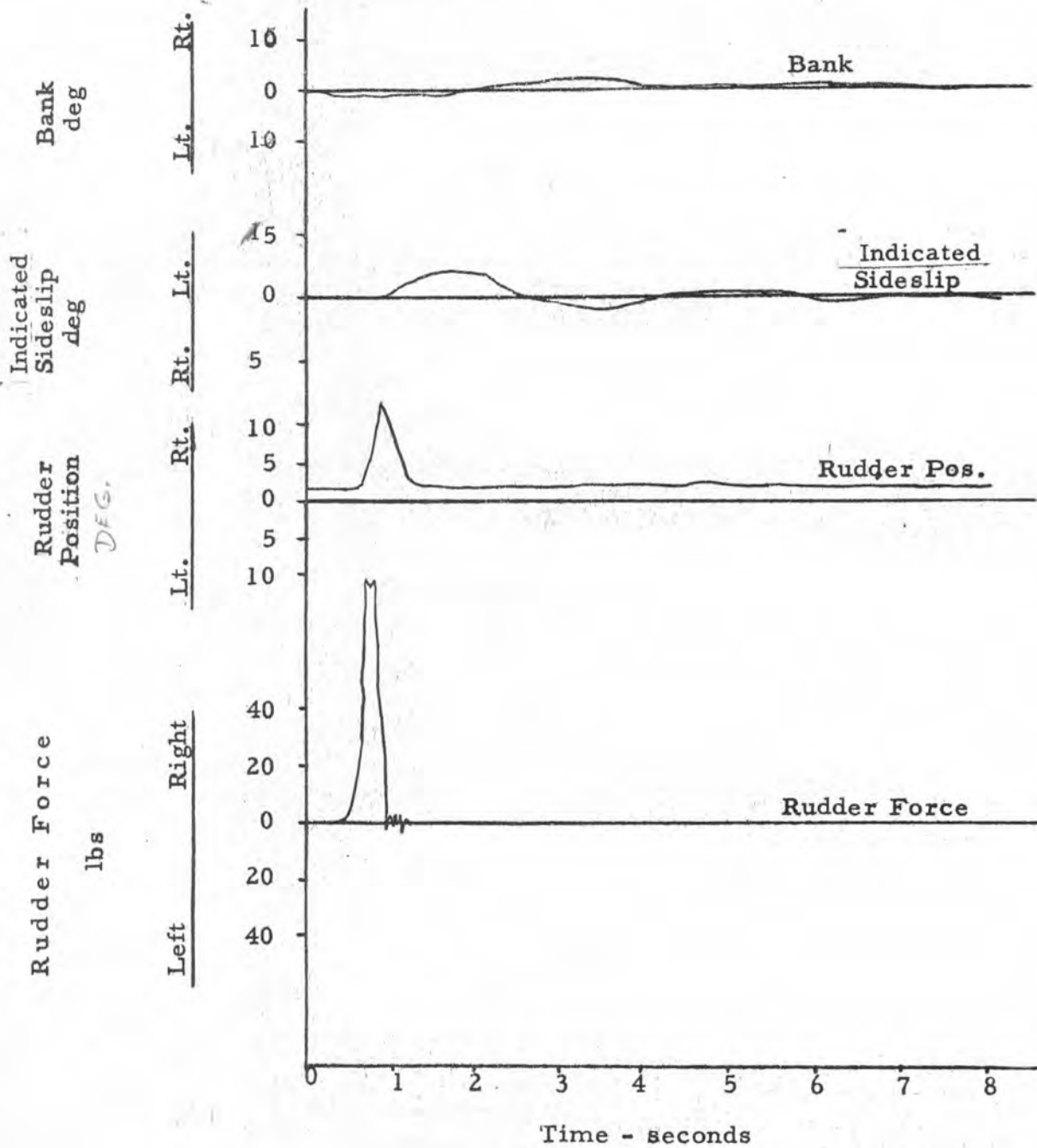
Figure No. 72

DYNAMIC DIRECTIONAL STABILITY
 YF-100A. USAF No. 52-5754
 Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
 CG 30.4 % MAC Weight 21,300 lbs
 Ave N_2 9240 RPM Rudder Pos. 2.3 degrees Rt.
 L. Aileron Position 0.5 degrees Up

No external Stores attached



Time - seconds

APPENDIX I

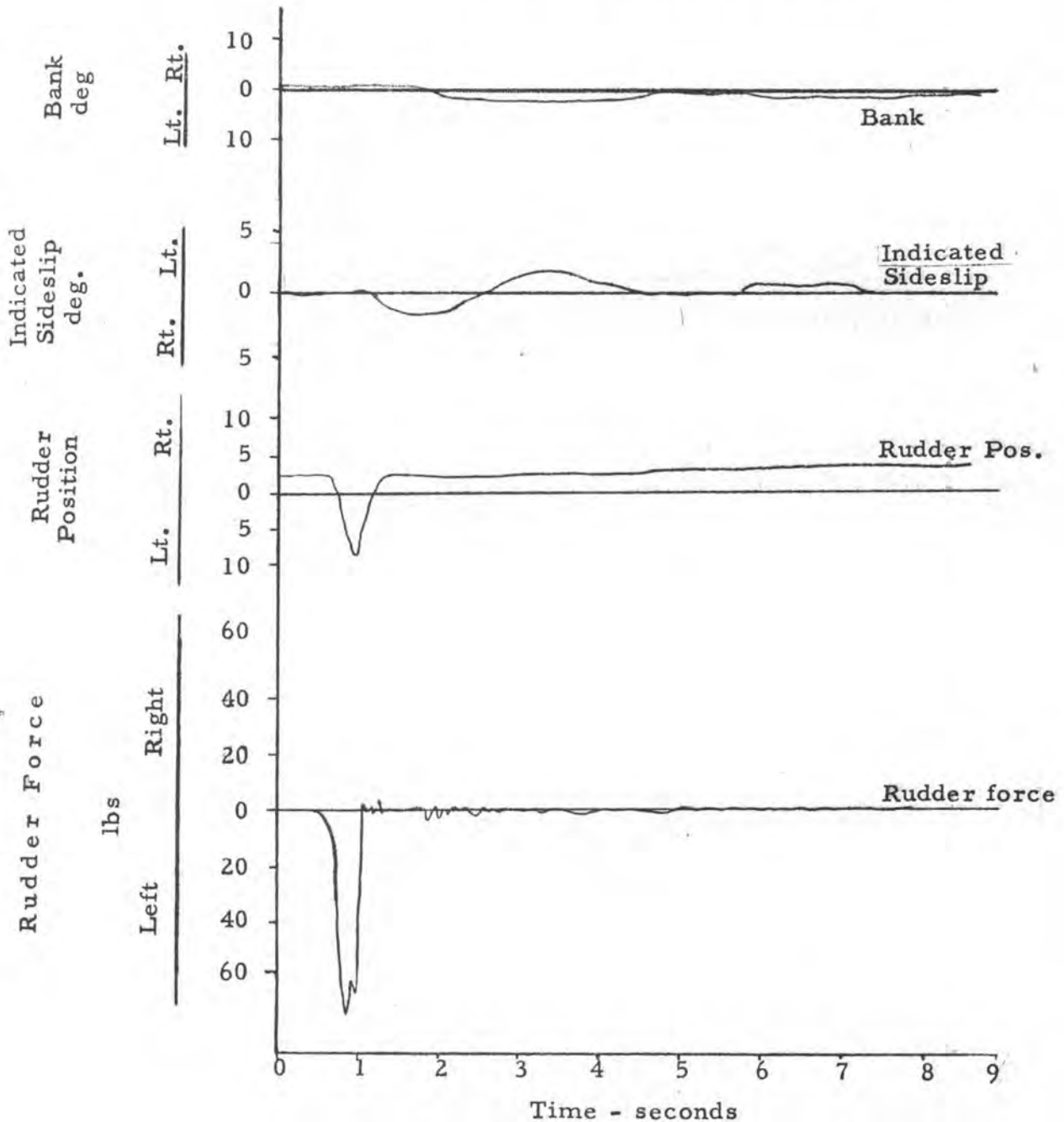
Figure No.73

DYNAMIC DIRECTIONAL STABILITY
YF-100 A, USAF No. 52-5754
Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
 CG 30.4 % MAC Weight 21,300 lbs
 Ave N₂ 9240 RPM Rudder Pos. 2.3 degrees Rt
 L. Aileron Position 0.5 degrees Up

No external stores attached



APPENDIX I

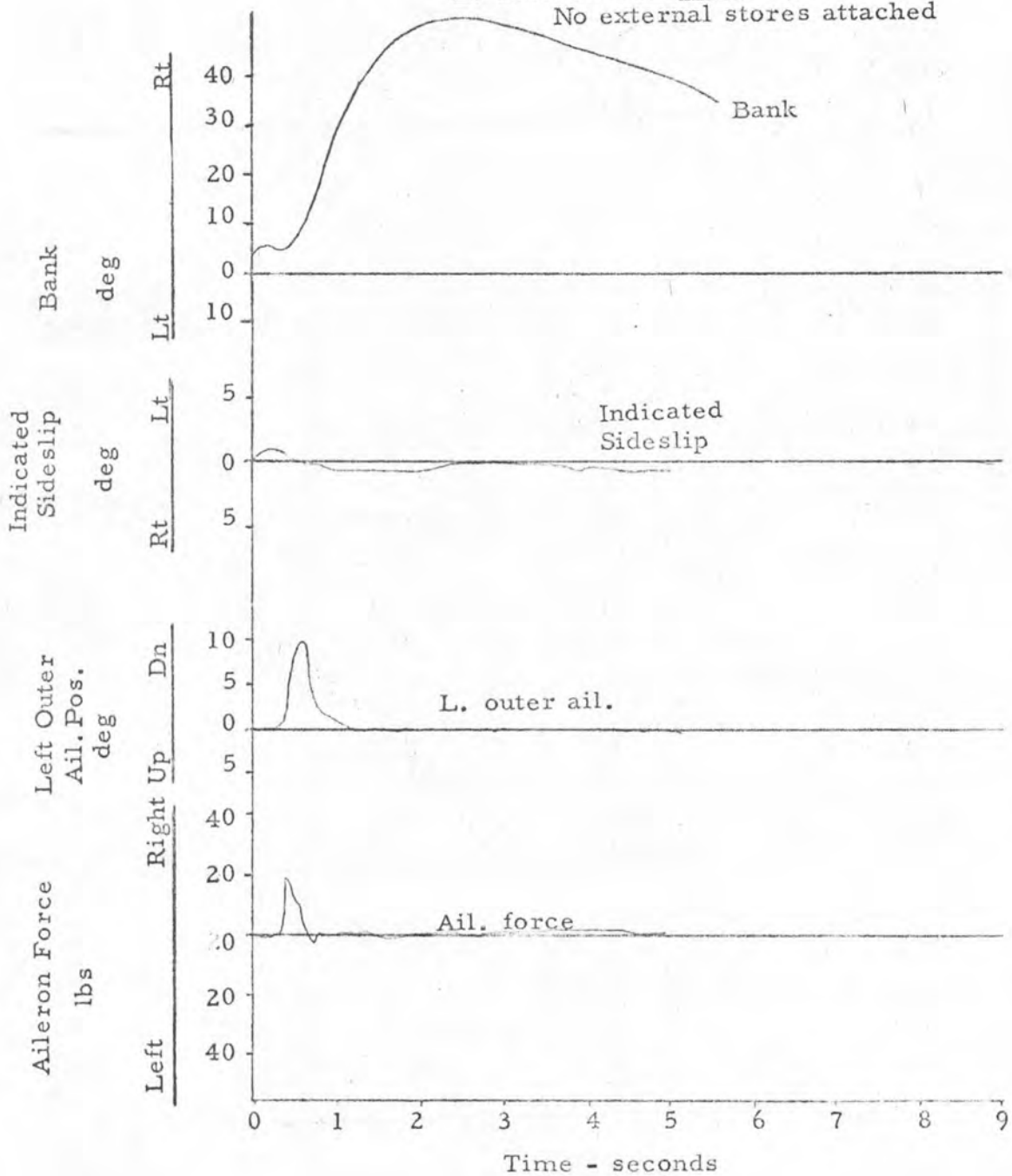
Figure No. 74

DYNAMIC LATERAL STABILITY
YF-100 A, USAF No. 52-5754

Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
CG 29.5 %MAC Weight 21,800 lbs
Ave N₂ 9070 RPM Rudder Pos. 0 deg.
L. Aileron Pos. 0.4 Degrees Up
No external stores attached



Time - seconds

APPENDIX I

Figure No. ° 75

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B On) Controls Free

TRIM CONDITIONS

CAS	<u>274</u>	knots	Altitude	<u>45,300</u>	feet
CG	<u>29.5</u>	o/o MAC	Weight	<u>21,800</u>	lbs
Ave N ₂	<u>9070</u>	RPM	Rudder Pos.	<u>0</u>	deg
			L. Aileron Position	<u>0.4</u>	degrees Up

No external stores attached

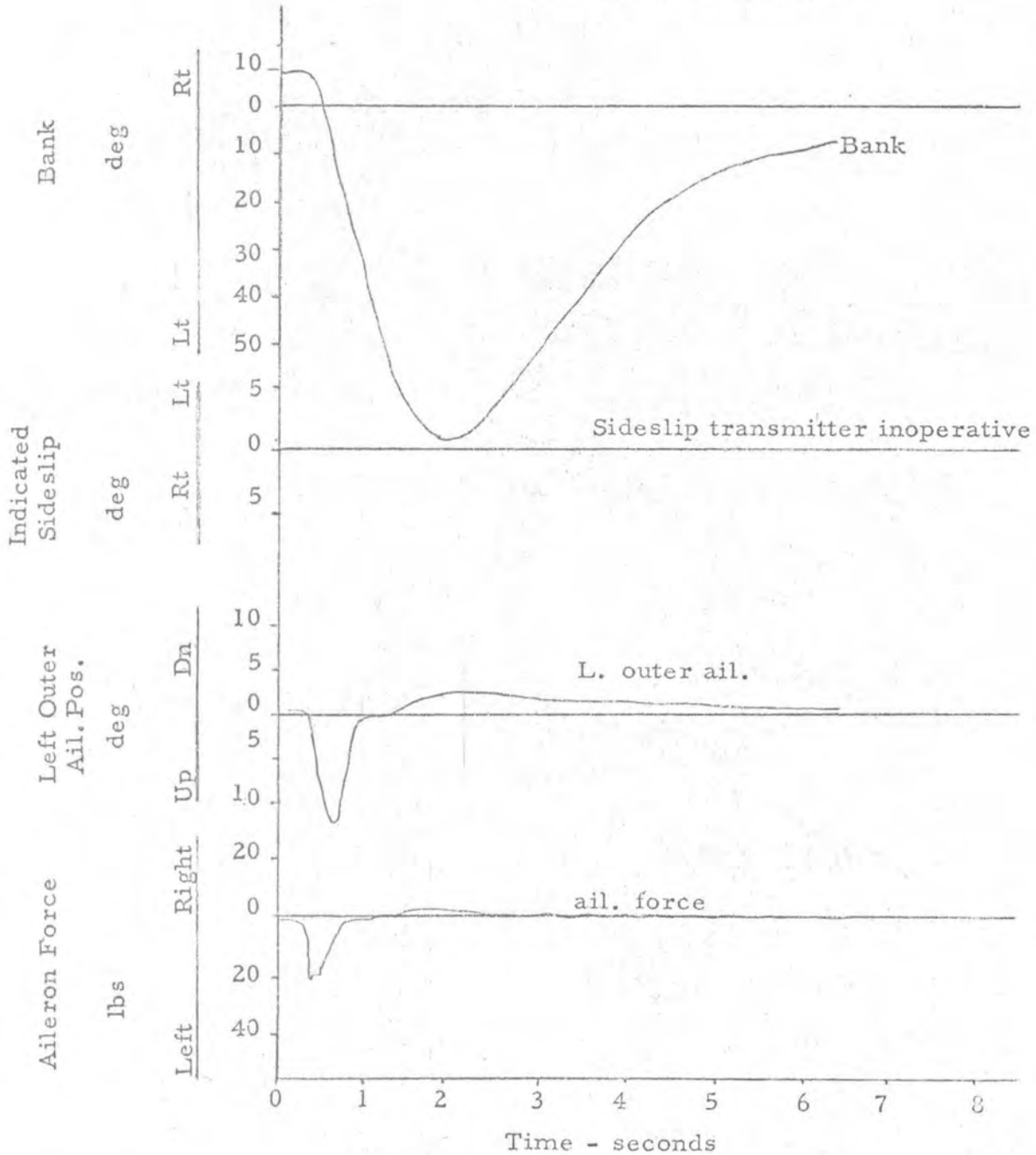


Figure No. 76

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.5 o/o MAC Weight 21,800 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 deg
 L Aileron Position 0.4 degrees Up
 No external stores attached

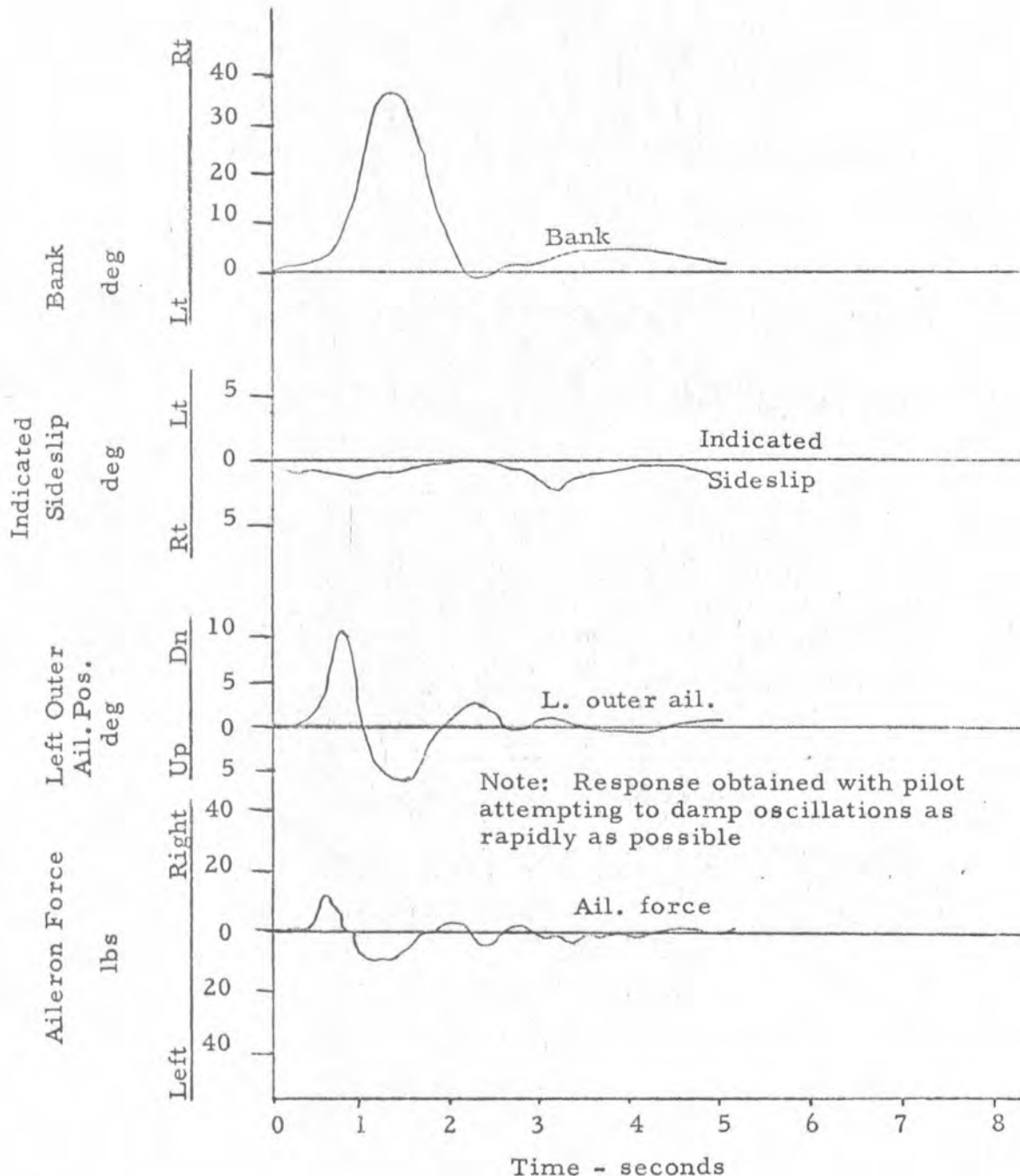


Figure No. 77

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B on)

TRIM CONDITIONS

CAS 274 knots Altitude 45,300 feet
 CG 29.5 o/o MAC Weight 21,800 lbs
 Ave N₂ 9070 RPM Rudder Pos. 0 deg
 L. Aileron Position 0.4 degrees Up

No external stores attached

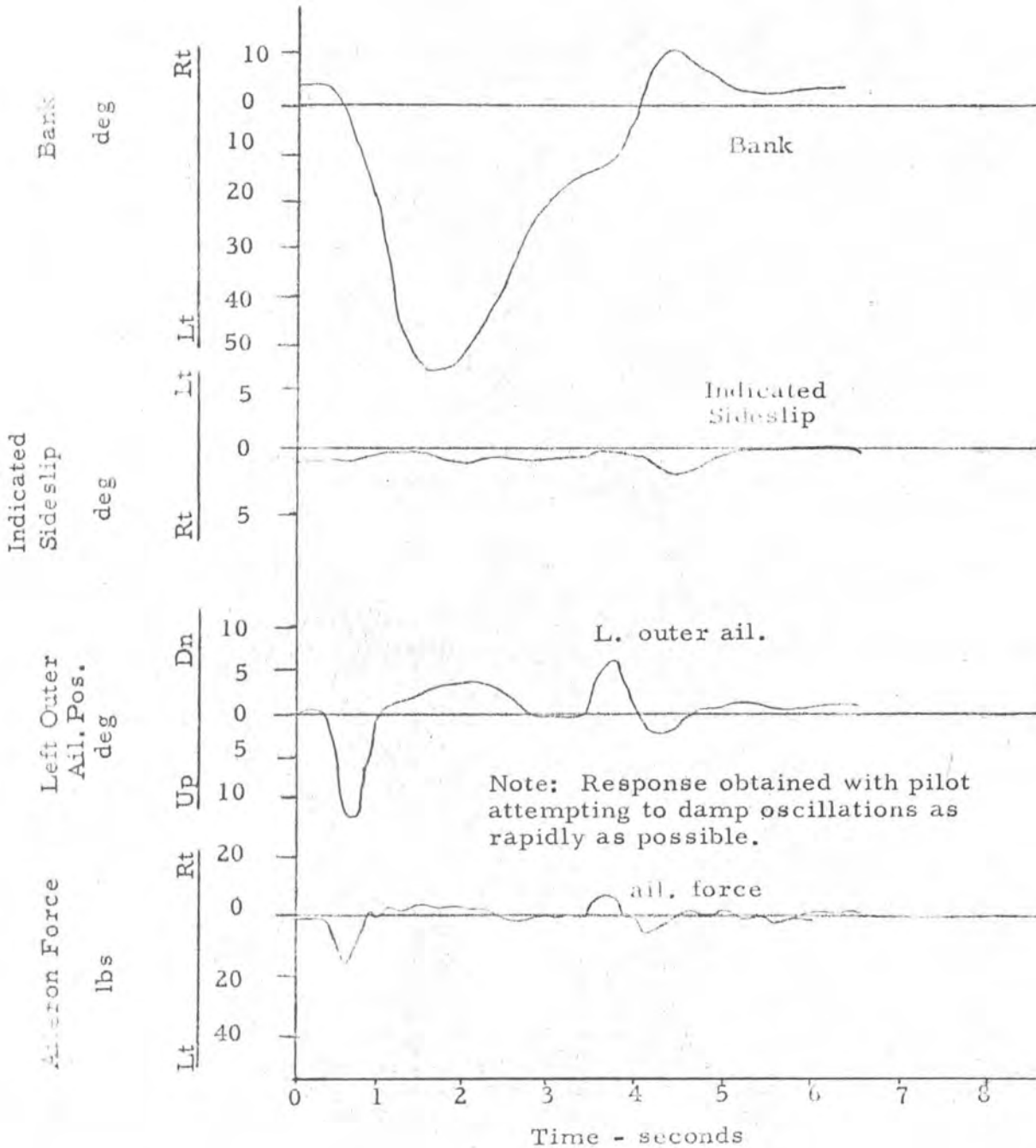
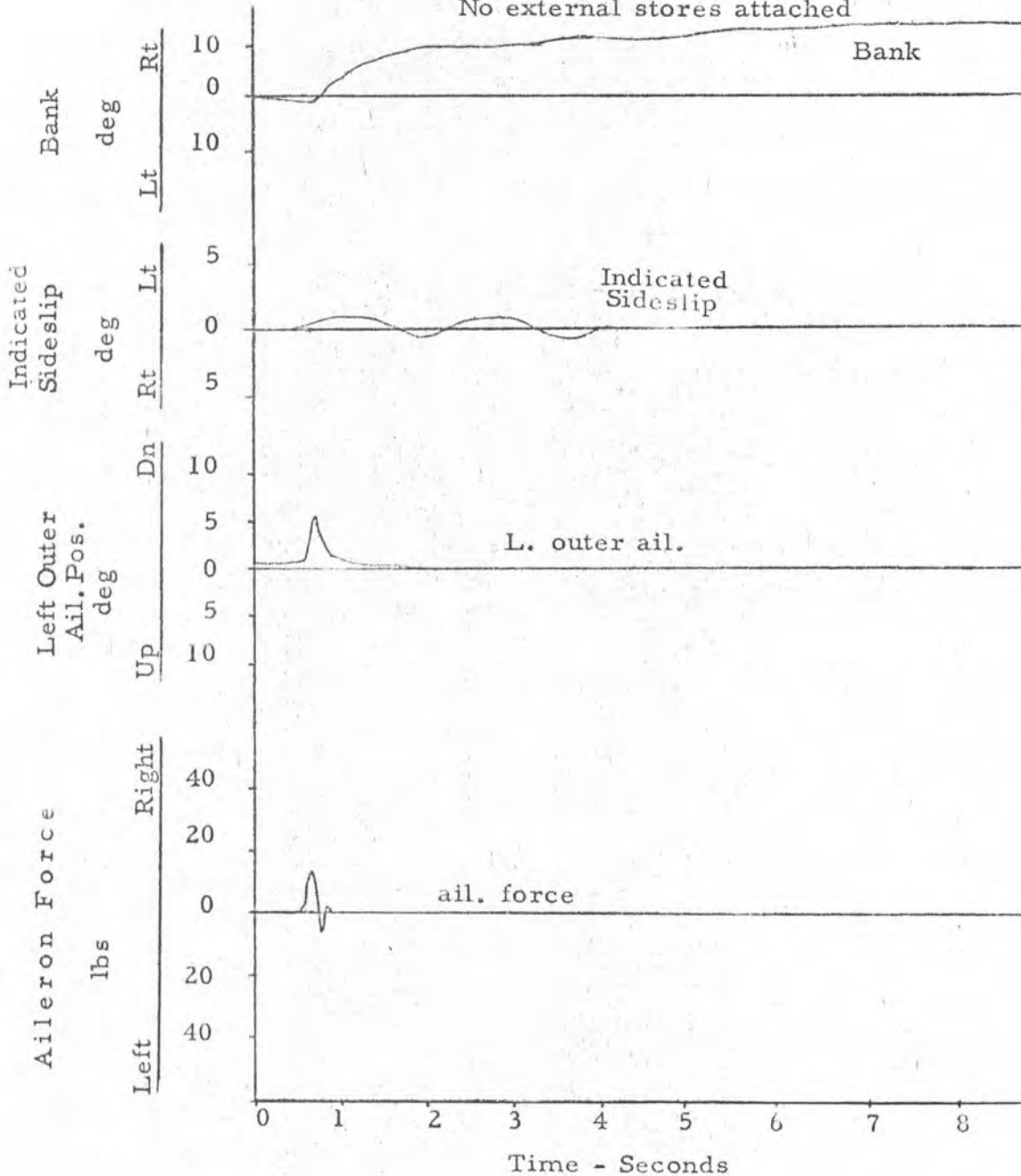


Figure No. 78

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 30.5 % MAC Weight 22,500 lbs
 Ave N₂ 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Pos. 0 degrees
 No external stores attached



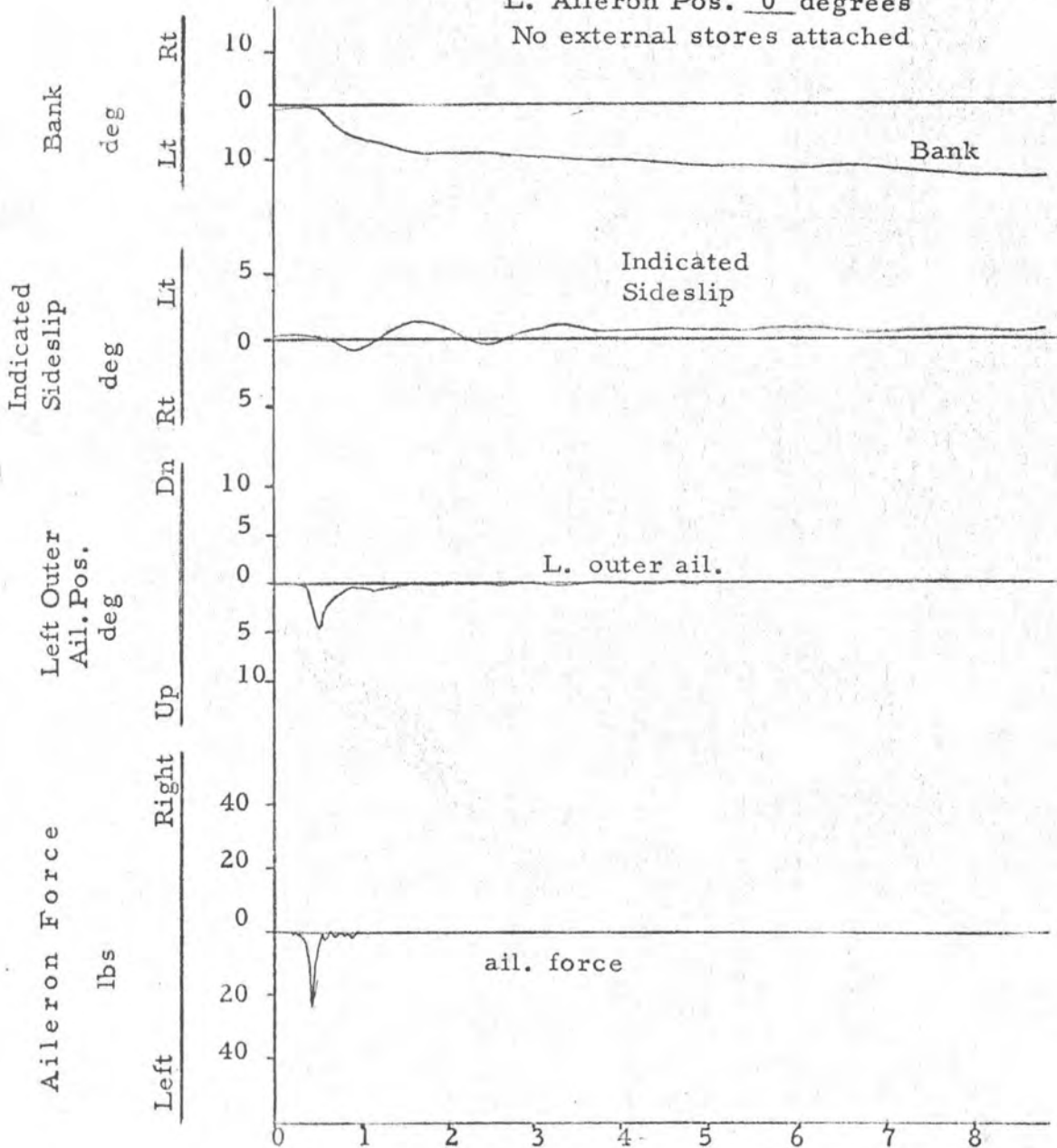
APPENDIX I

Figure No. 79

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Configuration (A/B Off) Controls Free

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
 CG 30.5 % MAC Weight 22,500 lbs
 Ave N₂ 9550 RPM Rudder Pos. 0.5 degrees Rt
 L. Aileron Pos. 0 degrees
 No external stores attached



Time - Seconds

APPENDIX I

Figure No. 80

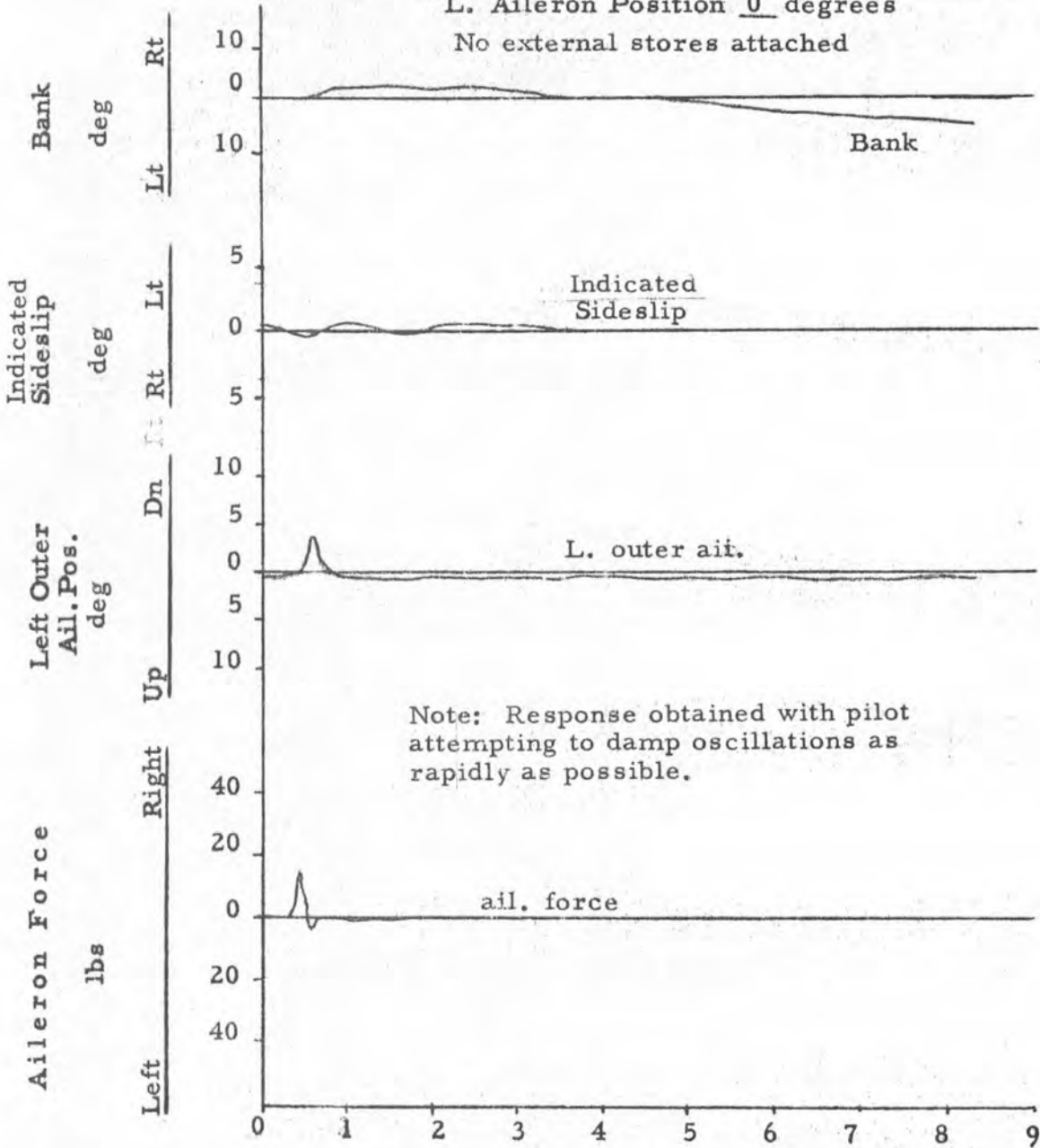
DYNAMIC LATERAL STABILITY
YF-100A, USAF No. 52-5754

Power Configuration (A/B off)

TRIM CONDITIONS

CAS 475 knots Altitude 10,500 feet
CG 30.5 % MAC Weight 22,500 lbs
Ave N₂ 9550 RPM Rudder Pos. 0.5 Degrees Rt
L. Aileron Position 0 degrees

No external stores attached



Time - Seconds

APPENDIX I

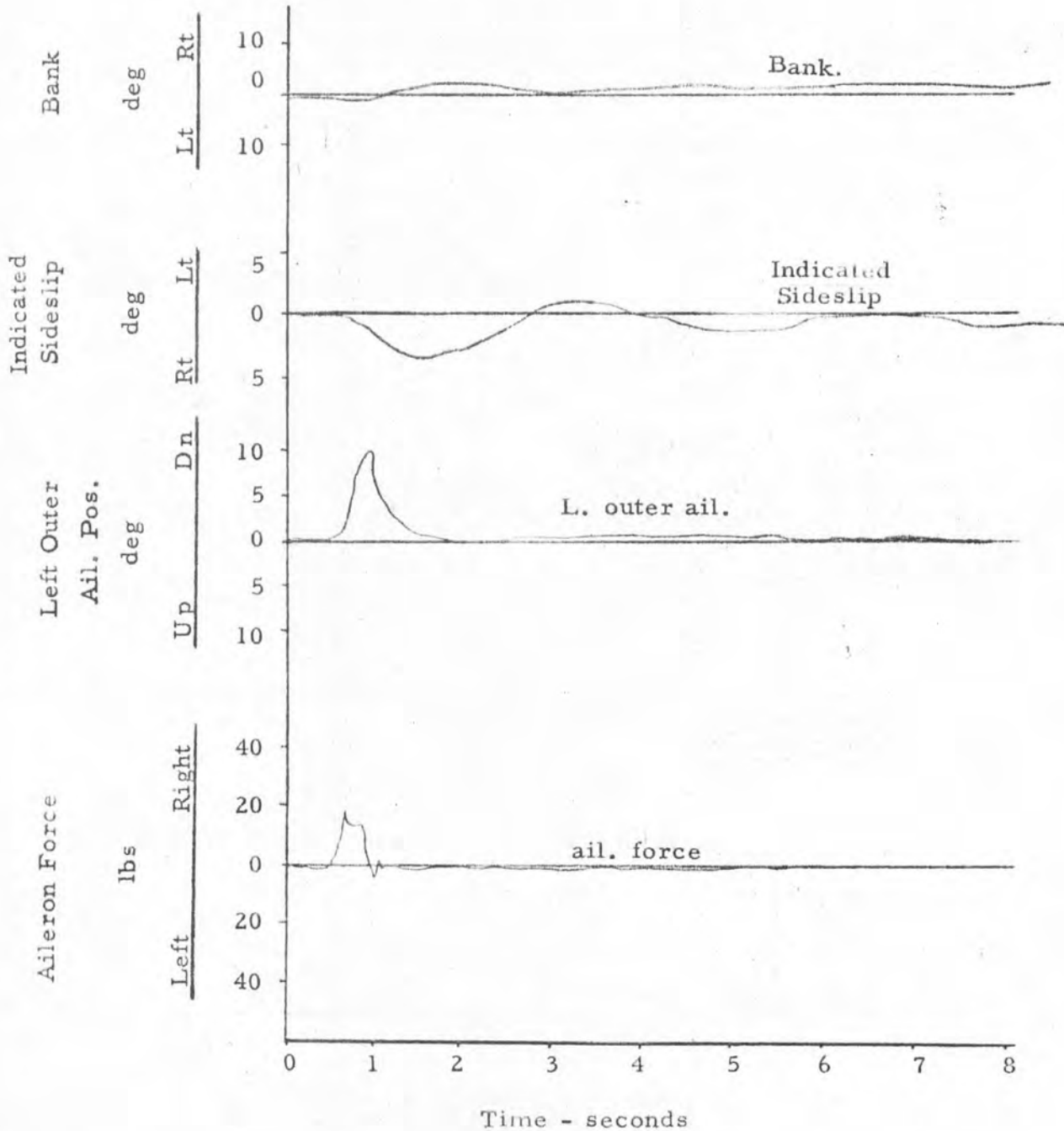
Figure No. 82

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
 CG 30.5 % MAC Weight 21,200 lbs
 Ave N_2 9240 RPM Rudder Pos. 2.3 degrees Rt
 L. Aileron Position 0.5 degrees Up

No external stores attached



Time - seconds

Figure No. 83

DYNAMIC LATERAL STABILITY
 YF-100A, USAF No. 52-5754
 Power Approach Configuration Controls Free

TRIM CONDITIONS

IAS 164 knots Altitude 9,900 feet
 CG 30.5 % MAC Weight 21,200 lbs
 Ave N₂ 9240 RPM Rudder Pos. 2.3 degrees Rt
 L. Aileron Position 0.5 degrees Up
 No external stores attached

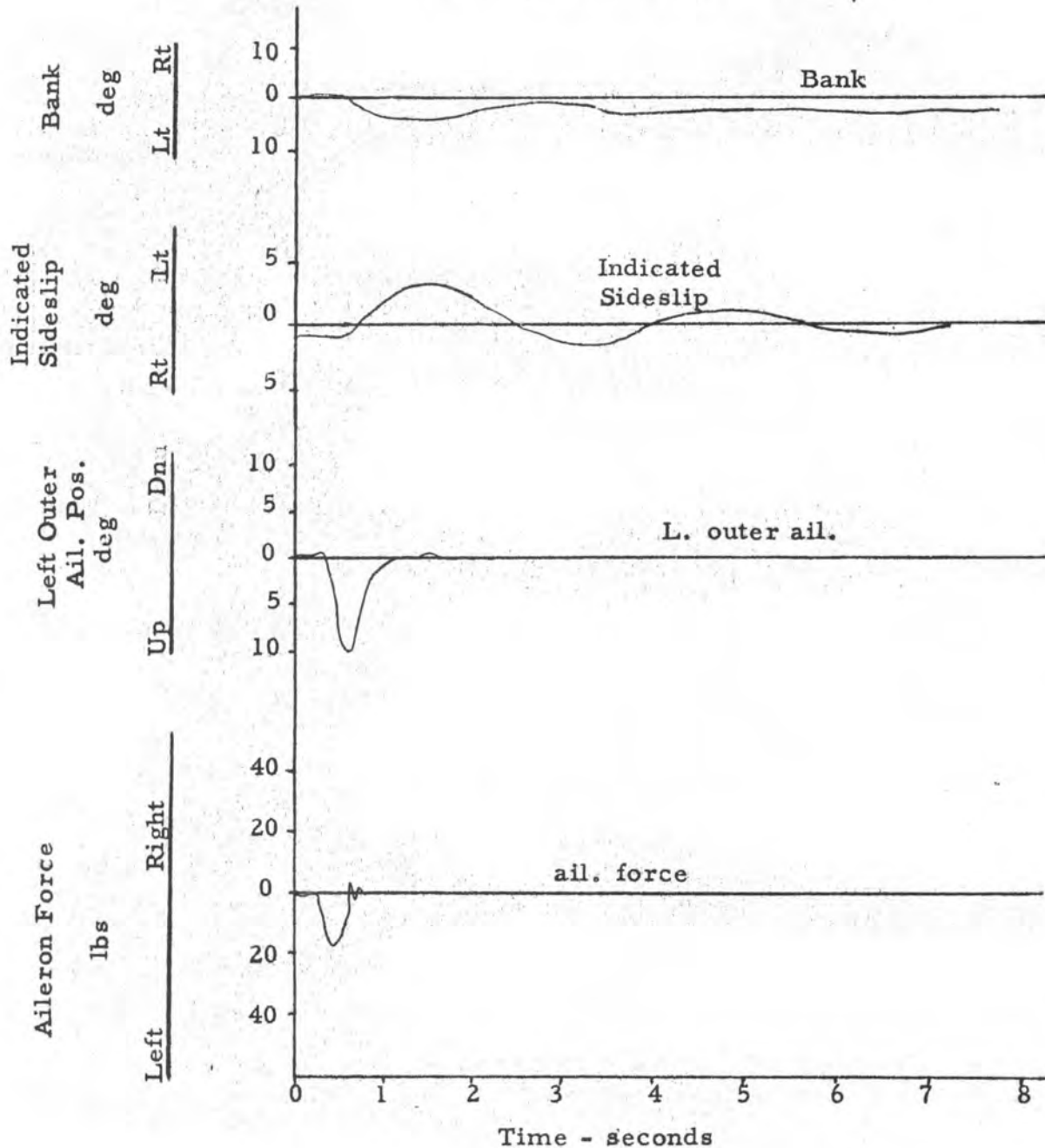


Figure No. 84

ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Approach Config.

TRIM CONDITIONS

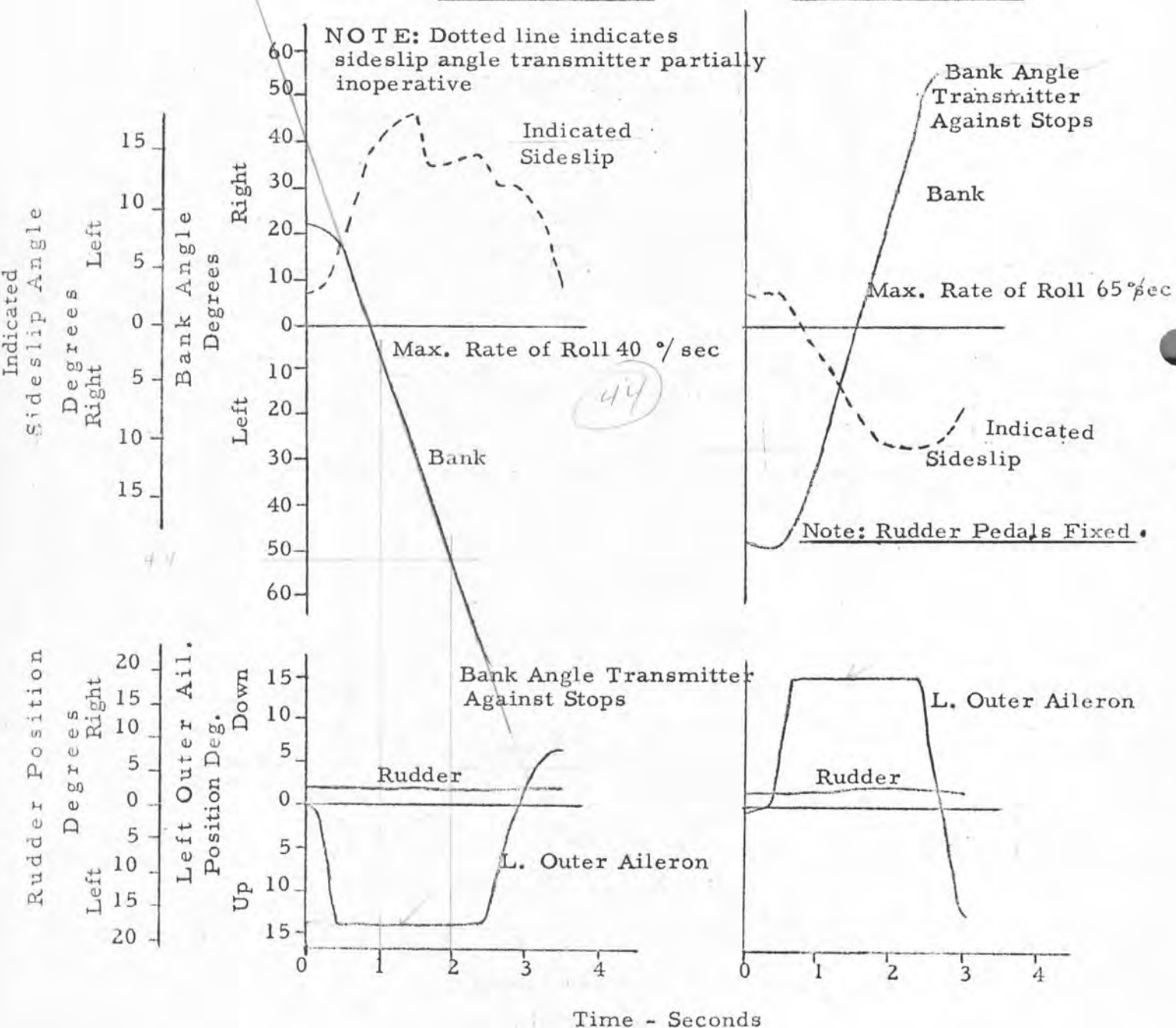
IAS 167 knots Altitude 10,400 feet
CG 31.4 %MAC Weight 23,000 lbs
Rudder Pos. 2° left L. Outer Ail. Pos. 0

No external stores attached

LEFT ROLL

RIGHT ROLL

NOTE: Dotted line indicates sideslip angle transmitter partially inoperative



Time - Seconds

APPENDIX I



Figure No. 85

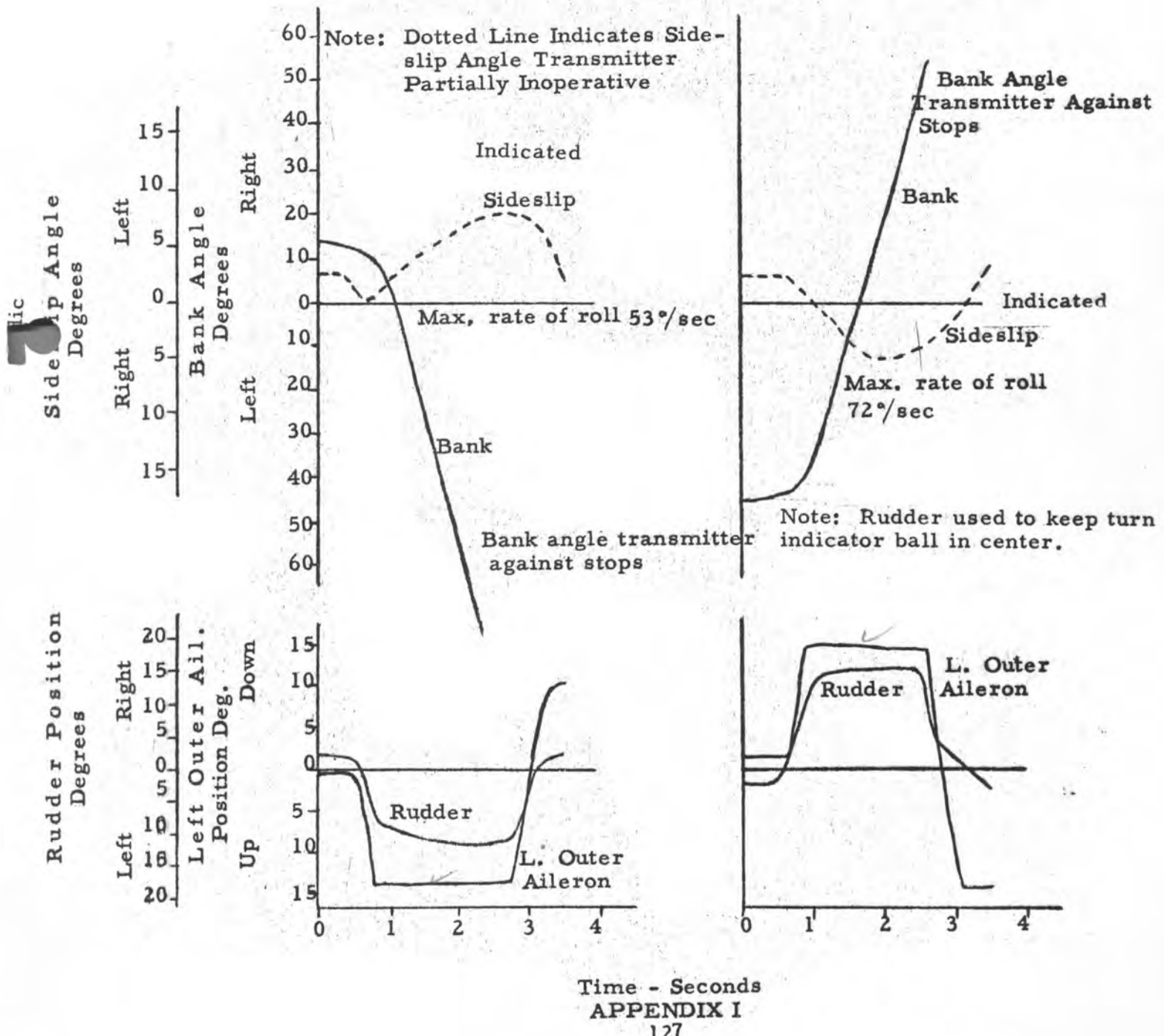
ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Approach Configuration

TRIM CONDITIONS

IAS 167 knots Altitude 10,400 feet
CG 31.4 %MAC Weight 23,000 lbs
Rudder Pos. 2° left L. Outer Ail. Pos. 0
No external stores attached

LEFT ROLL

RIGHT ROLL



ADVERSE YAW
YF-100A, USAF No. 52-5754

Power Configuration (A/B Off)
TRIM CONDITIONS

CAS 245 knots Altitude 45000 feet
CG 30.6 % MAC Weight 22,500 lbs
Rudder Pos. 2° left L. Outer Ail. Pos. 5° Dn
No external stores attached

LEFT ROLL

RIGHT ROLL

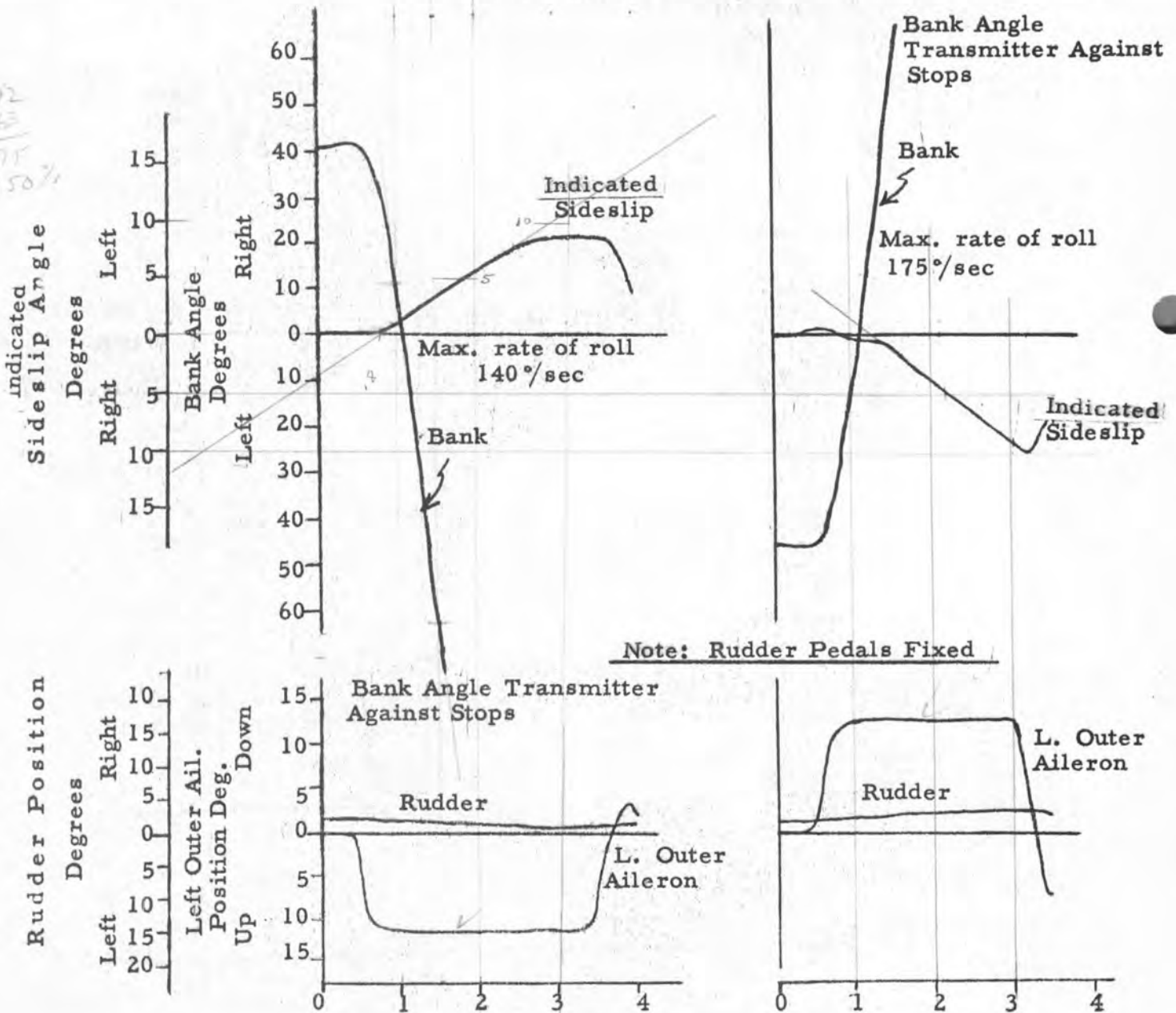


Figure No. 87

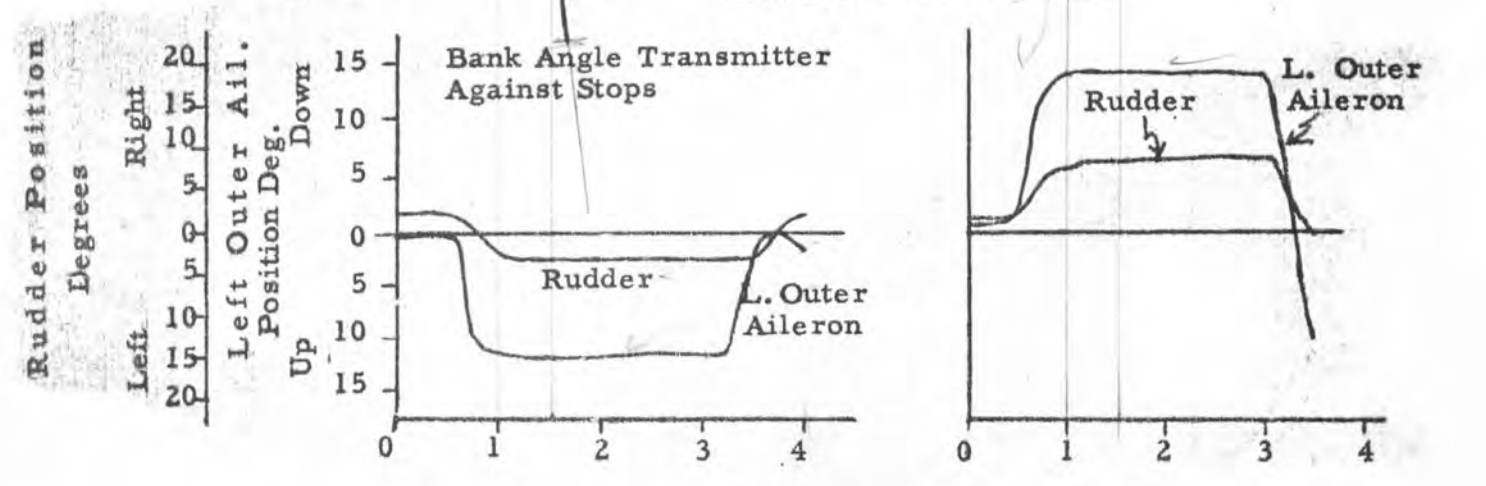
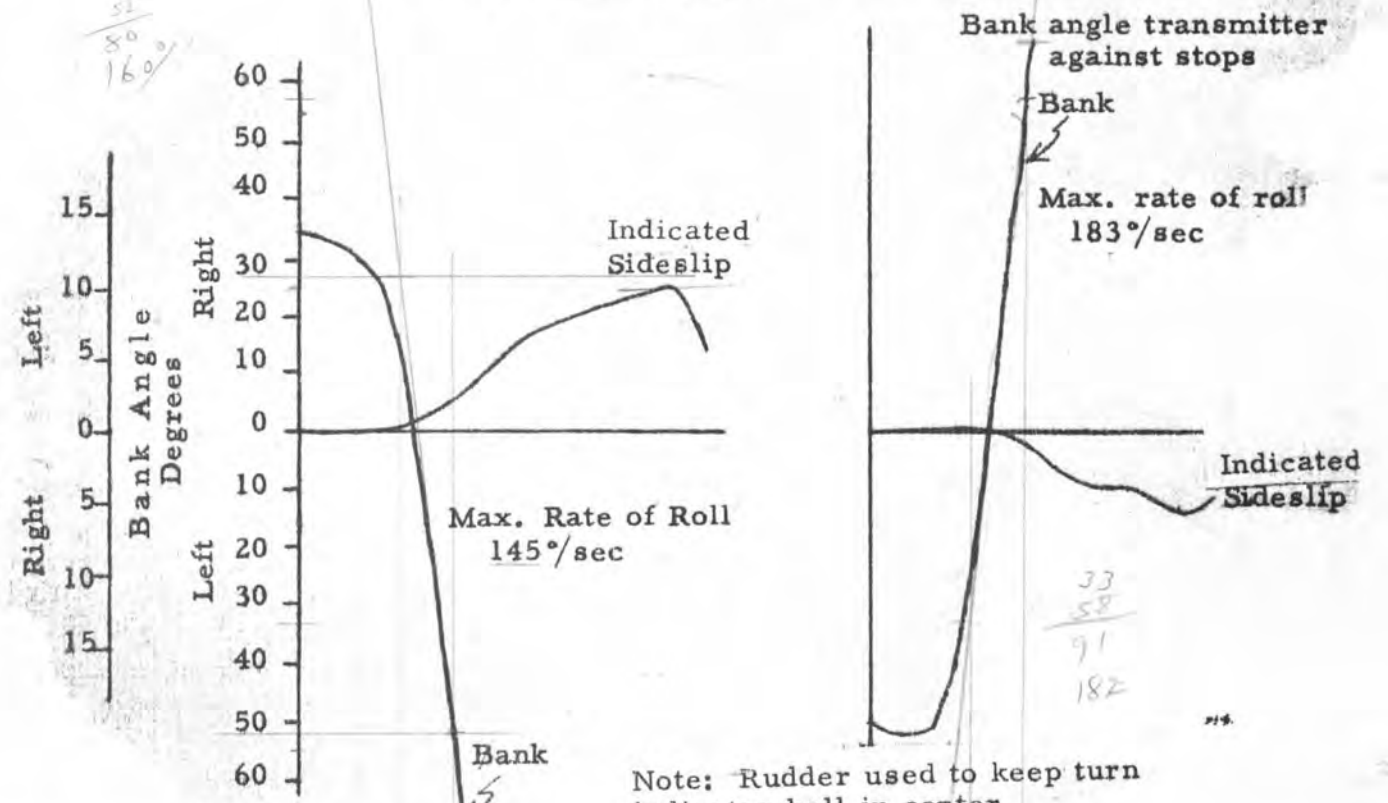
ADVERSE YAW
YF-100A, USAF No. 52-5754
Power Configuration (A/B Off)

TRIM CONDITIONS

CAS 244 knots Altitude 45000 feet
CG 30.6 %MAC Weight 22500 lbs
Rudder Pos. 2° left L. Outer Ail. Pos. .5° Dn
No external stores attached

LEFT ROLL

RIGHT ROLL



Time - Seconds
APPENDIX I

FIG No 88
SUMMARY OF ROLLING CHARACTERISTICS
YF-100A USAF No 52-5754
POWER CONFIGURATION
POWER FOR LEVEL FLIGHT
RUDDER PEDALS FIXED
ALTITUDE ~ 45000 FEET
NO EXTERNAL STORES ATTACHED

ALL DATA AT APPROXIMATELY
3/4 STICK THROW

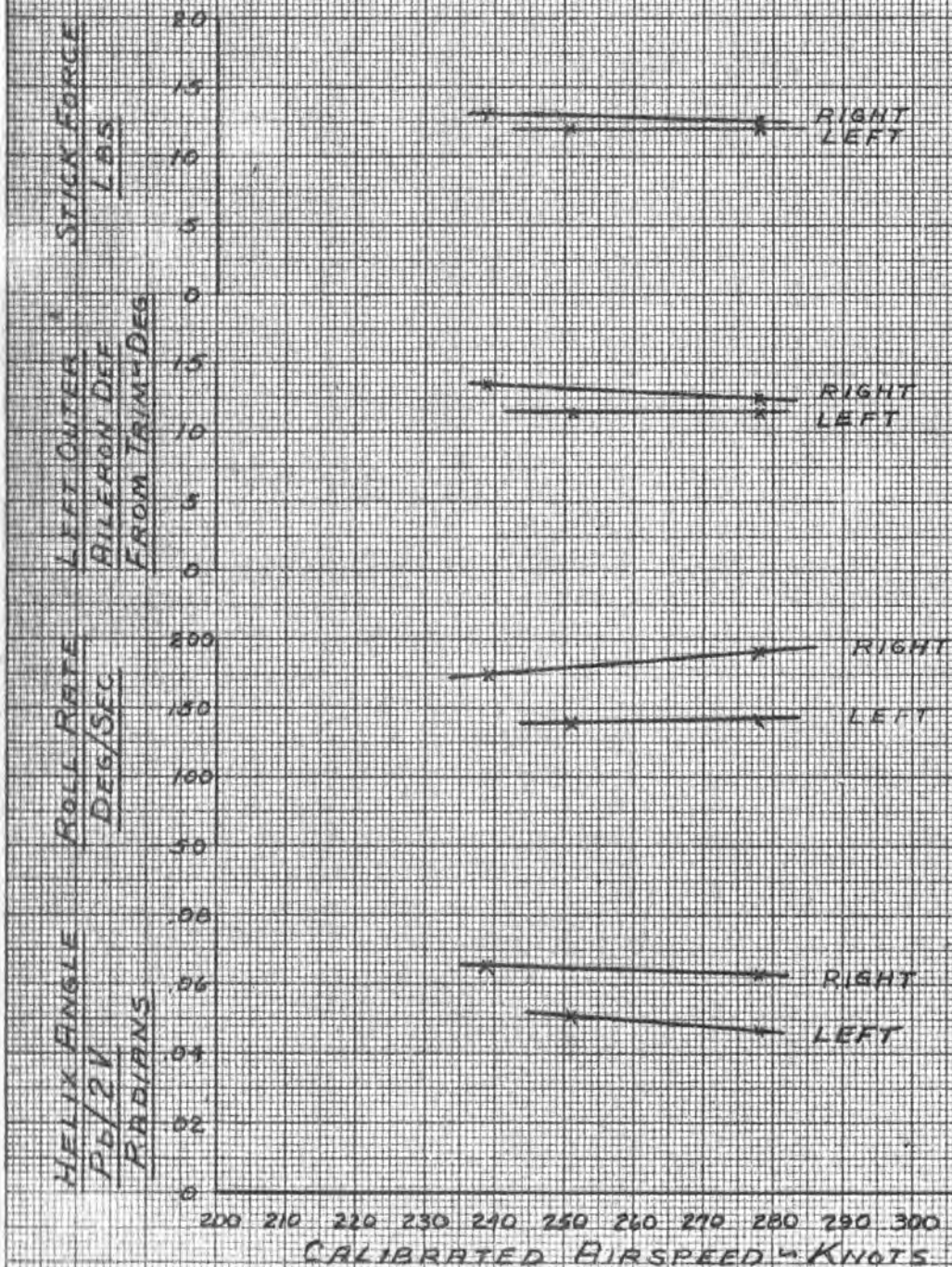


FIG No 89

SUMMARY OF ROLLING CHARACTERISTICS

YF-100A USAF No 52-5754

POWER CONFIGURATION

POWER FOR LEVEL FLIGHT

RUDDER PEDALS FIXED

ALTITUDE ~11500 FEET

NO EXTERNAL STORES ATTACHED

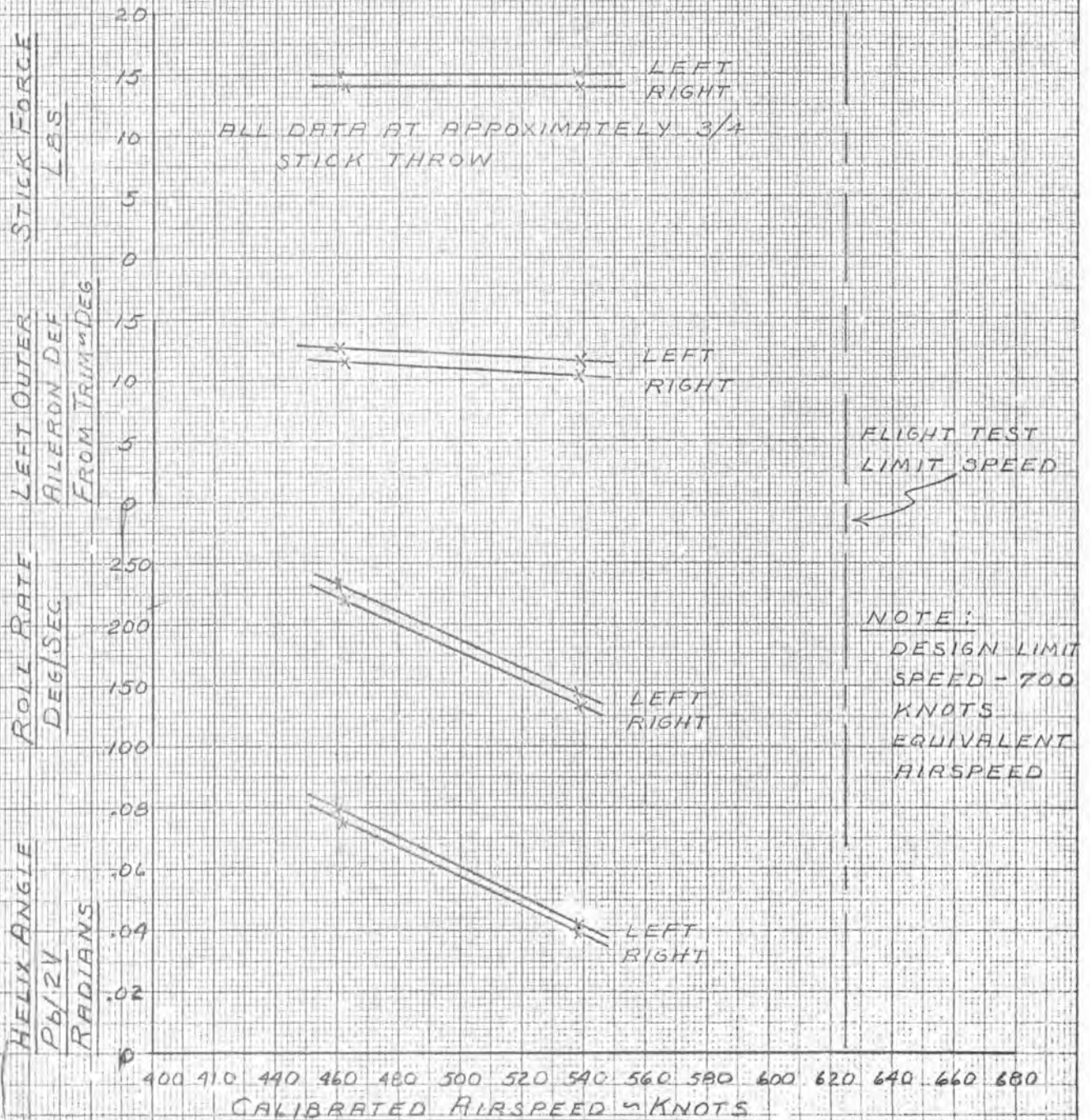


FIG No 90

AILERON CHARACTERISTICS
YF-100A USAF No 52-5754

POWER CONFIGURATION

POWER FOR LEVEL FLIGHT

ALTITUDE = 45000 FEET

SYM	CAS KNOTS	GR WT LBS	L OUTER AIL POS ° DEG	
○	270	23000	.5° DOWN	RUDDER FIXED
△	245	22500	.5° DOWN	RUDDER FIXED
▲	244	22500	.5° DOWN	FULL RUDDER

NO EXTERNAL STORES ATTACHED

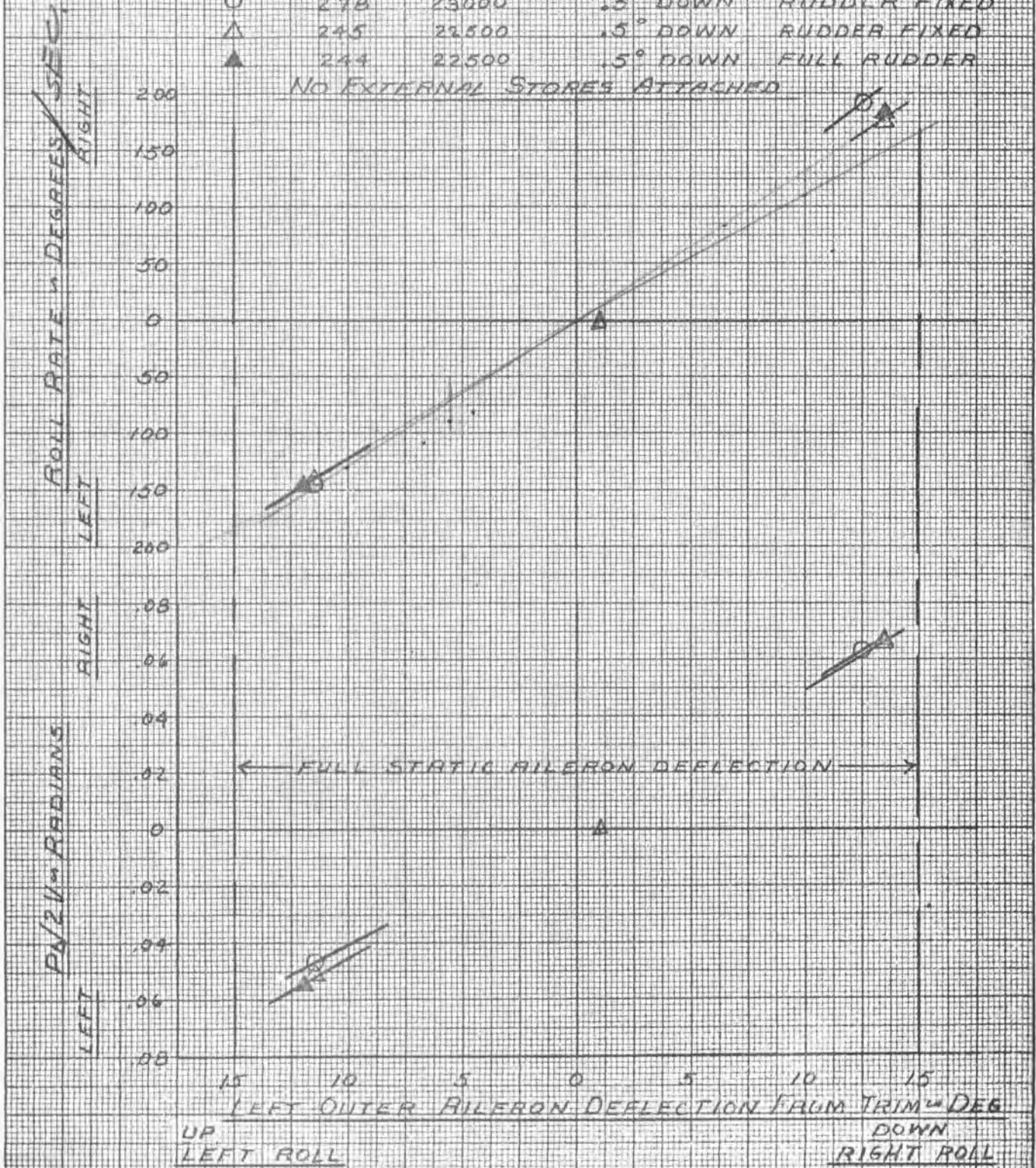


FIG No 91

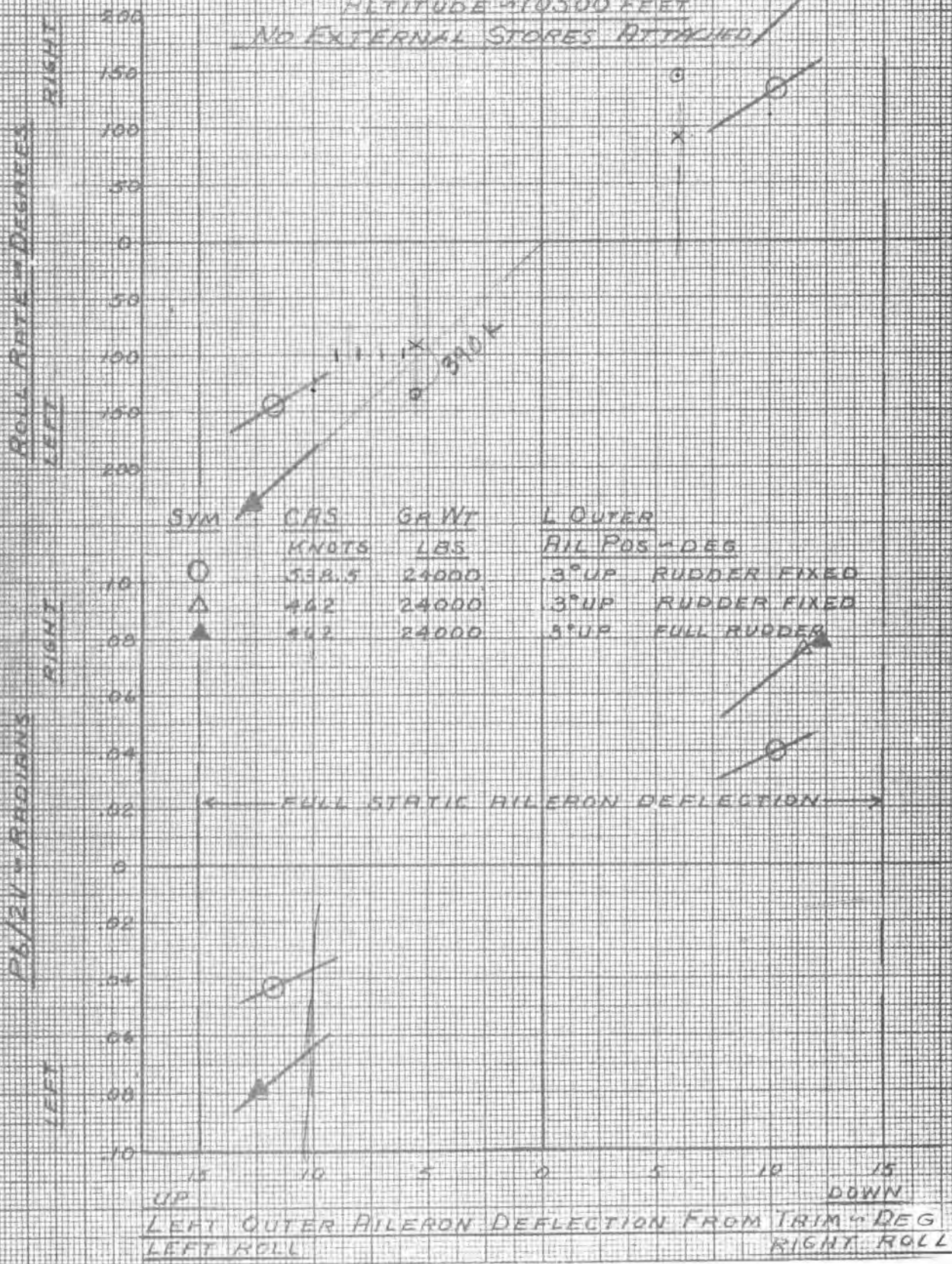
AILERON CHARACTERISTICS
YF-100A USRF No 52-5754

POWER CONFIGURATION

POWER FOR LEVEL FLIGHT

ALTITUDE ~10500 FEET

NO EXTERNAL STORES ATTACHED



SYM	CAS	GAWT	L OUTER
	KNOTS	LBS	AIR POS - DEG
○	538.5	24000	3° UP RUDDER FIXED
△	442	24000	3° UP RUDDER FIXED
▲	442	24000	3° UP FULL RUDDER

FIG No 92

AILERON CHARACTERISTICS
YF-100A USAF No 52-5754

POWER APPROACH CONFIGURATION

POWER FOR LEVEL FLIGHT

ALTITUDE 10400 FEET

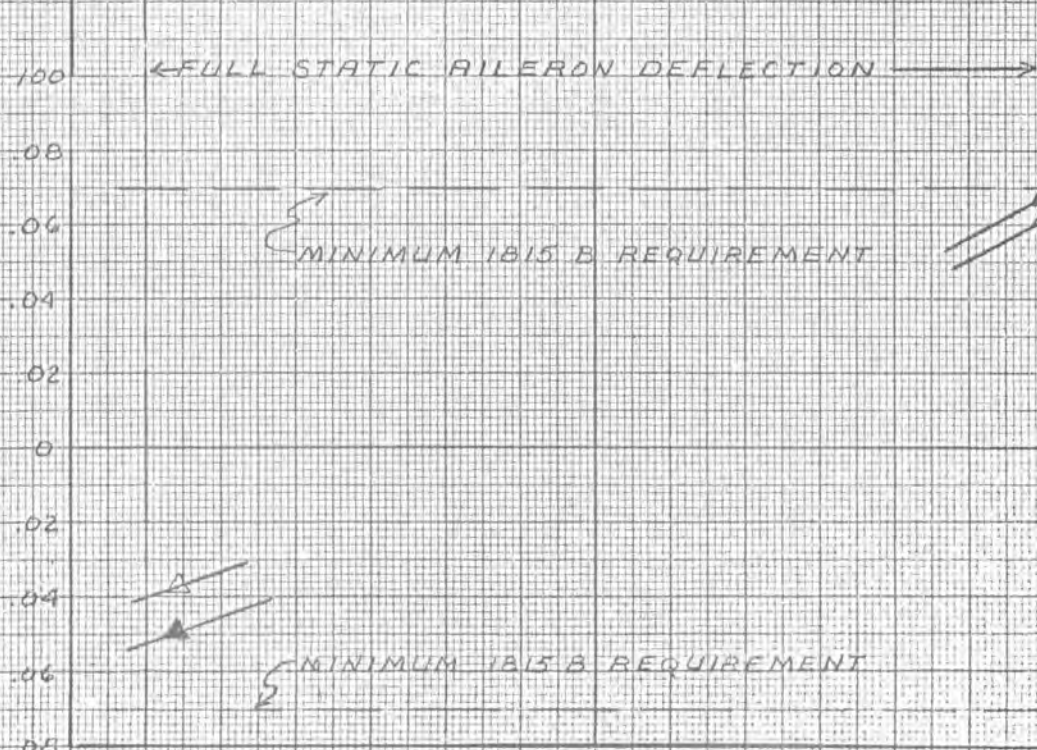
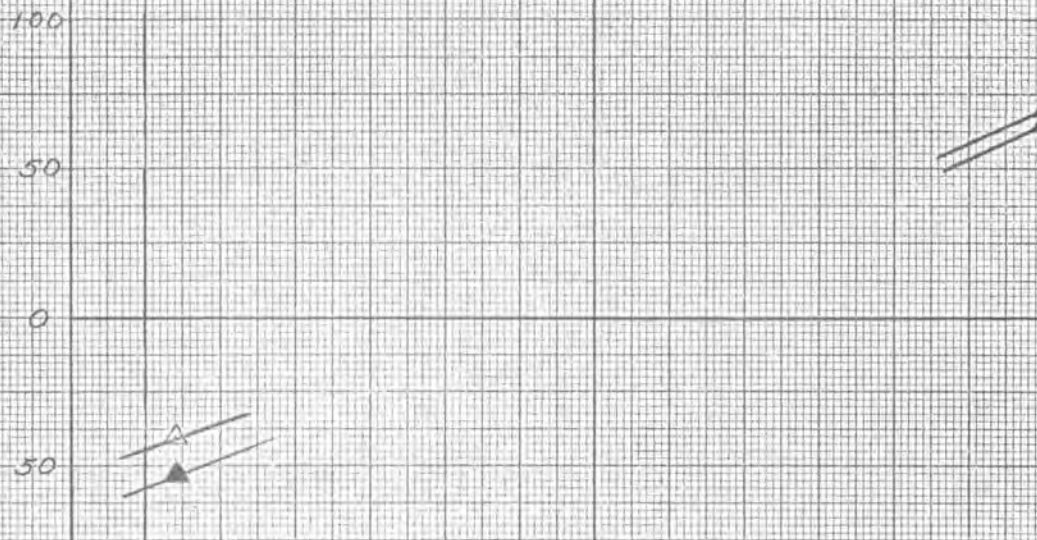
NO EXTERNAL STORES ATTACHED

SYM	CAS	GR WT	L. OUTER	
	KNOTS	LBS	AIL POS ^o DEG	
△	167	23000	0	RUDDER FIXED
▲	167	23000	0	FULL RUDDER

ROLL RATE^o DEGREES
RIGHT
LEFT

ROLL RATE^o DEGREES
RIGHT
LEFT

ROLL RATE^o DEGREES
RIGHT
LEFT



UP DOWN
LEFT OUTER AILERON DEFLECTION FROM TRIM^o DEG
LEFT ROLL RIGHT ROLL

Figure No. 93

MAXIMUM ALLOWABLE DEFLECTION

AILERON ROLLS

YF-100A, USAF No. 52-5754

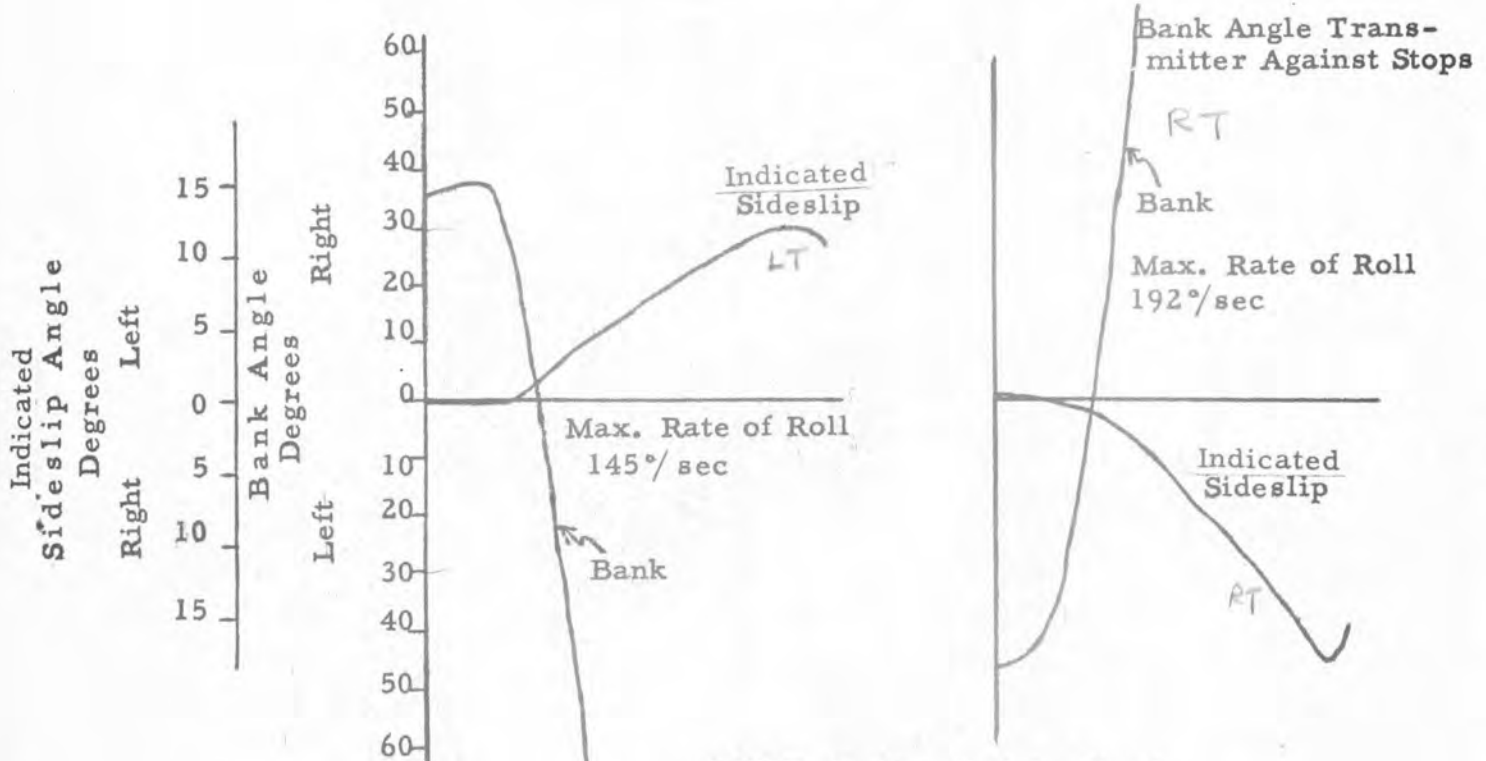
Power Configuration(A/B On)

TRIM CONDITIONS

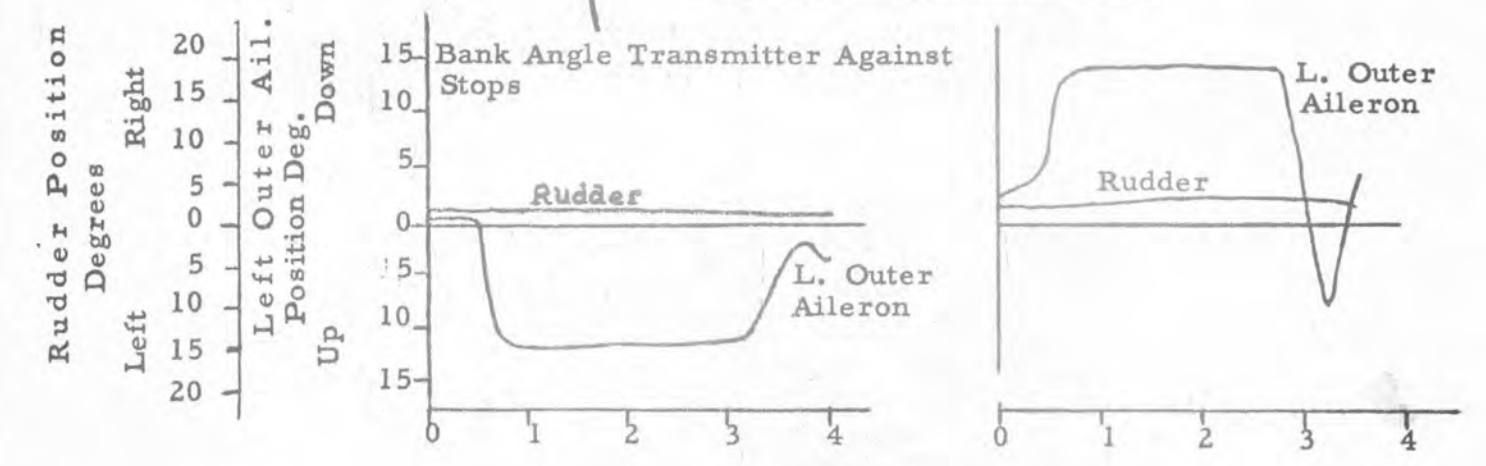
CAS 278 knots Altitude 45,000 feet
 CG 31.4 %MAC Weight 23,000 lbs
 Rudder Pos. 1.5° left L. Outer Ail. Pos. 5° Dn
 No external stores attached

LEFT ROLL

RIGHT ROLL



NOTE: Rudder Pedals Fixed



MAXIMUM ALLOWABLE DEFLECTION
AILERON ROLLS

YF-100A, USAF No. 52-5754
Power Configuration(A/B On)

TRIM CONDITIONS

CAS 538.5 knots Altitude 10,500 feet
CG 30.4 % MAC Weight 24,000 lbs
Rudder Pos. 0 L. Outer Ail. Pos. .3° up
No external stores attached

LEFT ROLL

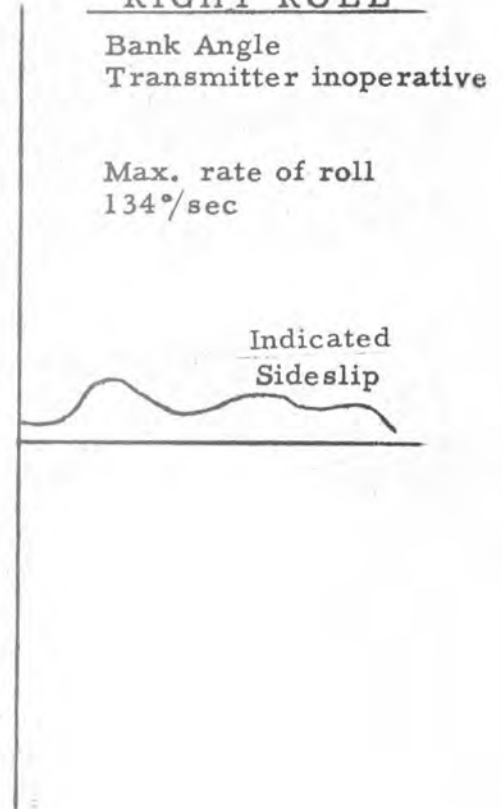
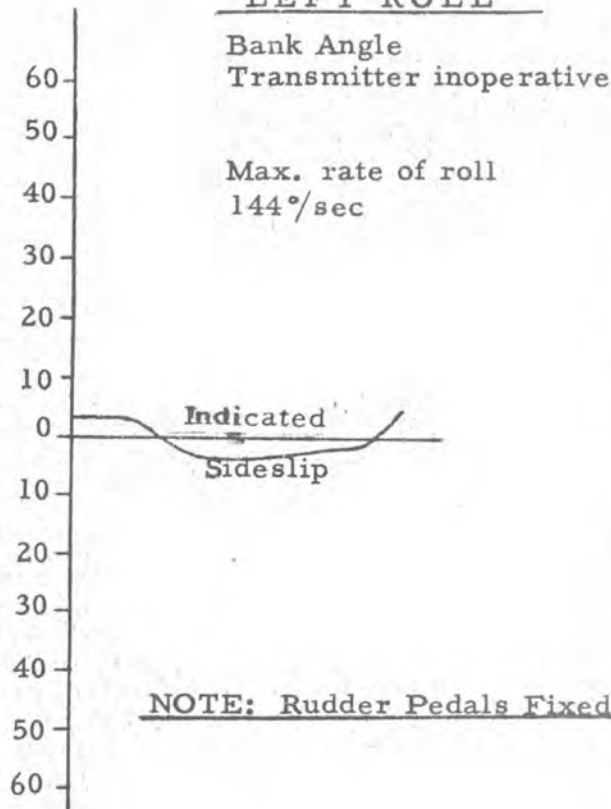
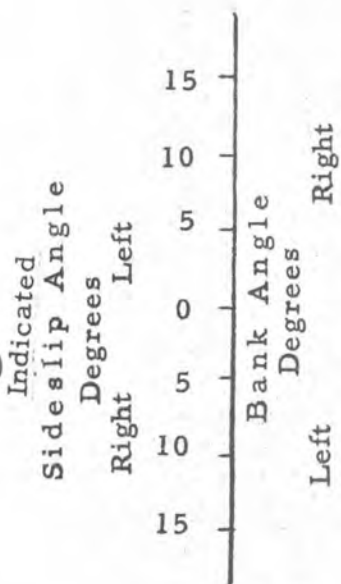
RIGHT ROLL

Bank Angle
Transmitter inoperative

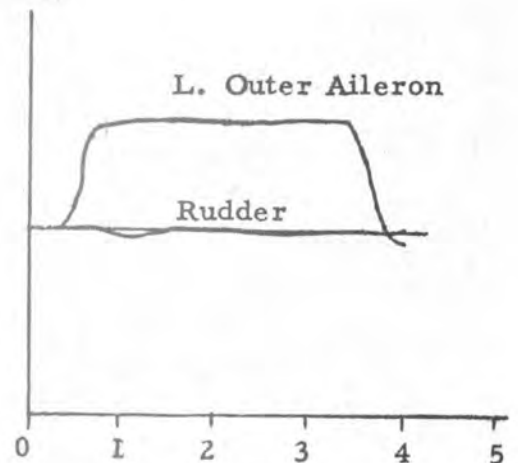
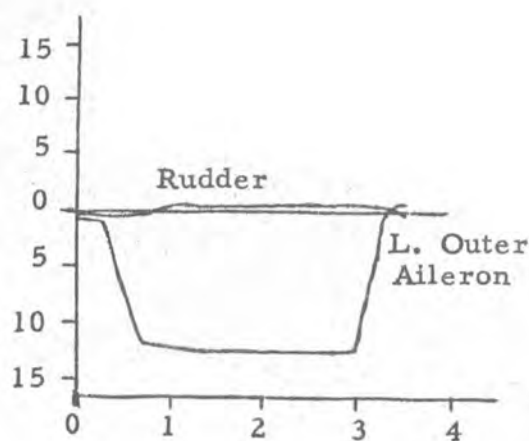
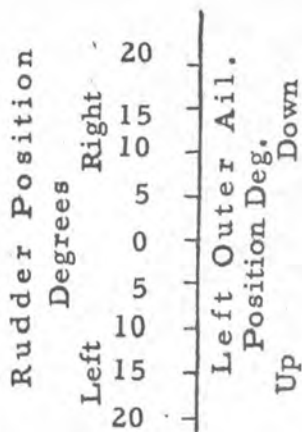
Bank Angle
Transmitter inoperative

Max. rate of roll
144°/sec

Max. rate of roll
134°/sec



NOTE: Rudder Pedals Fixed



Time - Seconds

APPENDIX I

APPENDIX II

INDEX

General Information and Photographs

YF100A USAF, No. 52-5754

	<u>Page No.</u>
A. Three-View Drawing	2
B. Dimensions and Design Data	3--5
C. Operational Limitations	6
D. Fuel System	6
E. Power Plant	7
F. Weight and Balance	7
G. Test Instrumentation	8--9
H. Control System Diagrams:	
Stabilizer	10
Aileron	11
Rudder	12
I. Photographs	
Front View	13
Left Front View	14
Left Front View (canopy open and ladder attached)	15
Left Side View	16
Rear View	17
Right Rear View	18
Right Side View	19
Front View (tanks on)	20
Left Front View (tanks on)	21
Rt. Rear View (tanks on)	22
Rear View (tanks on)	23
Cockpit (Instrument panel and left and right console)	24
Slats open and closed	25
Speedbrake open and closed	26
Oil cooler door open and closed	27
Surge bleed valve door open and closed	28
Stabilizer full nose up and full nose down	29
Parachute door open and closed	30
Exhaust nozzle open and closed	31

THREE VIEW DRAWING

45.77 FT.

14.44 FT.

7.5'

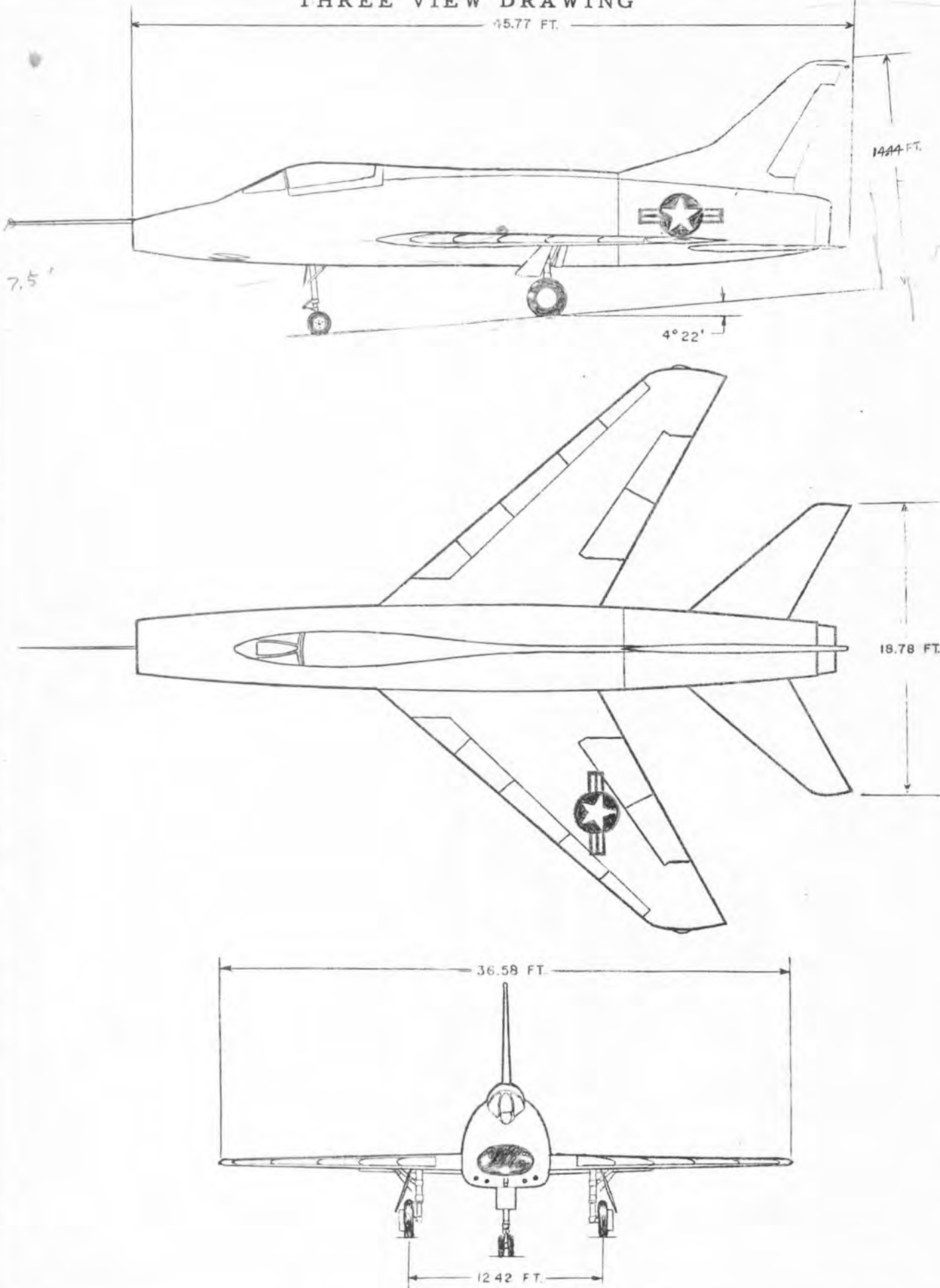
4° 22'

18.78 FT.

36.58 FT.

12.42 FT.

APPENDIX II



B. DIMENSIONS AND DESIGN DATA

The following dimensions and design data were obtained from North American Aviation Inc., Report No's NA-53-439 and NA-51-857.

1. Wing

Area	376.02 ft.
Span	36.58 ft.
Sweepback of 25 per cent element	45°
Aspect ratio	3.56
Taper ratio	.30
Mean aerodynamic chord	135.93
Fuselage station of 25 per cent C_w (WP -23.00)	301.97
Dihedral angle	0°
Incidence angle of root chord	0°
Incidence angle of tip chord	0°
Airfoil sections, root and tip, NACA 64A007 (parallel to plane of symmetry)	

2. Ailerons

(Flat sided type)	
Area (aft of hinge line), each	19.32 sq. ft.
Area moment (normal to hinge line)	41.51 cu. ft.
Span	93.67 in.
Spanwise location, inboard end	.323 $b_w/2$
Spanwise location, inboard end	.750 $b_w/2$
Ratio of aileron chord to wing chord	.250
Deflection, maximum	15° up, 15° down
Irreversible hydraulic boost and artificial feel	
Aerodynamic balance	none
Static balance	internal, lead weights

3. Leading Edge Slats

Span, equivalent	152.52 in.
Segments	5
Spanwise location, inboard end	.246 $b_w/2$
Spanwise location, outboard end	.941 $b_w/2$
Ratio of slat chord to wing chord (Parallel to F. R. L.)	.200
Rotation, maximum	10°
Center of rotation(measured in plane containing 13 per cent element line, downward from and normal to W. R. P.)	263 per cent slat chord

4. Horizontal Tail
- | | |
|---|----------------------------------|
| Area | 99.00 sq. ft. |
| Span | 18.78 ft. |
| Sweepback of 25 percent element | 45° |
| Aspect ratio | 3.56 |
| Taper ratio | .30 |
| Mean aerodynamic chord | 69.72 in. |
| Fuselage station of $.25\bar{C}_w$ (WP
-28.97) | 479.64 |
| Dihedral angle | 0° |
| Airfoil sections, root and tip,
NACA 64A007 (parallel to
plane of symmetry) | <i>check this for F.1007 etc</i> |
5. Horizontal stabilizer
- | | |
|--|---------------------------|
| Area | 67.35 sq. ft. |
| Area moment (normal to C_L
rotation) | 337.36 cu. ft. |
| Deflection, maximum | 5° nose up, 25° nose down |
| Axis of rotation | W.L. -22 F.S. 485 |
| Irreversible hydraulic boost and artificial feel | |
6. Vertical Tail
- | | |
|---|---------------|
| Area (excluding dorsal fin) | 39.84 sq. ft. |
| Span, unblanketed | 7.46 ft. |
| Sweep of 25 percent element | 45° |
| Aspect ratio | 1.49 |
| Taper ratio | .30 |
| Mean aerodynamic chord | 66.08 in. |
| Fuselage station of $25\bar{C}_v$ (WP
57.63) | 488.70 |
| Airfoil sections, root and tip,
NACA 64A007 (parallel to F. R. L.) | |
7. Vertical Fin
- | | |
|-----------------------------|---------------|
| Area (excluding dorsal fin) | 33.54 sq. ft. |
| Area of dorsal fin | 3.16 sq. ft. |
| Span, unblanketed | 7.46 ft. |
| Offset | 0° |
8. Rudder
- | | |
|---------------------------------------|--------------|
| Area (aft of hinge line) | 6.30 sq. ft. |
| Area moment (normal to hinge
line) | 6.37 cu. ft. |

Span	5.77 ft.
Spanwise location, inboard end	.031 b _v
Spanwise location, outboard end	.804 b _v
Ratio of rudder chord to vertical tail chord	.200
Deflection, maximum	20° L, 20° R
Aerodynamic balance	Overhanging, unsealed
9. Fuselage	
Side area (total)	277.86 sq. ft.
Length(afterburner nozzle closed)	45.03
Depth over canopy, maximum	76.50
Width, maximum	67.00
Fineness ratio(afterburner nozzle closed)	7.75
10. Speed Brake	
Area, surface	14.14 sq. ft.
Deflection, maximum	50°
Fuselage station of hinge line (WP -41.00)	223.75
11. Total Surface Area of Clean Airplane	1519.60 sq. ft.
12. Total Frontal Area of Clean Airplane	57.34 sq. ft.
13. Landing Gear	
Main gear tire size	30 x 8.8
Nose gear tire size	dual 18 x 4.4
Hydraulic steering	
Nose gear steering angle	35°L, 35°R
14. External Stores	
275 gallon drop tanks	2
Length	250 in.
Diameter, maximum	25 in.
Fin incidence	2°

C. OPERATIONAL LIMITATIONS

1. Structural design gross weight at combat radius (clean aircraft) 23,996 lbs.
2. Design load factors at (1 above) for symmetrical flight are:

Positive	+7.33
Negative	-3.33

Note: because structural integrity tests have not yet been made on the YF-100A aircraft, limit load factors for Phase II tests were given by the contractor as:

Positive	+5
Negative	-2

3. Maximum structural design flight weight (external tanks) 28,561 lbs.
4. Design load factors at (3 above) for symmetrical flight are:

Positive	+6
Negative	-3

 Flight test load factors:

Positive	+4
Negative	-2

5. Maximum take-off weight 28,561 lbs.
6. Minimum flight weight 19,448 lbs.
7. Ground load factors:

Design landing weight of	22,205 lbs.
Main gear 3.0	
Nose gear 3.0	

8. Limit airspeeds

(a) Clean aircraft above 3,000 feet	700 knots
(b) With external fuel tanks above 3,000 feet	600 knots

 Note: limit speed allowed for flight test was 625 knots IAS

9. Most forward aft cg position possible in flight:

<u>Clean</u>	
Fwd.	29.5 per cent of MAC
Aft	35.5 per cent of MAC

D. FUEL SYSTEM

1. Main system (internal). Fuel is supplied to the engine from five bladder

cells interconnected to form three tanks with capacities as follows:

a. sump tank (forward cell and underwing cell)	417 gal.
b. intermediate tank (left and right intermediate cells)	221 gal.
c. aft tank	119 gal.
total capacity	<u>757 gal.</u>

2. External fuel system:

Provisions are made for the installation of a 275 gallon droppable combat fuel tank beneath each wing.

total capacity 2 tanks 550 gal.

E. POWER PLANT

PRATT and WHITNEY XJ57-P7 ENGINE

<u>Power Conditions</u>	<u>Thrust lbs</u>	<u>STOL</u>
Uninstalled maximum rating with afterburning	13,200	1.98
Uninstalled military rating without afterburning	8,450	.86
Installed maximum rating with afterburning (9,495 rpm)*	10,200	2.2
Installed military rating without afterburning (9,495 rpm)*	6,800	.91

* rpm values are based on a compressor inlet temperature of 15°C.

F. WEIGHT AND BALANCE

<u>Flight no.</u>	<u>Basic* Weight-lbs</u>	<u>Crew lbs</u>	<u>Other lbs</u>	<u>Gross Wt lbs</u>	<u>CG ** %o MAC</u>
1-27	25,169	200	0	25,369	31.9
28	25,169	200	24 ^①	25,393	32.0
29	23,062 ^②	200	24 ^①	23,286	31.8
30	22,490 ^②	200	24 ^①	22,714	31.1
31-33	25,169	200	24 ^①	25,393	32.0
34	25,169	200	0	25,369	31.9
35	25,169	200	24 ^①	25,393	32.0
36	25,169	200	0	25,369	31.9
37	25,169	200	400 ^③	25,769	32.2
38-39	25,169	200	424 ^④	25,793	32.4

* Includes test instrumentation, 1289 lbs ballast, 757 gallons of fuel and 5.5 gallons of oil.

**CG computed with gear up; add 0.1 percent MAC for gear down percent MAC.

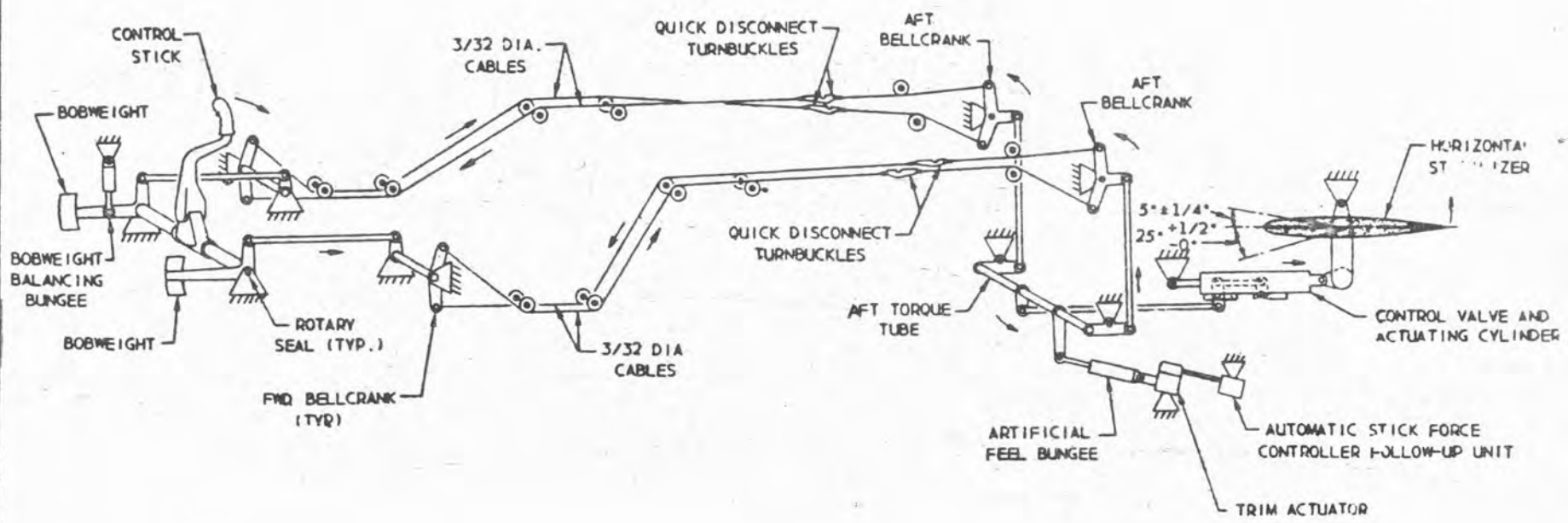
- ① Drag chute and deploy cable.
- ② Partial fuel load.
- ③ Two empty 275 gallon external fuel tanks.
- ④ Two empty 275 gallon external fuel tanks and drag chute and deploy cable.

G. TEST INSTRUMENTATION

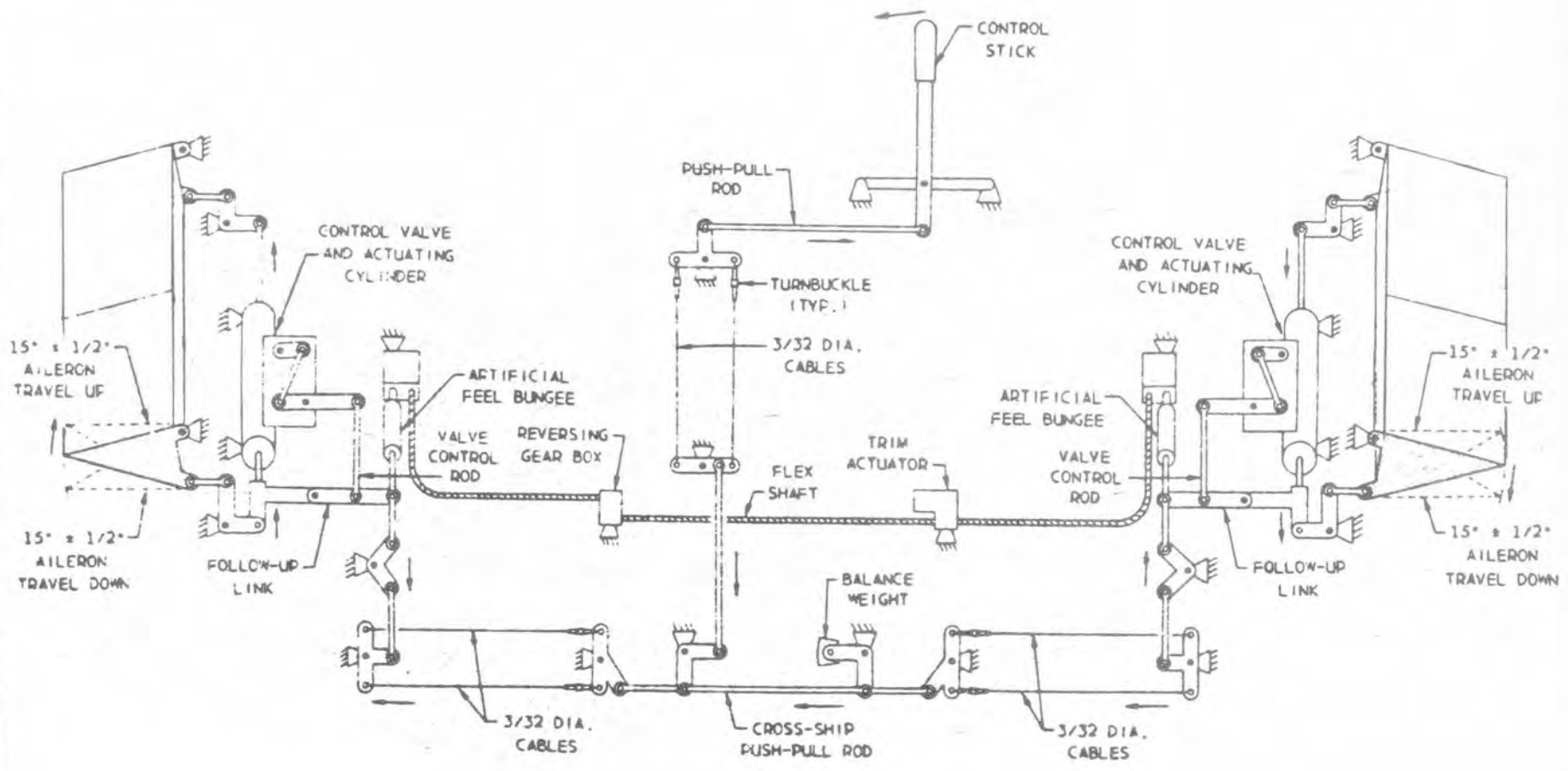
Complete instrumentation for performance and stability and control was installed in the test aircraft by the contractor. All instrument calibrations were accomplished by the contractor. The calibrations on several of the more critical instruments were checked by the AFFTC Instrumentation Branch. The following instruments were used for the Phase II tests:

1. Pilot's panel:
 - Airspeed
 - Altimeter
 - Free air temperature
 - Engine rpm (high pressure)
 - Exhaust gas temperature
 - Sideslip angle
 - Fuel quantity (forward cell)
 - Fuel quantity (intermediate cell)
 - Fuel quantity (aft cell)
 - Accelerometer
 - Engine fuel counter
 - A/B fuel counter
 - Tail pipe nozzle position
2. 35 mm Photorecorder:
 - Airspeed
 - Altitude
 - Free air temperature
 - Engine fuel counter
 - A/B fuel counter
 - Aft fuel cell quantity
 - Engine rpm (high pressure)
 - Engine rpm (low pressure)
 - Tailpipe temperature
 - Tailpipe total head pressure
 - Compressor inlet total head pressure
 - Fuel pressure at outlet of A/B control
 - Slat position
 - Vertical accelerometer
 - Clock
 - Oil cooler duct flap position
 - Nozzle position light
 - Surge bleed valve position lights

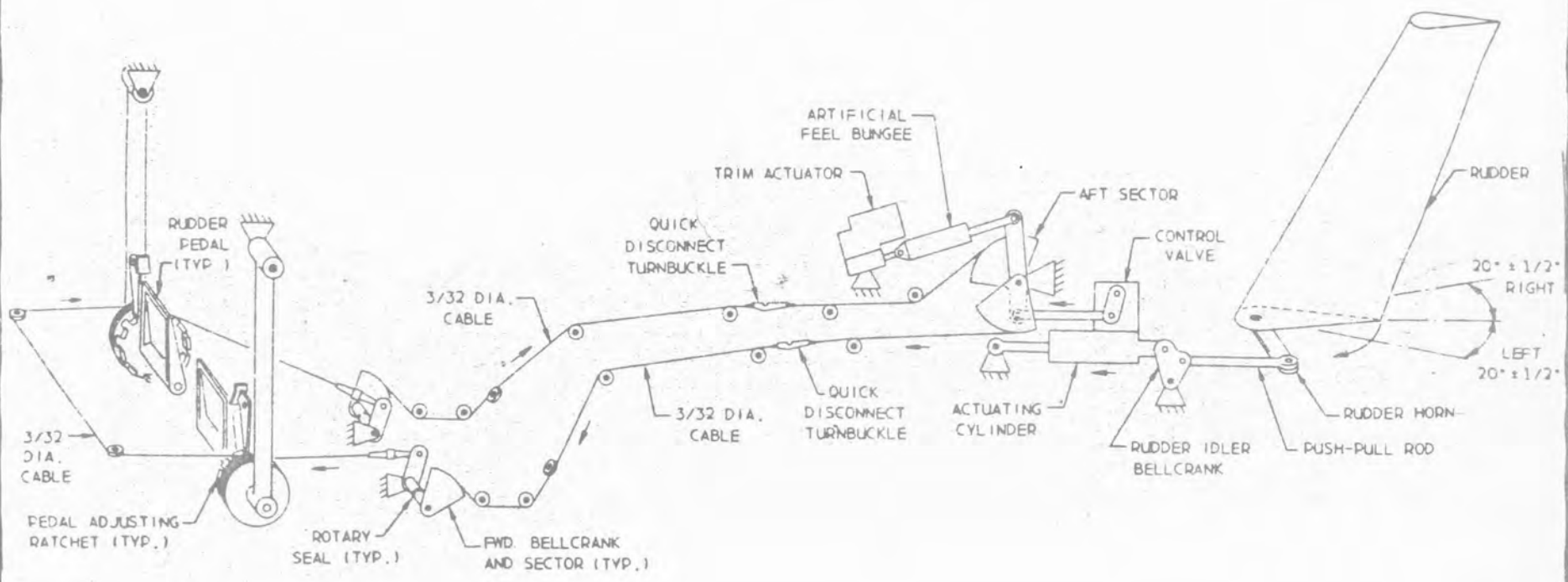
3. Eighteen channel Oscillograph:
 - Normal acceleration
 - Horizontal stabilizer position
 - Longitudinal stick position
 - Left aileron position (inboard)
 - Left aileron position (outboard)
 - Rudder position
 - Longitudinal stick force
 - Aileron stick force
 - Rudder pedal force
 - Speed brake position
 - Horizontal stabilizer trim actuator position
 - Angle of bank
 - Angle of attack
 - Sideslip angle
 - Photorecorder and pilot correlation trace
 - Nose wheel lift-off
 - Power control lever position



HORIZONTAL STABILIZER CONTROL SYSTEM
YF-100A



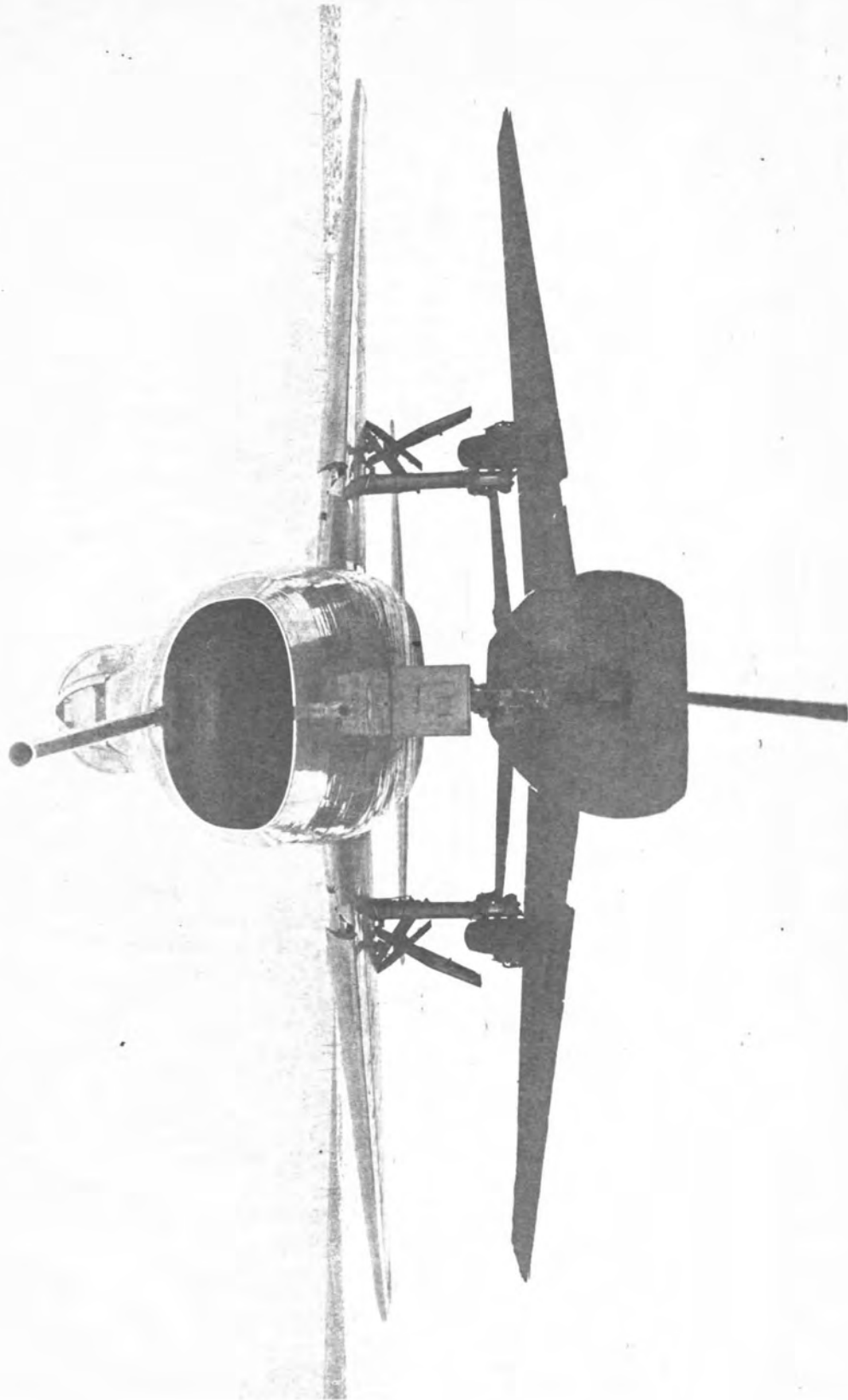
AILERON CONTROL SYSTEM
YF-100A



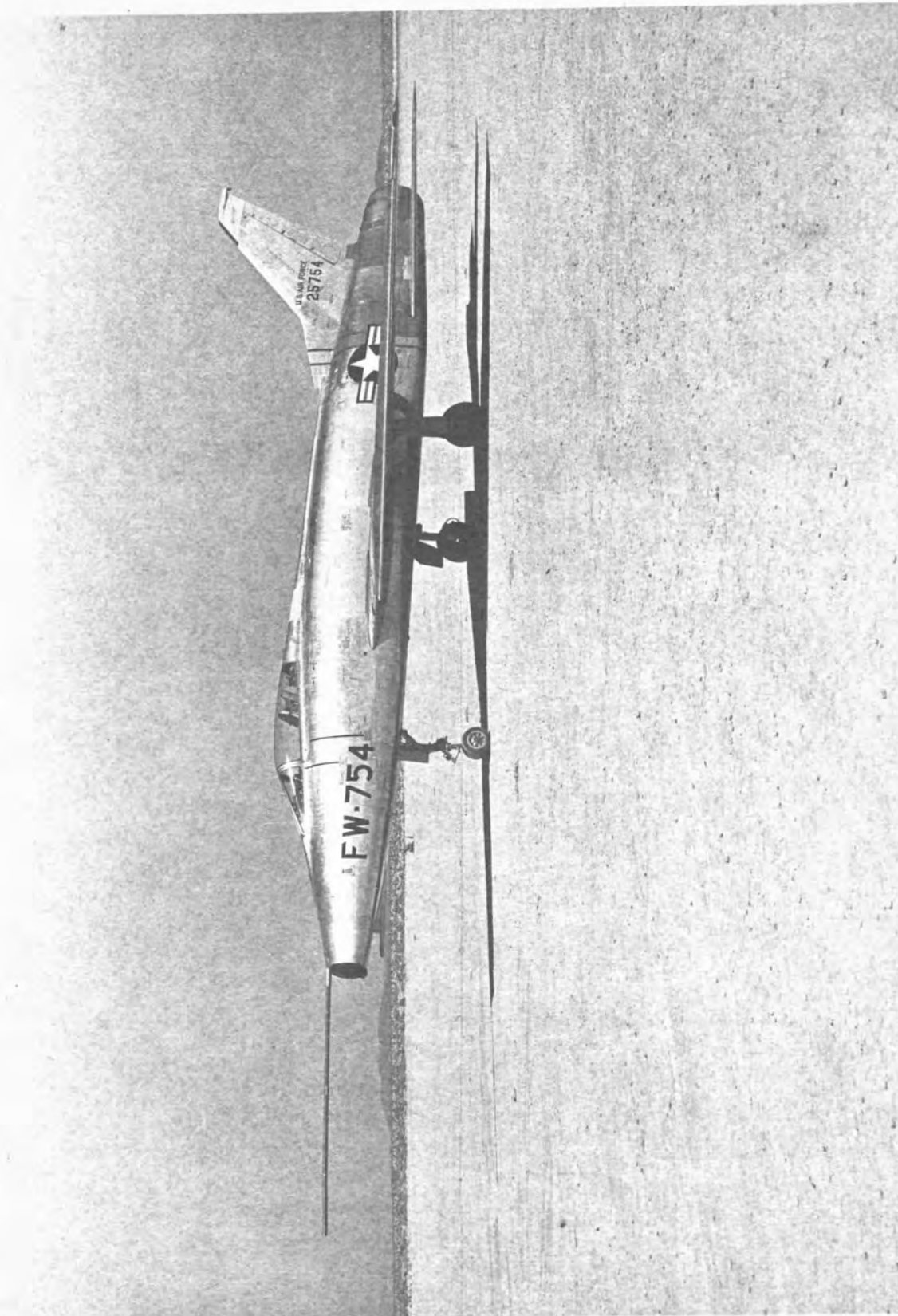
RUDDER CONTROL SYSTEM

YF-100A

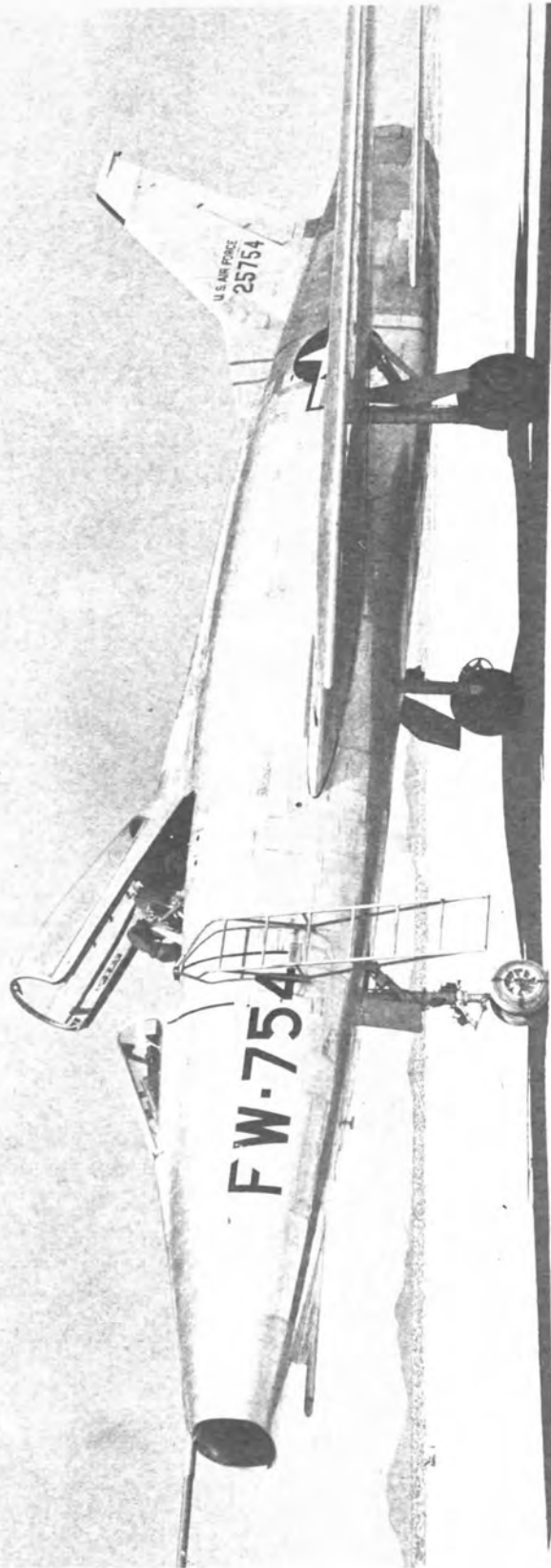




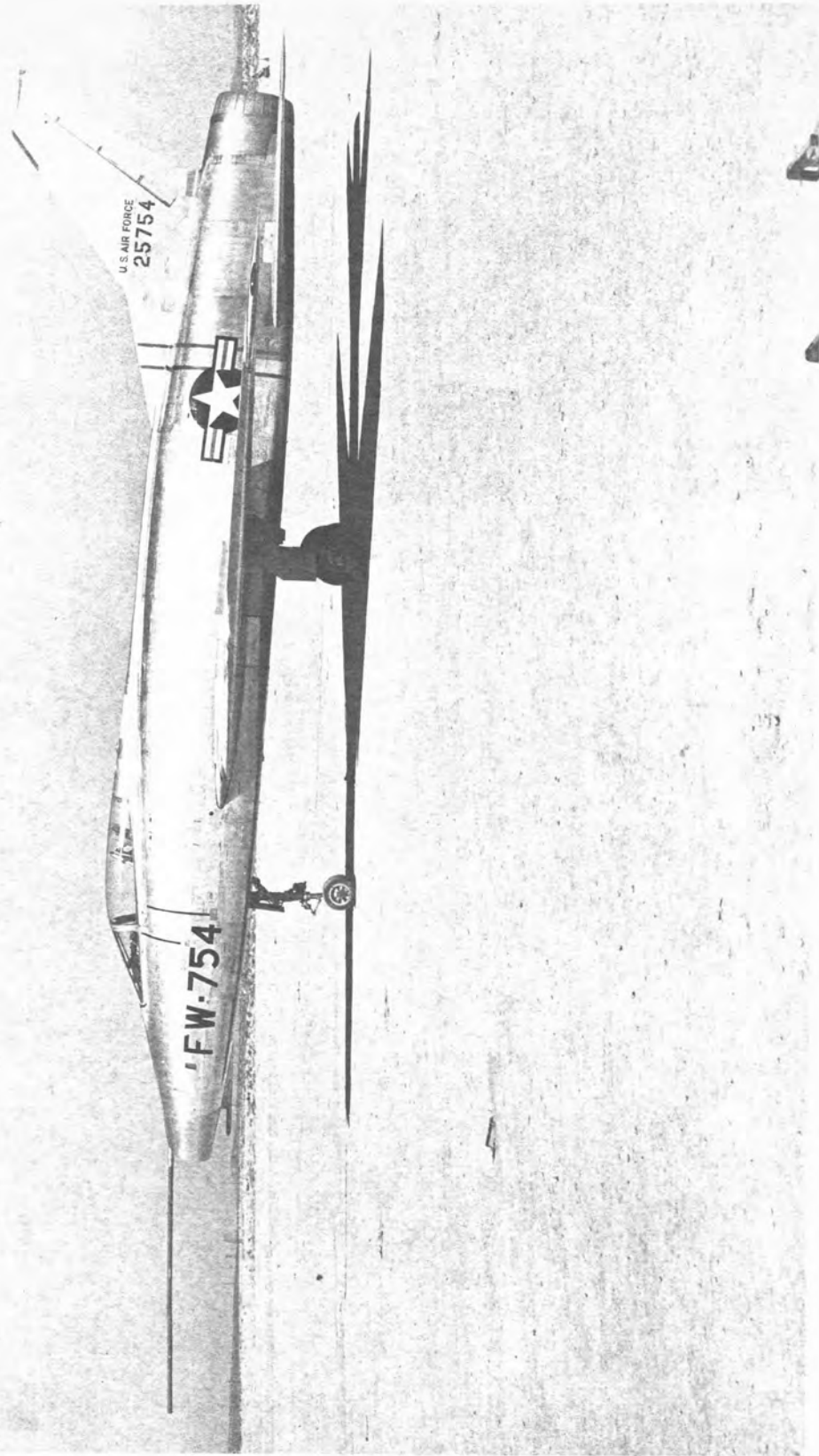
FRONT VIEW



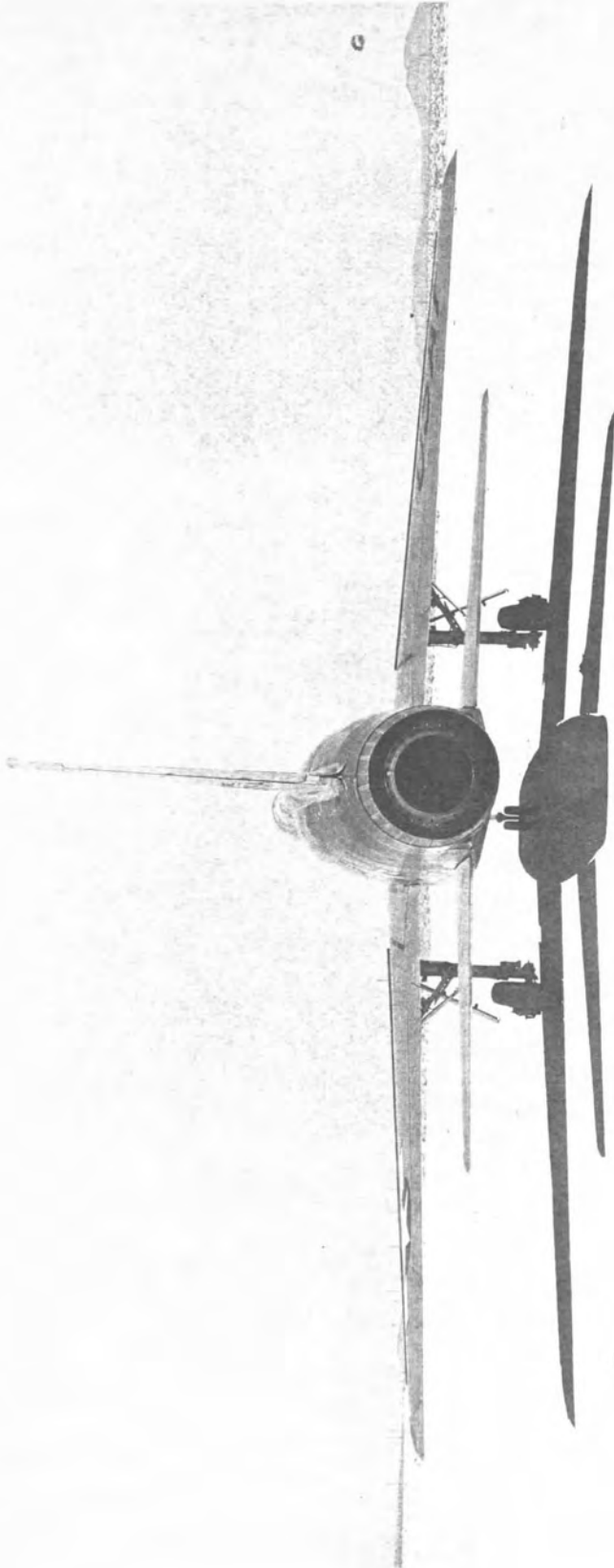
LEFT FRONT VIEW



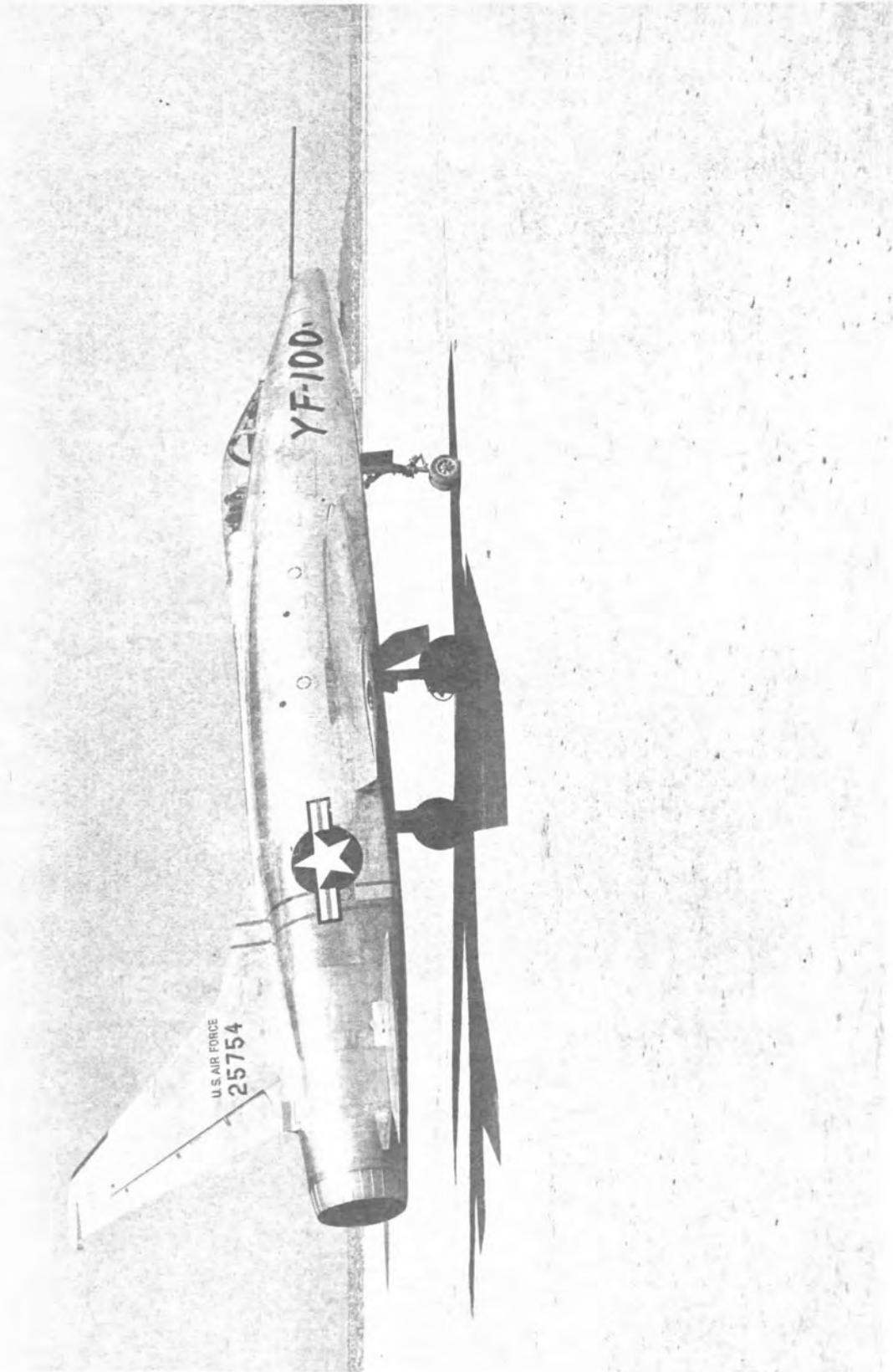
LEFT FRONT VIEW (Canopy open and ladder attached)



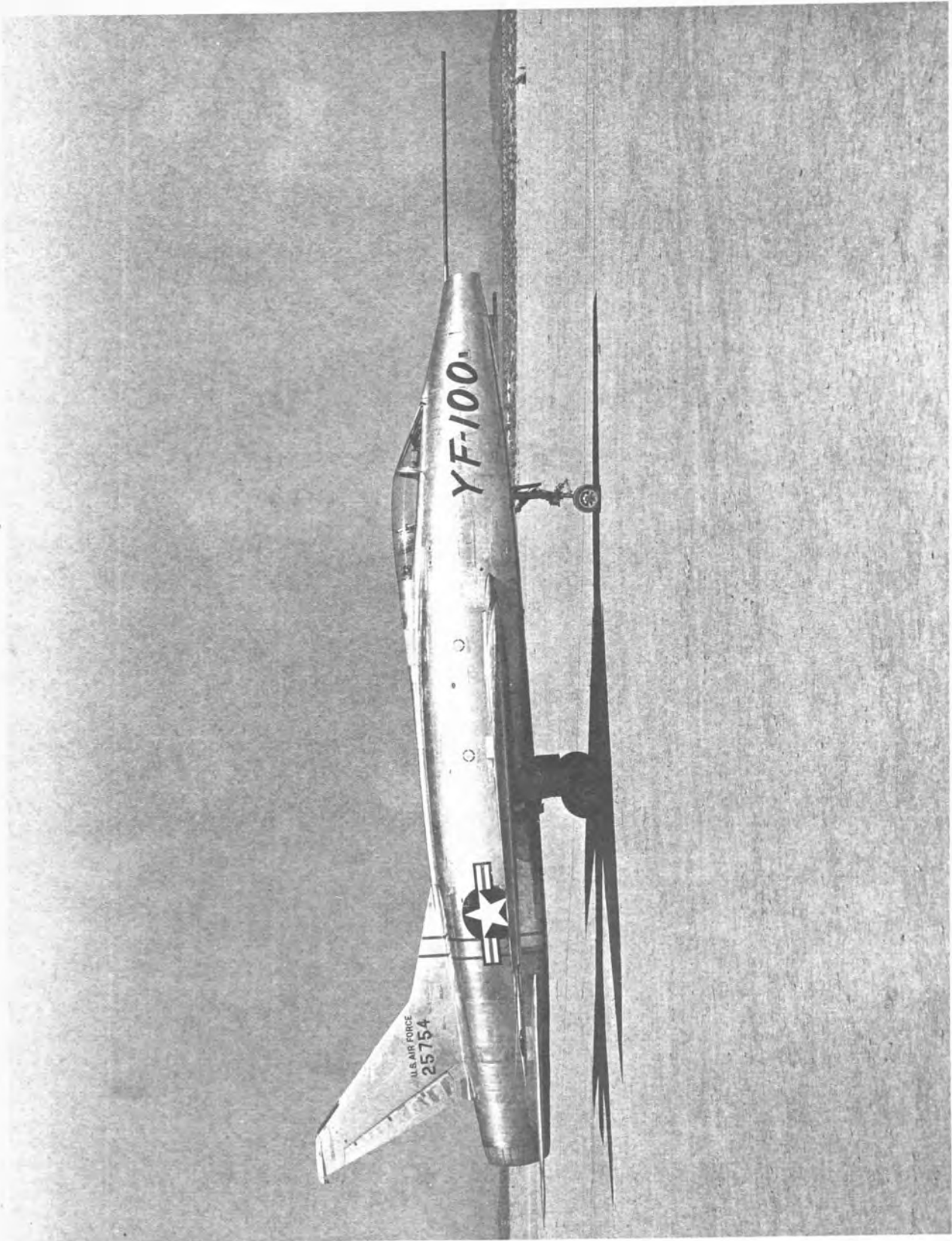
LEFT SIDE VIEW



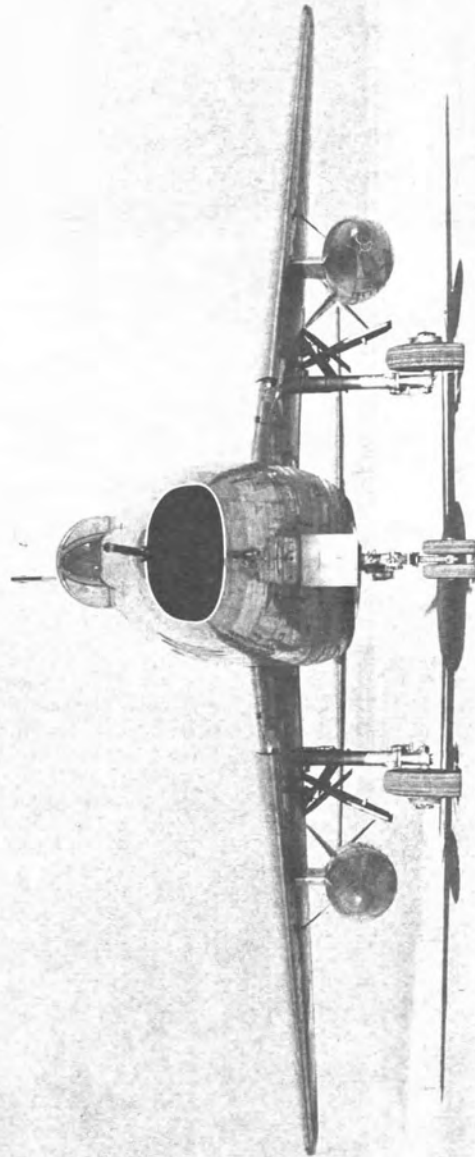
REAR VIEW



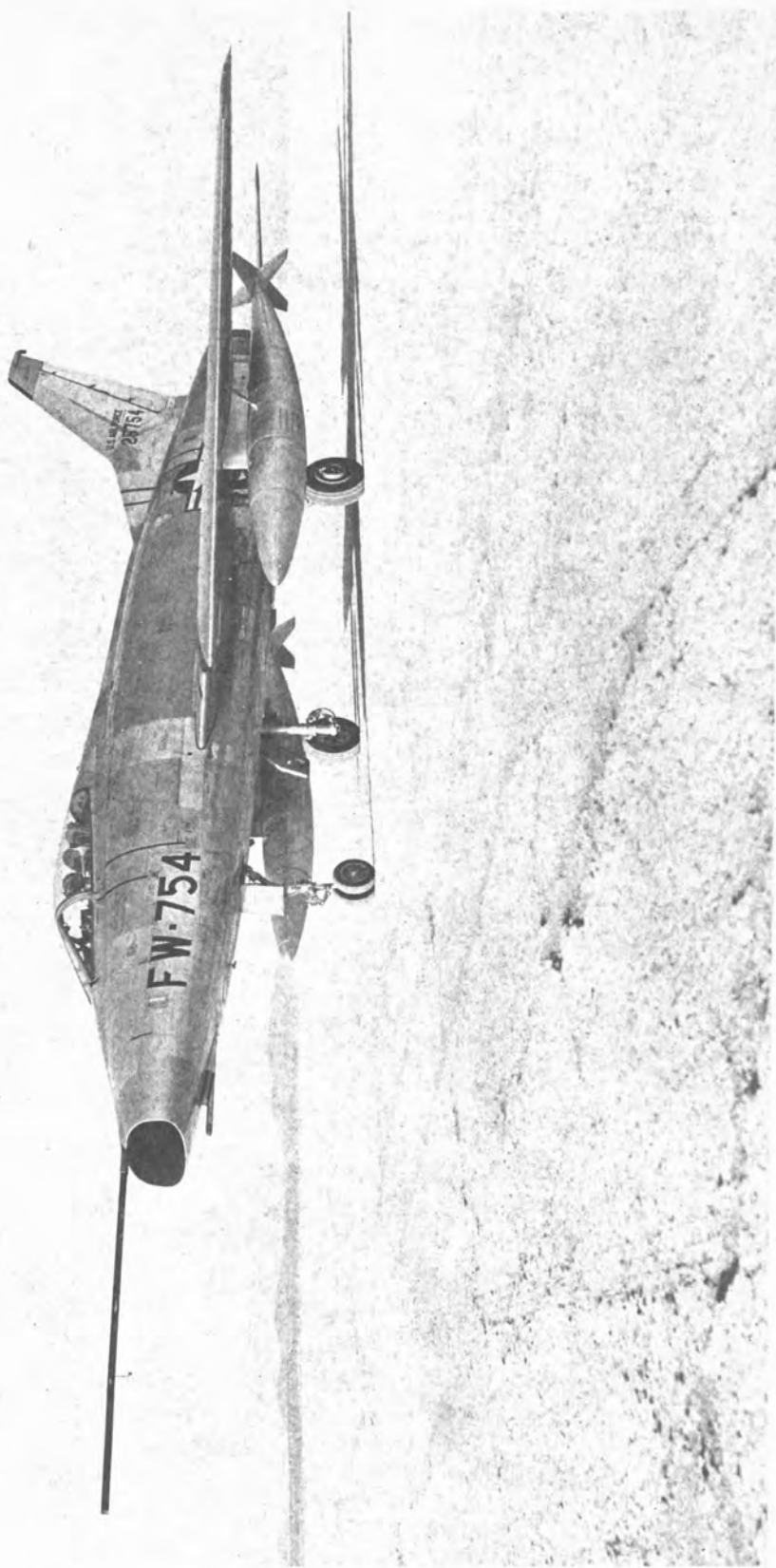
RIGHT REAR VIEW



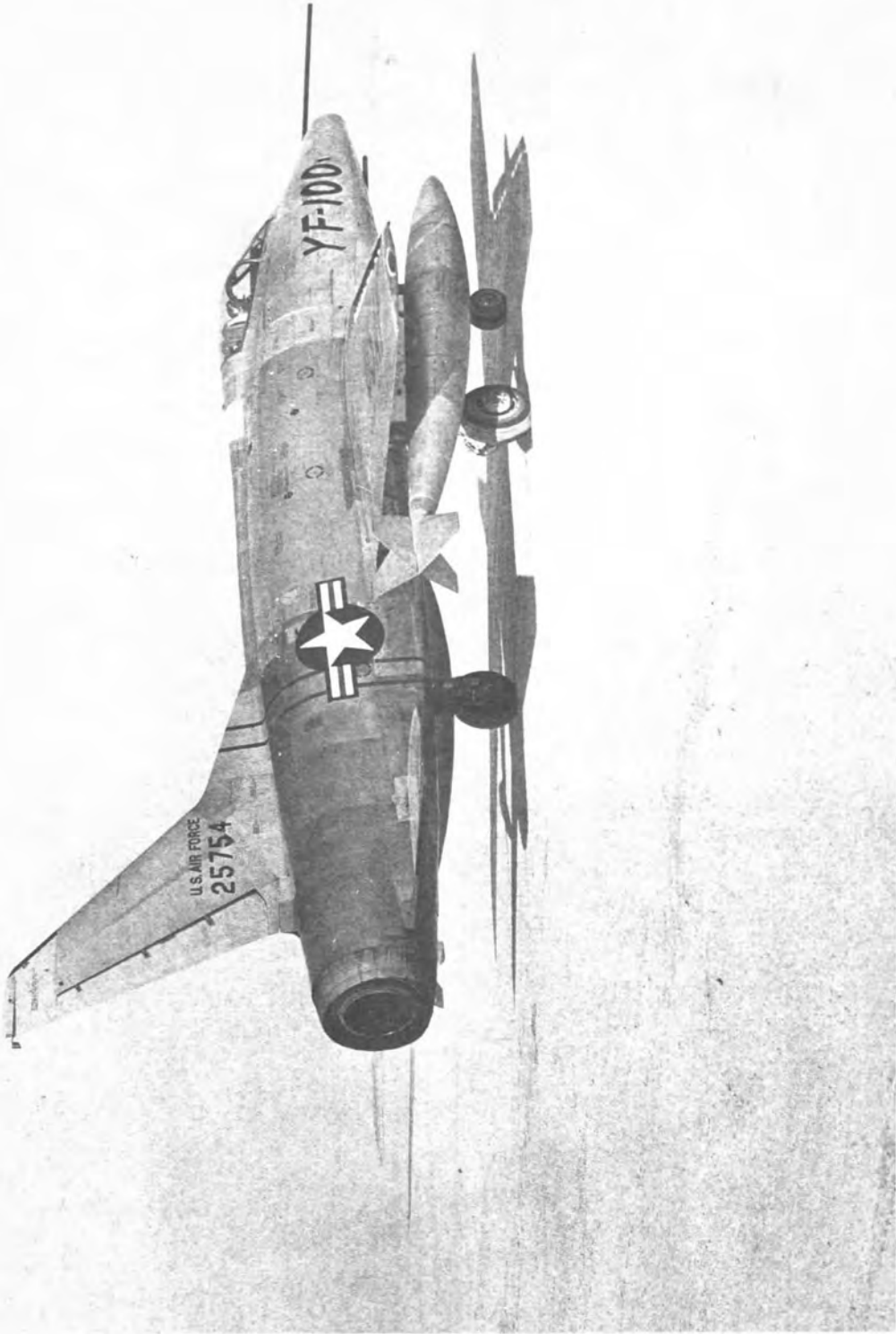
RIGHT SIDE VIEW



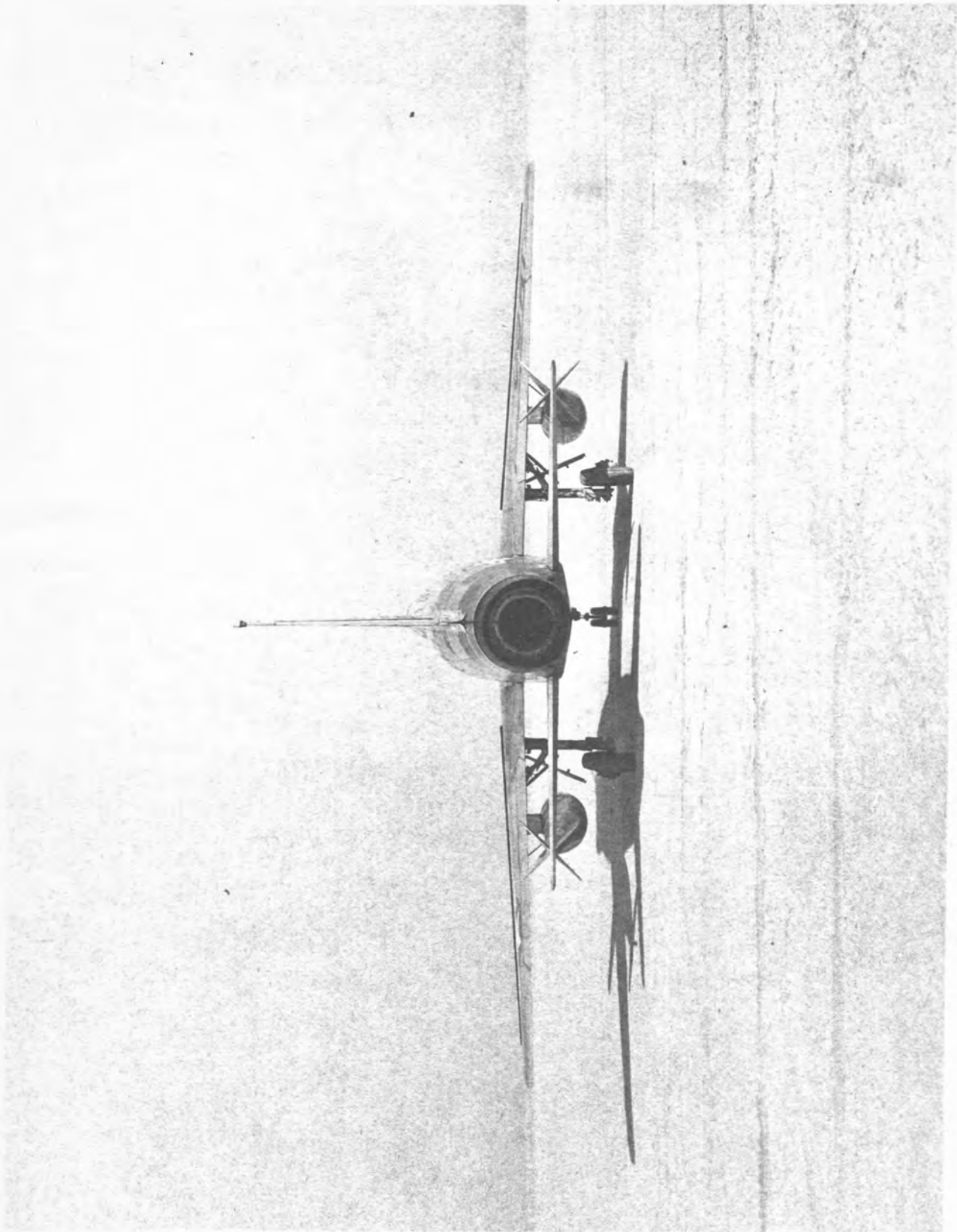
FRONT VIEW (Tanks on)



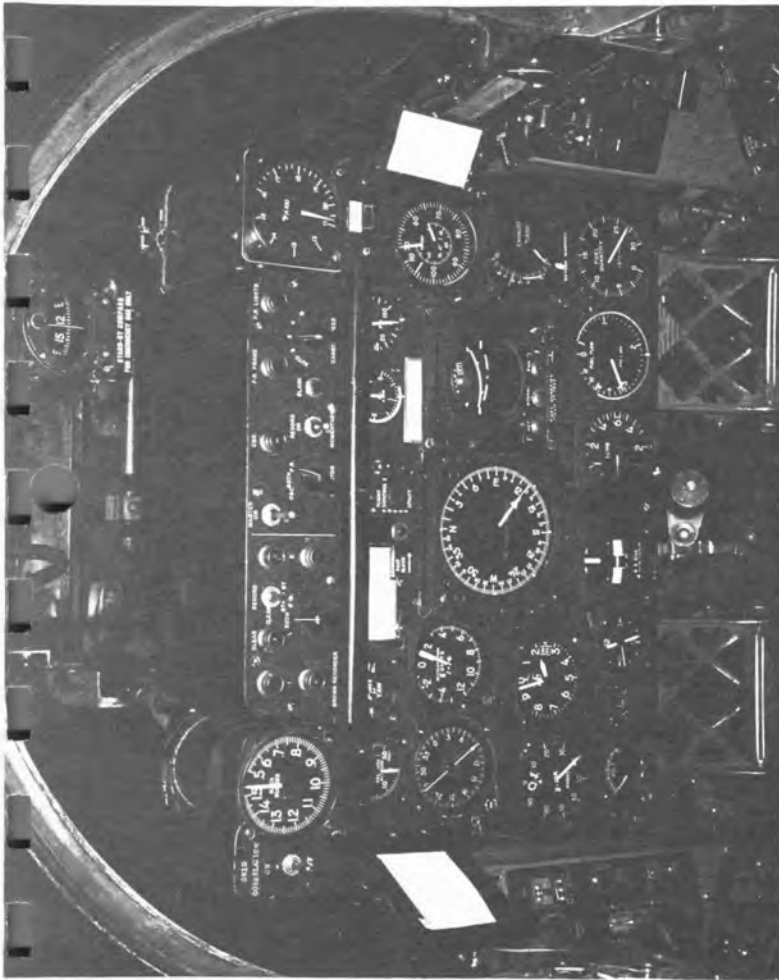
LEFT FRONT VIEW (Tanks on)



RIGHT REAR VIEW (Tanks on)



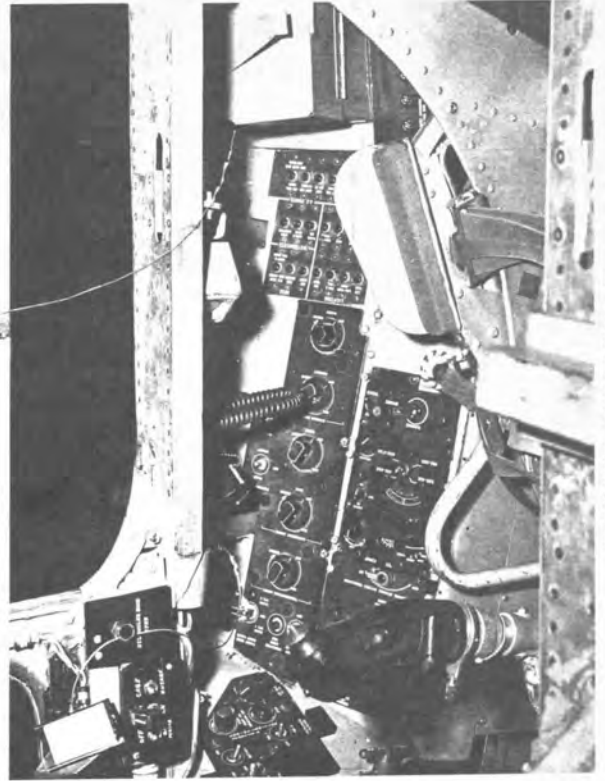
REAR VIEW (Tanks on)



INSTRUMENT PANEL

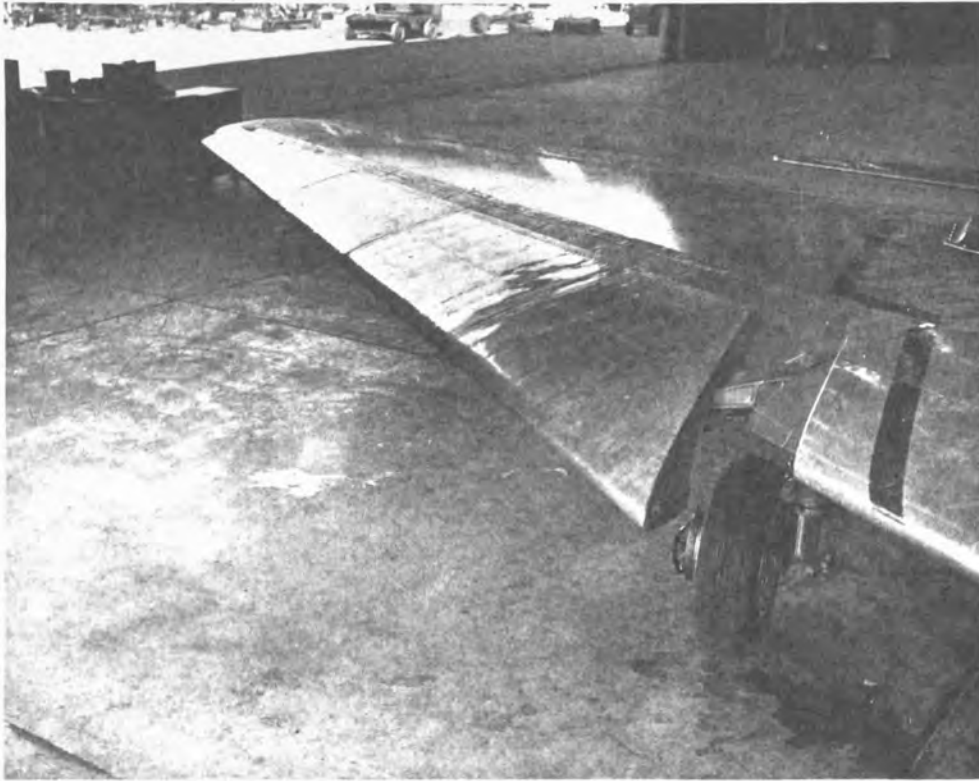


LEFT CONSOLE



RIGHT CONSOLE

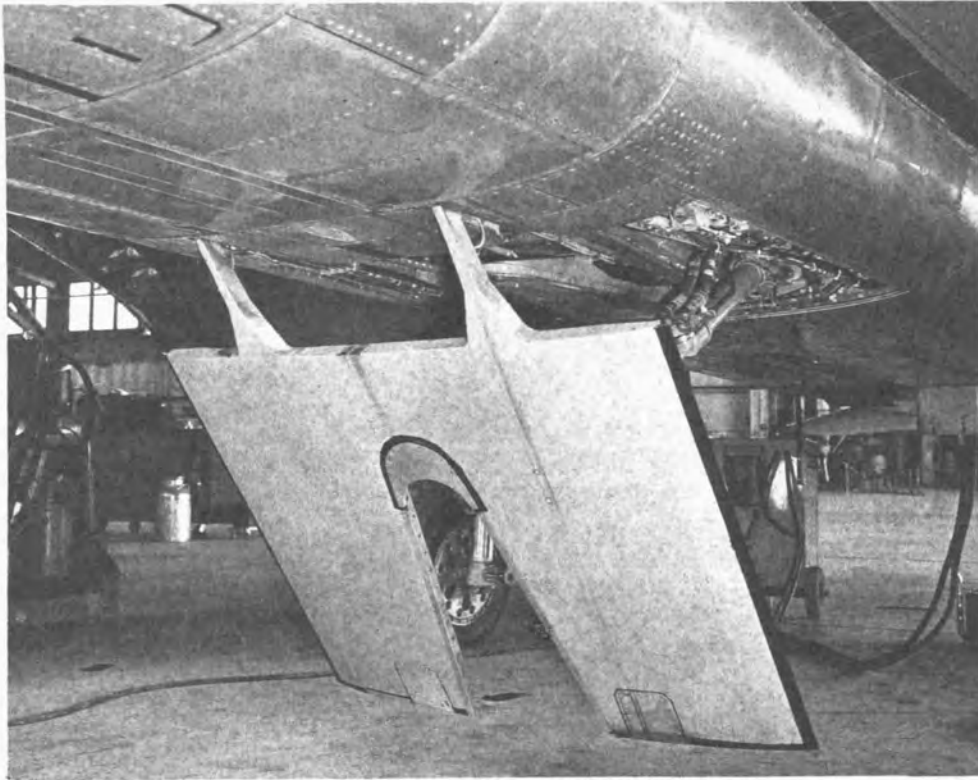
COCKPIT LAYOUT



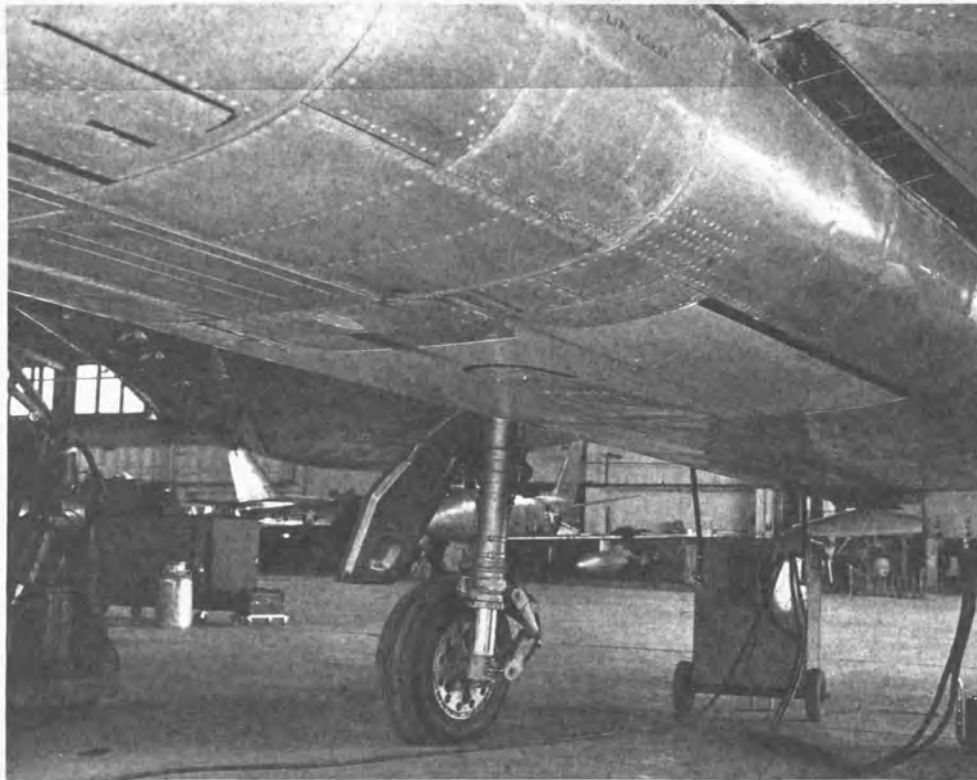
SLATS OPEN



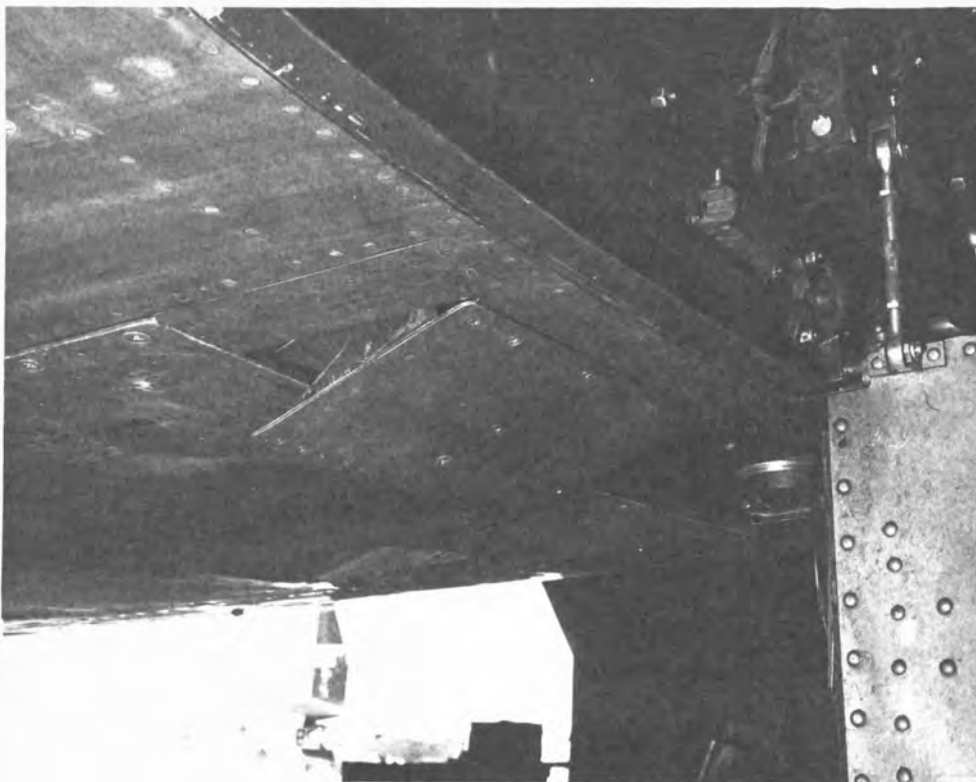
SLATS CLOSED



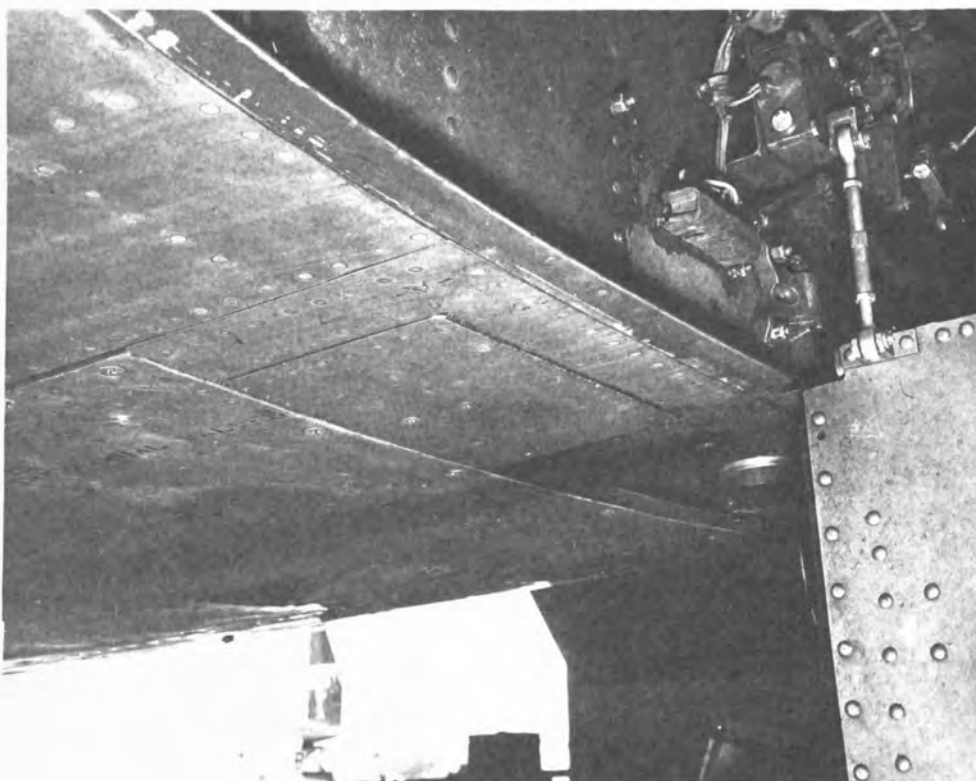
SPEED BRAKE OPEN



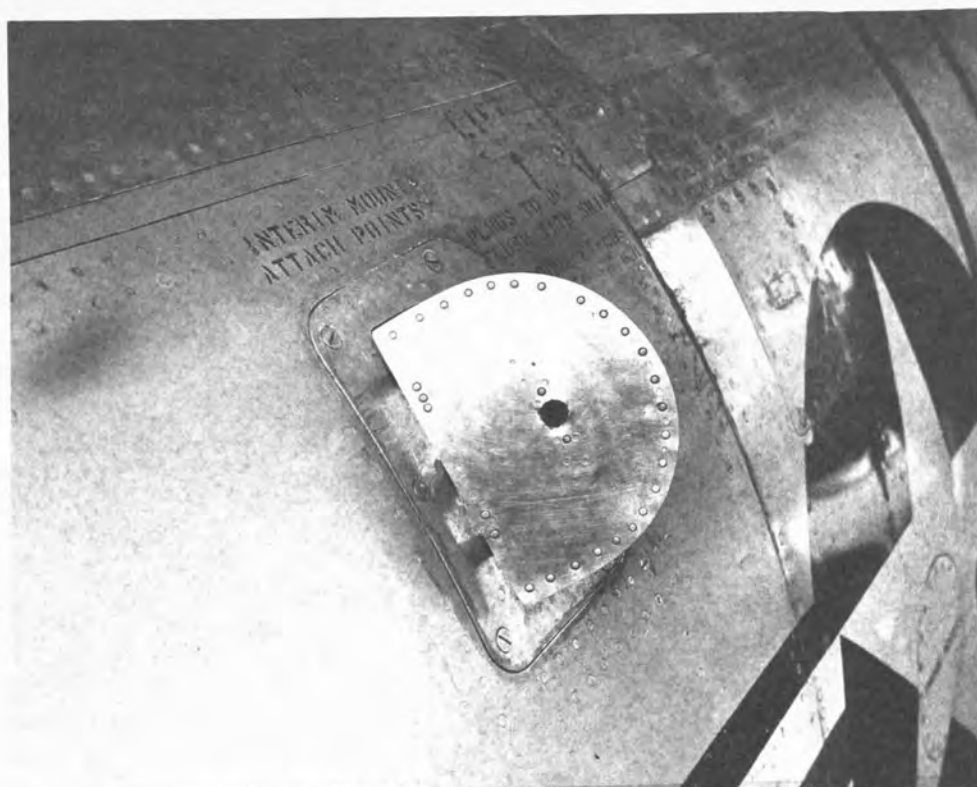
SPEED BRAKE CLOSED



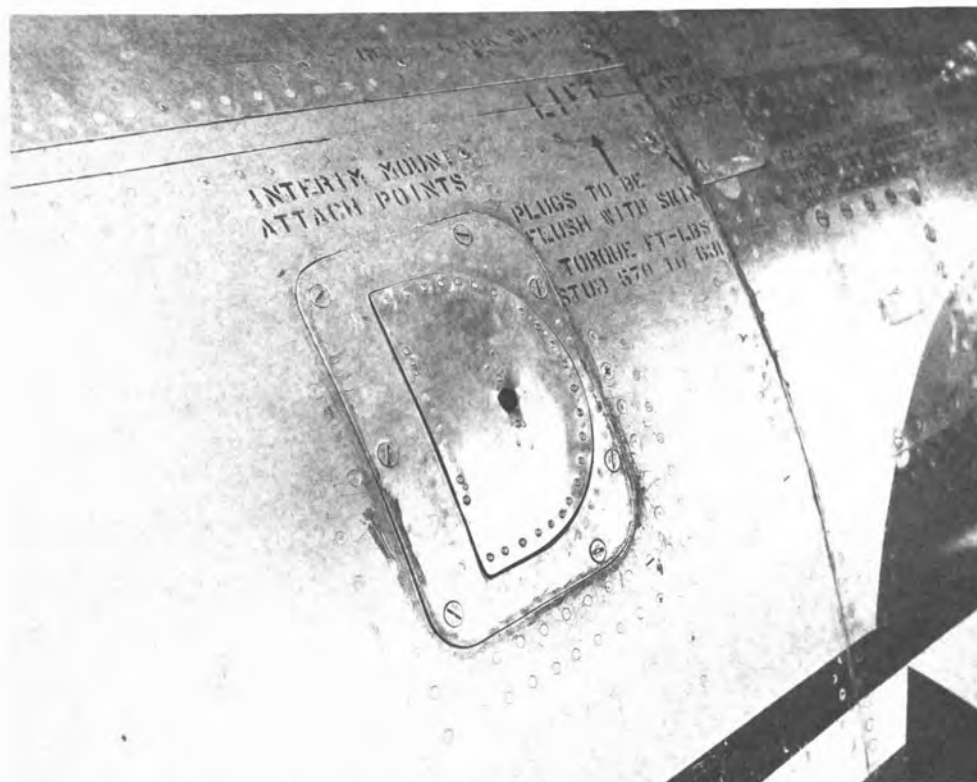
OIL COOLER DOOR OPEN



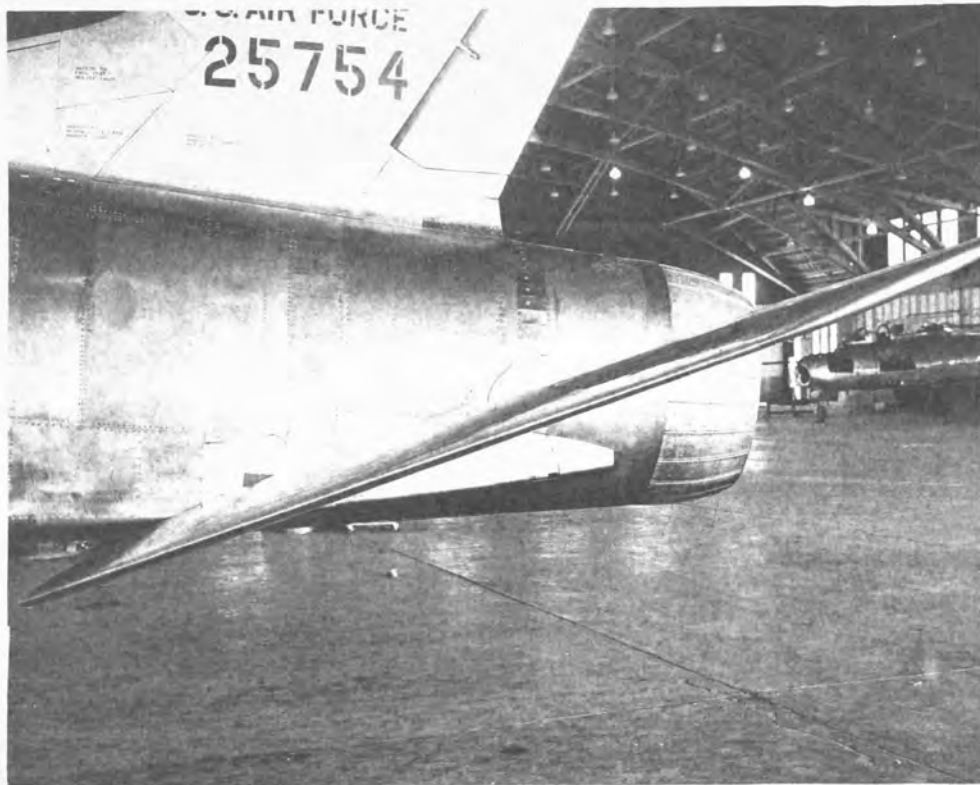
OIL COOLER DOOR CLOSED



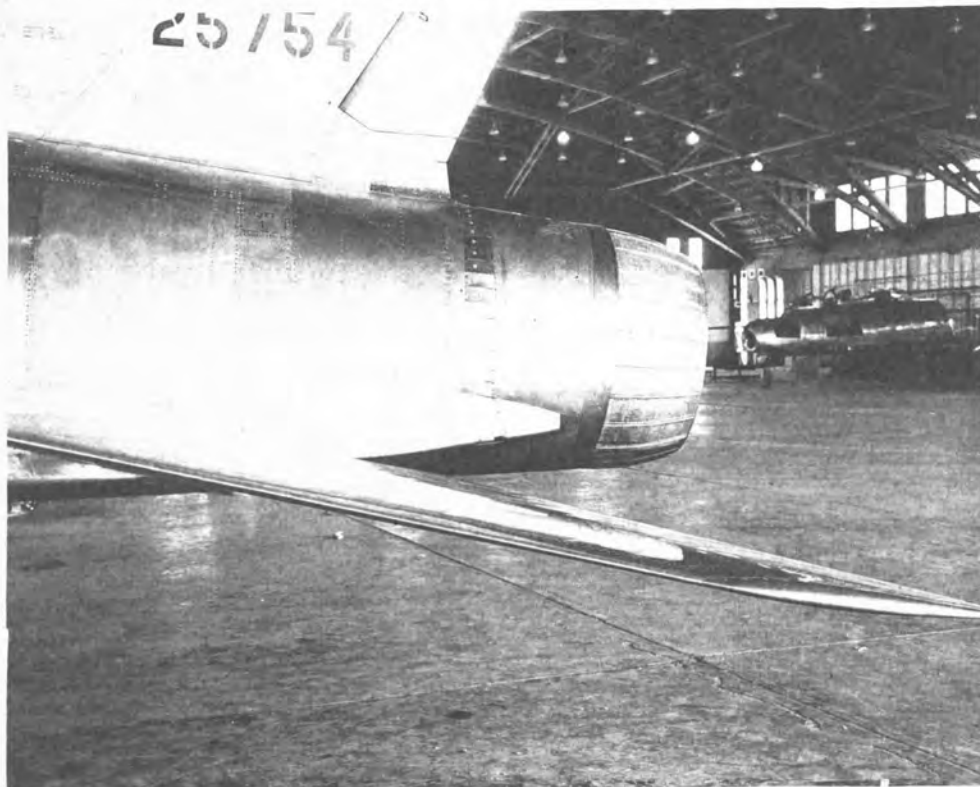
SURGE BLEED VALVE DOOR OPEN



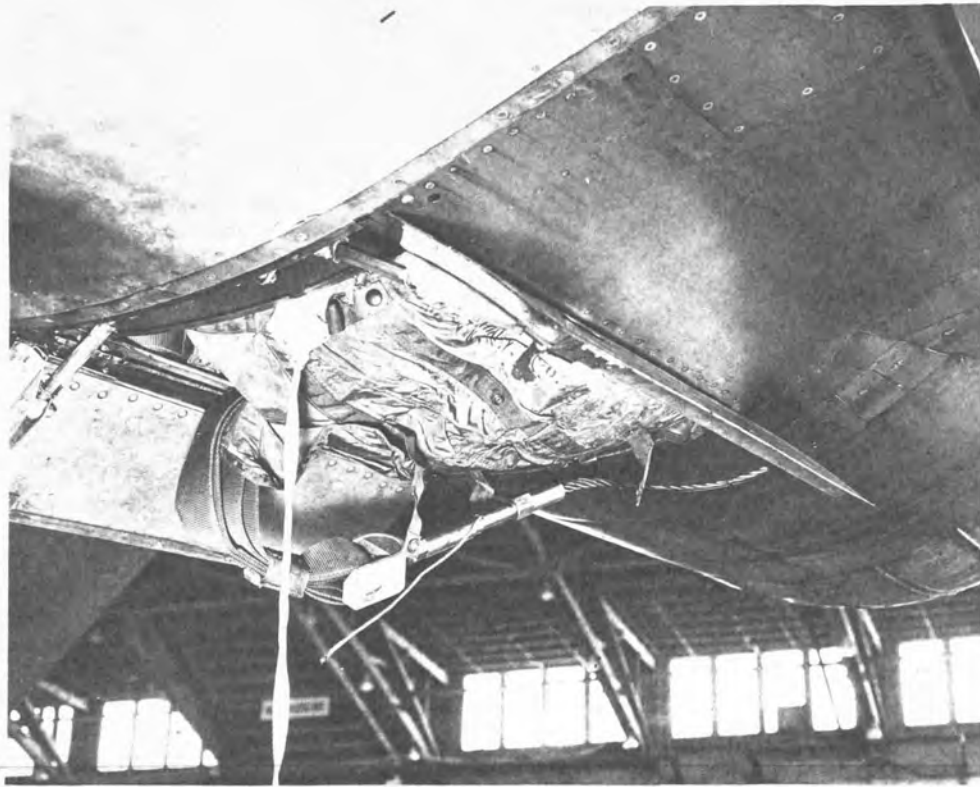
SURGE BLEED VALVE DOOR CLOSED



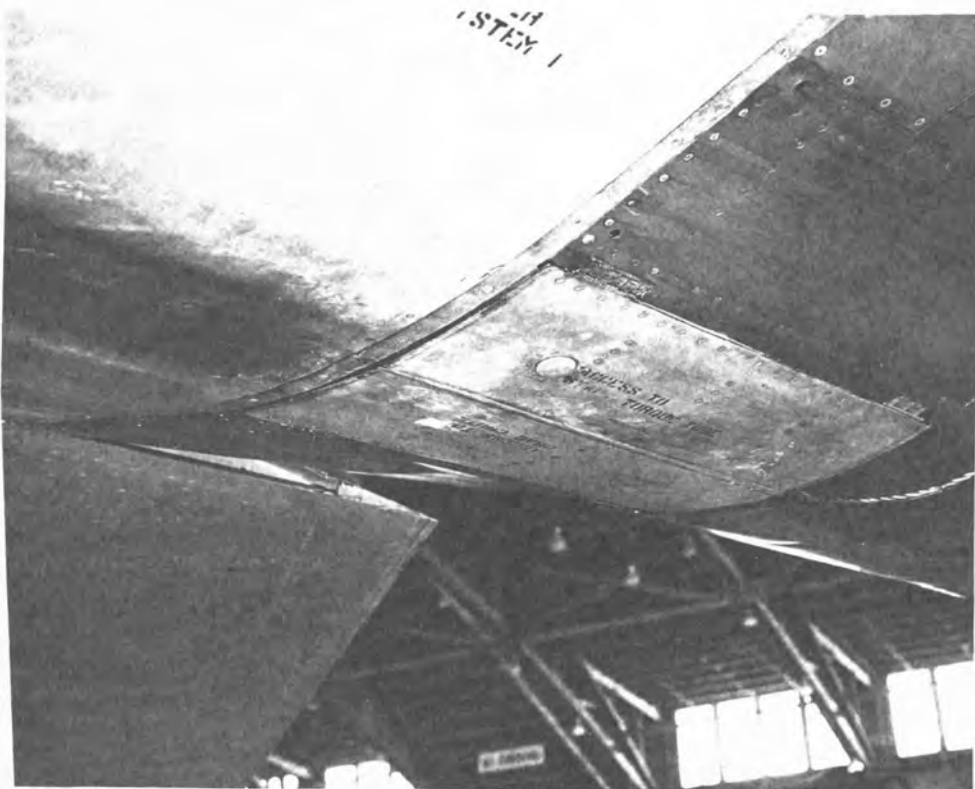
STABILIZER FULL NOSE UP



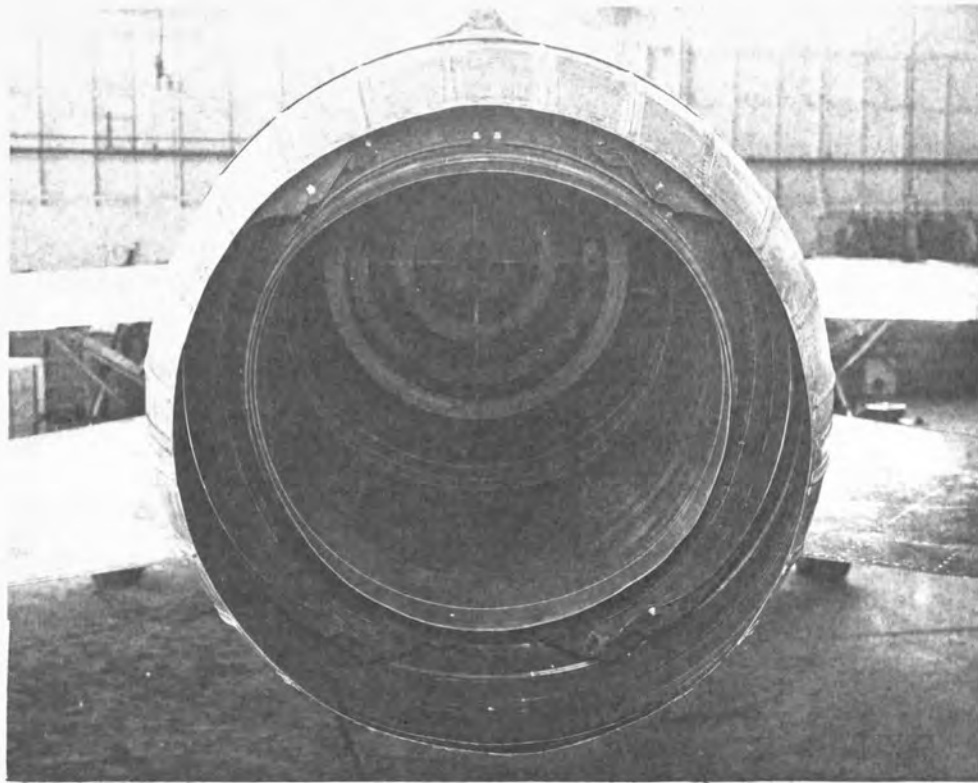
STABILIZER FULL NOSE DOWN



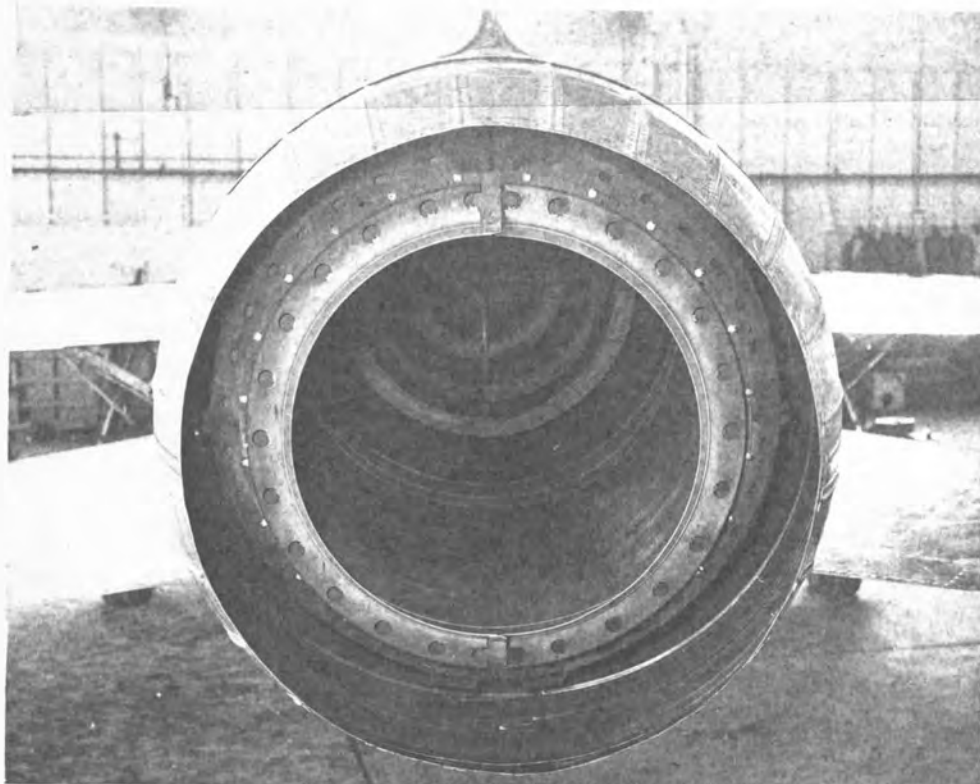
DRAG CHUTE DOOR OPEN



DRAG CHUTE DOOR CLOSED



EXHAUST NOZZLE OPEN



EXHAUST NOZZLE CLOSED

APPENDIX III

FLIGHT LOG AND ORIGINAL DATA
YF-100A, USAF No. 52-5754

	<u>Page No.</u>
Flight Log	2--4
Original Data (corrected for instrument error)	5--21
Take-Off Profiles	22--28
Landing Profiles	29--34

FLIGHT LOG

<u>Flight No.</u>	<u>Date</u>	<u>Flight Time</u>	<u>Pilot</u>	<u>Tests</u>
1	9-3-53	:58	Captain Hopkins	Military power check climb, Max. level flight speed at 45,000 feet.
2	9-3-53	:38	Captain Hopkins	Max. power check climb, airspeed calibration at 40,000 feet.
3	9-4-53	:28 ✓	Captain Hopkins	Performance T.O., airspeed calibration by tower fly-bys.
4	9-4-53	:47 ✓	Captain Hopkins	Performance and stability T.O., Dynamic longitudinal, lateral and directional stability in the power and power approach configuration at 10,000 feet, max. C_L landing.
5	9-4-53	:40	Captain Hopkins	Performance and stability T.O., max. power climb, speed-power at 41,500 feet, airspeed calibration.
6	9-4-53	:17	Captain Hopkins	Flight aborted due to engine malfunction.
7	9-10-53	:54	Lt. Col. Everest	Military power check climb, speed-power at 41,500 feet, Max. speed points at 35,000, Stalls at 35,000 feet.
8	9-10-53	:30	Lt. Col. Everest	Stability T.O., Max. power climb, Maneuvering flight at 45,000 feet.
9	9-11-53	:30	Lt. Col. Everest	Performance and stability T.O., Maneuvering flight at 10,000 ft, stalls at 10,000 feet.
10	9-11-53	:22	Lt. Col. Everest	Performance and stability T.O., Speed-power at 11,500 feet.
11	9-11-53	:27	Lt. Col. Everest	Speed-power at 25,000 feet.
12	9-11-53	:23	Lt. Col. Everest	Stability T.O., max. speed at 35,000 feet, max. "g" supersonic diving turn at 35,000 feet, Static thrust run.

<u>Flight No.</u>	<u>Date</u>	<u>Flight Time</u>	<u>Pilot</u>	<u>Tests</u>
13	9-12-53	:20	Lt. Col. Everest	Maximum speed at 2500 feet, longitudinal trim change with dive brake opening, stability and performance landing.
14	9-12-53	:54	Lt. Col. Everest	Performance T.O., longitudinal trim change, performance landing, performance T.O. (Mil. power), performance landing.
15	9-12-53	:40	Lt. Col. Everest	Performance T.O., Comparative performance with F-86-E.
16	9-12-53	:42	Lt. Col. Everest	Speed-power at 25,000 ft, Stalls at 35,000 ft.
17	9-12-53	:31	Lt. Col. Everest	Max. power check climb, Dynamic longitudinal, Lateral and direction stability at 45,000 feet. Stalls at 45,000 feet.
18	9-13-53	:40	Captain Hopkins	Max. power climbs. Static longitudinal stability at 35,000 feet.
19	9-13-53	:34	Major Yeager	Familiarization flight.
20	9-14-53	:25	Lt. Col. Everest	Performance T.O., max. power climb. Max. speed at 50,000 feet. Dive.
21	9-14-53	:31	Lt. Col. Everest	Max. power climb. Dive.
22	9-14-53	:24	Major Murray	Familiarization flight.
23	9-14-53	:29	Lt. Col. Everest	Max. power climb. Aileron rolls at 45,000 ft. Dive.
24	9-14-53	:27	Lt. Col. Everest	Max. power climb. Max. speed at 51,000 ft. Dive. Performance landing.

<u>Flight No.</u>	<u>Date</u>	<u>Flight Time</u>	<u>Pilot</u>	<u>Tests</u>
25	9-14-53	:04	Lt. Col. Everest	Night flight--gear failed to retract--flight aborted.
26	9-15-53	:40	Major Stephens	Familiarization flight.
27	9-15-53	:30	Col. Hanes	Familiarization flight.
28	9-15-53	:19	Lt. Col. Everest	Performance T.O. Side-slips at 10,000 ft. Performance landing.
29	9-15-53	:13	Lt. Col. Everest	Performance T.O. Aileron rolls at 10,000 ft. Performance landing.
30	9-15-53	:03	Lt. Col. Everest	Performance T.O. Performance landing.
31	9-15-53	:28	Lt. Col. Everest	Night flight evaluation. Level flight acceleration at 20,000 and 35,000 ft.
32	9-16-53	:29	Lt. Col. Everest	Performance T.O. Aileron rolls at 10,000 ft.
33	9-16-53	:27	Lt. Col. Everest	Static longitudinal stability at 10,000 feet.
34	9-16-53	:23	Lt. Col. Everest	Max. speed at 50,000 ft. Dive.
35	9-17-53	:45	Maj. Gen. Boyd	Evaluation flight.
36	9-17-53	:28	Lt. Col. Everest	Level flight acceleration at 20,000 and 35,000 feet. Static longitudinal stability at 10,000 feet.
37	9-17-53	:20	Lt. Col. Everest	Evaluation of external tank installation.
38	9-17-53	:25	Captain Hopkins	Speed-power at 25,000 ft. with external tanks.
39	9-17-53	:57	Captain Hopkins	Speed-power at 25,000 ft. with external tanks.
		<u>TOTAL 19:42</u>		

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	STATIC THRUST RUN								
Flight No.	-	-	-	-	-	-	-	-	-
Run No.	1*	2	3	4	5	6	7	8	9
Altitude - Ft	-	-	-	-	-	-	-	-	-
IAS - Knots	-	-	-	-	-	-	-	-	-
OAT - °C	21	21	20.6	20.6	20.6	20	20	20	20
RPM (high press)	9480	9480	9405	9215	9025	8600	8270	7630	6825
RPM (low press)	5770	5690	5360	5320	5045	4365	3900	3120	2530
Exhaust Gas Temp - °C	562	552	542	462	430	405	367	275	265
Fuel Used (Eng) - Gal	-	-	-	-	-	-	-	-	-
Fuel Used (A/B) - Gal	-	-	-	-	-	-	-	-	-
Fuel Flow (total) - GPH	3158	818	776	670	552	377	280	187	143.4
Oil Cooler GAP - $\frac{PSI}{G}$	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48
Nozzle Position	0	C	C	C	C	C	C	C	C
Time - Sec	-	-	-	-	-	-	-	-	-
GROSS THRUST - LBS	9095	5788	5408	4685	3937	2507	1917	1250	758
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597

* NB ON

Test	STATIC THRUST RUN								
Flight No.	-	-	-	-	-	-	-	-	-
Run No.	10	11	12	13	14	15	16	17	18
Altitude - Ft	-	-	-	-	-	-	-	-	-
IAS - Knots	-	-	-	-	-	-	-	-	-
OAT - °C	20	20	20	20	20	20	20	20	20
RPM (high press)	6050	5680	5475	6020	6845	7610	8230	8620	9020
RPM (low press)	5175	1980	1975	2170	2695	3280	3870	4410	5020
Exhaust Gas Temp - °C	265	265	250	265	265	278	355	425	430
Fuel Used (Eng) - Gal	-	-	-	-	-	-	-	-	-
Fuel Used (A/B) - Gal	-	-	-	-	-	-	-	-	-
Fuel Flow (total) - GPH	1249	1095	110.1	1505	1538	202	2945	395	558
Oil Cooler GAP - $\frac{PSI}{G}$	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4	9.4
Bleed Valve Position	C	C	C	C	C	C	0	0	C
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48
Nozzle Position	C	C	0	C	C	C	C	C	C
Time - Sec	-	-	-	-	-	-	-	-	-
GROSS THRUST - LBS	465	382	220	475	783	1240	1848	2540	3857
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597	27.597

Test	STATIC THRUST RUN								
Flight No.	-	-	-	-	-	-	-	-	-
Run No.	19	20	21	22*	23*	24	25	26	27
Altitude - Ft	-	-	-	-	-	-	-	-	-
IAS - Knots	-	-	-	-	-	-	-	-	-
OAT - °C	20	20	20.3	20.3	20.3	-	-	-	-
RPM (high press)	9240	9420	9510	9540	9020	-	-	-	-
RPM (low press)	5290	5430	5260	5740	4960	-	-	-	-
Exhaust Gas Temp - °C	500	540	558	558	448	-	-	-	-
Fuel Used (Eng) - Gal	-	-	-	-	-	-	-	-	-
Fuel Used (A/B) - Gal	-	-	-	-	-	-	-	-	-
Fuel Flow (total) - GPH	663	740	808.5	3045	2350	-	-	-	-
Oil Cooler GAP - $\frac{PSI}{G}$	9.4	9.4	9.4	9.4	9.4	-	-	-	-
Bleed Valve Position	C	C	C	C	C	-	-	-	-
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	-	-	-	-
Nozzle Position	C	C	C	0	0	-	-	-	-
Time - Sec	-	-	-	-	-	-	-	-	-
GROSS THRUST - LBS	4630	5265	5700	8745	6035	-	-	-	-
BARO. PRESS - IN Hg	27.597	27.597	27.597	27.597	27.597	-	-	-	-

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	AIRSPEED CALIBRATION (TOWER FLY-BY)							
Flight No.	3	3	3	3	3	3	3	3
Run No.	1	2	3	4	5	6	7	8
Altitude - Ft	1950	2105	2200	2260	2300	1850	2140	2150
IAS - Knots	600	495	412.5	301.5	250	600	565	452
OAT - °C								
RPM (high press)								
RPM (low press)								
Exhaust Gas Temp - °C								
Fuel Used (Eng) - Gal								
Fuel Used (A/B) - Gal								
Fuel Flow (total) - GPH								
Oil Cooler GAP - in.								
Bleed Valve Position								
Fuel Weight - lbs/gal								
Nozzle Position								
Time - Sec								
PRESSURE ALT-TRUE - Ft	2507	2409	2527	2461	2455	2448	2540	2510
AVG GROSS WT LBS	← 23000 LBS →							
GEAR POS	UP	UP	UP	UP	UP	UP	UP	UP

Test	AIRSPEED CALIBRATION (AIRPLANE FLY-BY)				
Flight No.	2	2	2	2	2
Run No.	1	2	3	4	5
Altitude - Ft	38765	39160	39355	39425	39415
IAS - Knots	304.5	296	274.5	262.5	192.5
OAT - °C	-	-15	-19	-22	-35.5
RPM (high press)					
RPM (low press)					
Exhaust Gas Temp - °C					
Fuel Used (Eng) - Gal					
Fuel Used (A/B) - Gal					
Fuel Flow (total) - GPH					
Oil Cooler GAP - in.					
Bleed Valve Position					
Fuel Weight - lbs/gal					
Nozzle Position					
Plan - 800	← 23000 LBS →				
PACER CAS MPH	261.5	249	261.5	264.5	226.5
PACER ALT FT	39865	39865	39870	39860	39705

Test	AIRSPEED CALIBRATION (PACER)	
Flight No.	2	5
Run No.	5	1
Altitude - Ft	39415	227
IAS - Knots	192.5	423.60
OAT - °C	-35.5	-37
RPM (high press)		
RPM (low press)		
Exhaust Gas Temp - °C		
Fuel Used (Eng) - Gal		
Fuel Used (A/B) - Gal		
Fuel Flow (total) - GPH		
Oil Cooler GAP - in.		
Bleed Valve Position		
Fuel Weight - lbs/gal		
Nozzle Position		
Time - Sec	← 23000 LBS →	
PACER CAS MPH	226.5	265.5
PACER ALT FT	39705	42555

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	LEVEL FLIGHT SPEED POWER (CLEAN)									
	13	13		10	10	10	10	14	14	
Flight No.	13	13		10	10	10	10	14	14	
Run No.	2	3		1	2	3	4	1	2	
Altitude - Ft	2545	2545		9775	12265	13460	14190	10220	11000	
IAS - Knots	596.5	470		533	492	463.5	418	444.5	392.5	
OAT - °C	83	64		66	54.5	47	38	52	41.5	
RPM (high press)	9545	9550		9530	8940	9510	9255	9440	9190	
RPM (low press)	5350	5500		5440	4815	5615	5365	5485	5270	
Exhaust Gas Temp - °C	455	475		500	410	510	450	475	425	
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH	3560	829		3273	2170	764	578	696	561	
Oil Cooler GAP - in.	9.6	9.6		9.5	0.25	9.45	0.2	9.4	9.5	
Bleed Valve Position	C	C		C	C	C	C	C	C	
Fuel Weight - lbs/gal	6.47	6.47		6.55	6.55	6.55	6.55	6.48	6.48	
Nozzle Position	0	C		0	0	C	C	C	C	
Time - Sec										
Gross Weight ~ lbs.	23050	22380		22810	21630	21180	20940	23840	23660	

Test	LEVEL FLIGHT SPEED POWER (CLEAN)									
	14	14	14	14	14	14	14	14	14	
Flight No.	14	14	14	14	14	14	14	14	14	
Run No.	3	4	5	6	7	8	9	10		
Altitude - Ft	11600	12190	12490	12220	12060	12000	12000	12200		
IAS - Knots	359	309	272	237	192.5	160	141.5	95.6		
OAT - °C	32.5	26.5	22.0	18.5	15	—	13	—		
RPM (high press)	9100	8900	8820	8815	8760	9020	9060	9010		
RPM (low press)	5235	5060	5005	5005	4950	—	5285	—		
Exhaust Gas Temp - °C	410	375	375	375	375	—	450	—		
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH	514	423	397	370.5	368	—	493.5	—		
Oil Cooler GAP - in.	0.7	1.3	1.8	3.15	5.2	—	5.35	—		
Bleed Valve Position	C	0	0	0	0	C	C	C		
Fuel Weight - lbs/gal	6.48	6.48	6.48	6.48	6.48	6.48	6.48	6.48		
Nozzle Position	C	C	C	C	C	C	C	C		
Time - Sec										
Gross Weight ~ lbs.	23560	23430	23300	23200	23040	22900	22780	22600		

Test	LEVEL FLIGHT SPEED POWER (CLEAN)									
	11	11	11	11	11	11	16	16	16	
Flight No.	11	11	11	11	11	11	16	16	16	
Run No.	1	2	3	4	5	6	1	3	4	
Altitude - Ft	26340	27120	27700	28280	28680	28780	27150	27890	27910	
IAS - Knots	403.5	390.5	381	370	347	318	289	228	185.5	
OAT - °C	22.5	20	—	15.5	10.5	4.1	-0.5	—	-18.5	
RPM (high press)	9510	9140	8700	9475	9040	9025	8750	8750	8980	
RPM (low press)	5770	5335	—	5840	5375	5365	—	—	5540	
Exhaust Gas Temp - °C	545	475	—	535	460	450	425	—	495	
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH	2438	1946	—	584	373.3	370.5	320.8	—	387	
Oil Cooler GAP - in.	9.7	0.6	—	1.65	1.6	3.9	2.39	—	5.3	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal	6.47	6.47	6.47	6.47	6.47	6.47	6.43	6.43	6.43	
Nozzle Position	0	0	—	C	C	C	C	C	C	
Time - Sec										
Gross Weight ~ lbs.	23380	21930	21670	21420	21270	21140	23390	23100	23010	

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	LEVEL FLIGHT SPEED POWER (CLEAN)							
Flight No.	16	16	7	7	12	12	22	
Run No.	5	6	2	3	1	2	1	
Altitude - Ft	27740	27940	34700	34920	34130	34850	33710	
IAS - Knots	155	292.5	335	314	341.5	317.5	344.5	
OAT - °C	-21.5	-3.0	-2.0	-7.5	0.5	-6.0	0	
RPM (high press)	8990	8730	9215	9120	9330	9200	9260	
RPM (low press)	5565	5765	5655	5710	5735	5735	5670	
Exhaust Gas Temp - °C	495	415	525	500	530	510	525	
Fuel Used (Eng) - Gal								
Fuel Used (A/B) - Gal								
Fuel Flow (total) - GPH	364.0	319	180.5	424.8	197.0	398	185.6	
Oil Cooler GAP - in.	5.3	3.4	0.2	0.6	1.4	2.2	0	
Bleed Valve Position	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal	6.43	6.43	6.425	6.425	6.48	6.48	6.42	
Nozzle Position	C	C	0	C	0	C	0	
Time - Sec								
Gross Weight ~ lbs.	22860	22450	22420	22200	22180	21900	22050	

Test	LEVEL FLIGHT SPEED POWER (CLEAN)							
Flight No.	5	5	5	5	7		1	1
Run No.	1	2	3	4	1		1	2
Altitude - Ft	40970	42330	42770	43160	40870		44530	44630
IAS - Knots	290.5	273.5	245.5	226.5	203.5		265.5	244
OAT - °C	-15.5	-21	-28.5	-35	-37		-22	-
RPM (high press)	9135	8590	8970	8850	8890		9030	8955
RPM (low press)	5090	5065	5730	5655	5700		5660	-
Exhaust Gas Temp - °C	545	450	520	500	510		530	-
Fuel Used (Eng) - Gal								
Fuel Used (A/B) - Gal								
Fuel Flow (total) - GPH	1380	1075	302	278.6	290		1320	-
Oil Cooler GAP - in.	9.7	1.65	2.6	5.3	5.3		5.3	-
Bleed Valve Position	C	C	C	C	C		C	C
Fuel Weight - lbs/gal	6.38	6.38	6.38	6.38	6.425		6.38	6.38
Nozzle Position	0	0	C	C	C		0	C
Time - Sec								
Gross Weight ~ lbs.	22270	21880	21630	21340	22920		21950	21600

Test	LEVEL FLIGHT SPEED POWER (CLEAN)							
Flight No.	20							
Run No.	1							
Altitude - Ft	49550							
IAS - Knots	291.5							
OAT - °C	-28							
RPM (high press)	8970							
RPM (low press)	5695							
Exhaust Gas Temp - °C	570							
Fuel Used (Eng) - Gal								
Fuel Used (A/B) - Gal								
Fuel Flow (total) - GPH	1032							
Oil Cooler GAP - in.	5.3							
Bleed Valve Position	C							
Fuel Weight - lbs/gal	6.465							
Nozzle Position	0							
Time - Sec								
Gross Weight ~ lbs.	22060							

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	LEVEL FLIGHT SPEED POWER (TANKS)									
Flight No.	38	38	38	38	39	39	39	39	39	39
Run No.	1	2	3	4	1	2	3	4	5	
Altitude - Ft	26180	27050	27700	27840	25900	26510	26790	27290	27645	
IAS - Knots	394	385	368.5	356	329	296	250	205	179	
OAT - °C	16	14	10	6.5	2	-3	-10	-18	-22	
RPM (high press)	9465	9260	8750	9300	9040	8885	8785	8745	8845	
RPM (low press)	5725	5530	5130	5725	5310	5275	5280	5200	5425	
Exhaust Gas Temp - °C	530	500	400	500	455	425	425	430	450	
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH	2390	2078	1471	5645	454	382.9	350.5	328.9	361	
Oil Cooler GAP - in.	9.4	9.4	0.9	0.25	0.1	0.1	0.6	1.22	2.4	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal	6.47	6.47	6.47	6.47	6.38	6.38	6.38	6.38	6.38	
Nozzle Position	0	0	0	C	C	C	C	C	C	
Time - Sec										
Gross Weight ~ lbs	22900	22200	21730	21480	23460	22960	22580	22170	22050	

Test										
Flight No.										
Run No.										
Altitude - Ft										
IAS - Knots										
OAT - °C										
RPM (high press)										
RPM (low press)										
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.										
Bleed Valve Position										
Fuel Weight - lbs/gal										
Nozzle Position										
Time - Sec										

Test										
Flight No.										
Run No.										
Altitude - Ft										
IAS - Knots										
OAT - °C										
RPM (high press)										
RPM (low press)										
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.										
Bleed Valve Position										
Fuel Weight - lbs/gal										
Nozzle Position										
Time - Sec										

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	3960	4925	5990	6980	8030	8965	9970	10960	11960
IAS - Knots	469.0	481.0	479.0	469.0	460.5	454.0	459.5	453.5	446.5
OAT - °C	65.5	65.5	63.9	60.0	56.0	53.5	51.5	50.0	49.5
RPM (high press)	9530	9530	9540	9520	9530	9530	9530	9525	9530
RPM (low press)	5965	5960	5980	5985	5505	5530	5540	5550	5560
Exhaust Gas Temp - °C	490	490	495	515	520	520	525	525	530
Fuel Used (Eng) - Gal	37.6	41.6	43.5	45.2	46.6	48.0	49.7	51.2	52.6
Fuel Used (A/B) - Gal	99.4	112.2	119.4	124.0	128.4	132.4	136.0	143.0	147.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2	0.2
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	0	17.9	26.7	34.2	41.0	47.1	55.0	62.7	69.5

Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	12940	12995	15020	15990	16960	18010	18980	19945	20945
IAS - Knots	444.5	435.5	427.0	419.0	416.0	413.0	407.0	416.0	399.5
OAT - °C	47.0	44.0	44.0	41.5	38.5	36.0	34.0	32.5	30.0
RPM (high press)	9530	9540	9545	9540	9540	9540	9535	9530	9530
RPM (low press)	5570	560.0	561.5	563.0	564.5	566.0	567.0	568.0	579.0
Exhaust Gas Temp - °C	535	540	545	545	545	555	560	560	555
Fuel Used (Eng) - Gal	53.8	54.9	56.1	57.1	58.3	59.6	60.6	61.9	62.8
Fuel Used (A/B) - Gal	151.0	154.0	157.4	160.4	164.0	168.0	171.0	174.4	177.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.2	0.2	0.2	0.2	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	79.7	80.2	85.5	90.6	96.8	103.2	109.0	115.0	120.5

Test	CHECK CLIMB ~ MAX. POWER								
Flight No.	2	2	2	2	2	2	2	2	2
Run No.									
Altitude - Ft	21970	22950	24015	24990	25985	26930	27880	28940	29975
IAS - Knots	399.5	391.0	383.0	375.5	370.5	368.0	362.5	356.5	351.0
OAT - °C	26.5	27.5	21.5	18.0	16.0	13.0	10.5	7.0	4.0
RPM (high press)	9520	9520	9520	9520	9520	9520	9520	9525	9520
RPM (low press)	5720	5750	5780	5795	5810	5840	5860	5890	5910
Exhaust Gas Temp - °C	550	545	545	545	560	560	560	565	565
Fuel Used (Eng) - Gal	64.0	65.1	66.1	67.1	68.1	69.2	70.0	71.0	72.0
Fuel Used (A/B) - Gal	181.0	189.0	187.0	189.0	192.4	195.4	198.0	200.4	203
Fuel Flow (total) - GPH									
Oil Cooler GAP - in °C	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	126.7	132.7	138.0	143.3	149.4	155.2	160.2	165.5	171.2

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX. POWER									
Flight No.	2	2	2	2	2	2	2	2	2	2
Run No.										
Altitude - Ft	30825	31800	32750	33695	34710	35675	36635	37560	38480	
IAS - Knots	343.0	335.5	323.0	329.0	316.5	308.0	303.5	299.5	293.0	
OAT - °C	1.0	-2.0	-4.0	-6.0	-9.0	-10.5	-13.5	-14.0	-14.0	
RPM (high press)	9510	9510	9445	9420	9405	9370	9330	9320	9295	
RPM (low press)	5920	5910	5890	5900	5900	5875	5865	5860	5840	
Exhaust Gas Temp - °C	565	565	565	565	565	565	565	560	560	
Fuel Used (Eng) - Gal	73.0	73.6	74.6	75.5	76.3	77.1	78.0	79.2	80.0	
Fuel Used (A/B) - Gal	205.0	208.0	210.4	212.8	215.2	217.4	220.0	224.0	226.4	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in. °C	0.1	0.1	0.1	0.1	0.35	0.35	0.45	1.4	1.5	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	176.0	181.6	187.7	193.9	199.8	205.2	212.2	222.3	230.0	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	2	2	2			20	20	20	20	
Run No.										
Altitude - Ft	39470	40435	41405			3090	3925	4965	5895	
IAS - Knots	279.5	272.5	262.5			4845	4890	483.0	475.5	
OAT - °C	-17.0	-20.0	-22.5			66	68	64.5	62	
RPM (high press)	9295	9270	9265			9510	9510	9510	9505	
RPM (low press)	5860	5870	5930			5990	5995	5965	5985	
Exhaust Gas Temp - °C	560	565	565			490	495	495	510	
Fuel Used (Eng) - Gal	80.8	81.4	82.4			743	76.3	77.8	79.0	
Fuel Used (A/B) - Gal	229.0	230.4	231.0			93.0	99.2	104.2	108.0	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in. °C	2.3	2.9	3.5			0.1	0.2	0.2	0.2	
Bleed Valve Position	C	C	C			C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0			0	0	0	0	
Time - Sec	236.5	242.5	252.0			0	13.5	20.8	26	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	20	20	20	20	20	20	20	20	20	
Run No.										
Altitude - Ft	6950	7925	8930	9930	10915	11935	12965	13950	14915	
IAS - Knots	461.0	453.0	452.0	451.5	447.5	443.0	439.5	434.0	430.0	
OAT - °C	58	56	53	51.5	49	46	43	40	37.5	
RPM (high press)	9500	9500	9495	9490	9485	9480	9470	9460	9420	
RPM (low press)	5955	5985	5985	5915	5920	5920	5940	5965	5980	
Exhaust Gas Temp - °C	515	515	520	520	520	520	525	527	535	
Fuel Used (Eng) - Gal	80.3	81.6	83.4	85.0	86.6	88.1	89.7	91.0	92.5	
Fuel Used (A/B) - Gal	112.0	116.4	121.6	127.0	131.8	136.6	141.2	145.2	149.2	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	32	38.5	47	55.1	62.9	70.5	78.5	85	91	

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	20	20	20	20	20	20	20	20	20	20
Run No.										
Altitude - Ft	15920	16980	17970	19015	20055	20990	22010	23015	24030	
IAS - Knots	426.0	421.0	417.0	413.0	403.0	401.0	394.0	385.5	378.0	
OAT - °C	29	31	29	28.8	26	25	23	20	17	
RPM (high press)	9500	9480	9480	9490	9480	9470	9480	9470	9470	
RPM (low press)	5608	5635	5650	5665	5685	5690	5710	5725	5755	
Exhaust Gas Temp - °C	540	540	540	543	543	543	543	540	543	
Fuel Used (Eng) - Gal	93.5	94.7	95.8	97.0	98.1	99.1	100.2	101.2	102.1	
Fuel Used (A/B) - Gal	152.6	156.2	159.4	162.6	166.0	168.8	171.8	174.6	177.2	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	97.1	109	109	115	121	126.2	131.7	137	142.8	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	20	20	20	20	20	20	20	20	20	20
Run No.										
Altitude - Ft	25065	26060	26980	27990	29015	29980	30995	31975	33010	
IAS - Knots	373.0	366.0	361.5	356.5	351.5	346.0	339.0	332.0	327.0	
OAT - °C	15.5	13.5	11.5	8.5	6.5	3.5	0	-2	-5.5	
RPM (high press)	9470	9465	9465	9465	9445	9415	9390	9365	9340	
RPM (low press)	5770	5785	5800	5820	5825	5820	5820	5800	5785	
Exhaust Gas Temp - °C	543	545	550	553	553	553	550	550	545	
Fuel Used (Eng) - Gal	103.2	104.3	105.1	106.2	107.3	108.2	109.2	110.2	111.2	
Fuel Used (A/B) - Gal	180.2	183.2	185.6	188.4	191.4	194.0	196.6	199.2	202.0	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0	0	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	148.7	154.5	160	166	172.7	178.8	179.4	191	198	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	20	20	20	20	20	20	20	20	20	20
Run No.										
Altitude - Ft	33970	34955	35925	36990	37935	38915	39910	40930	41970	
IAS - Knots	318.5	309.0	303.5	296.5	289.0	285.0	277.5	268.0	263.0	
OAT - °C	-7.5	-10	-13.5	-15	-14.5	-16.5	-16.5	-18	-19.5	
RPM (high press)	9315	9265	9230	9220	9190	9165	9165	9140	9110	
RPM (low press)	5785	5760	5750	5750	5730	5716	5735	5735	5720	
Exhaust Gas Temp - °C	545	545	543	543	543	540	543	545	545	
Fuel Used (Eng) - Gal	112.0	112.9	114.0	115.1	116.0	117.1	118.1	119.3	120.4	
Fuel Used (A/B) - Gal	209.2	2070	210.0	213.0	216.0	219.0	222.0	225.2	228.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0	0	0	0.2	0.33	0.7	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	204	211	220	228.8	237.9	247.4	258	269.1	281.9	

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	72870	72975	73795	73990	74920	75010	75985	76015	76180
IAS - Knots	262.0	262.0	261.5	257.5	253.5	248.5	245.5	242.0	238.5
OAT - °C	-20.5	-20.5	-21.5	-22	-23	-24	-24.8	-26	-27
RPM (high press)	9110	9110	9098	9095	9085	9080	9070	9060	9045
RPM (low press)	5725	5720	5720	5730	5735	5740	5740	5745	5735
Exhaust Gas Temp - °C	550	550	550	550	553	553	557	557	563
Fuel Used (Eng) - Gal	121.8	122.0	122.8	123.5	124.2	124.9	125.5	126.2	126.9
Fuel Used (A/B) - Gal	232.8	233.4	235.6	237.8	239.6	241.6	243.4	245.6	247.6
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	1.4	1.93	2.3	2.6	3.05	3.33	3.50	4.40	5.10
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	296.9	299	308	316.4	324	332.1	340	349	358

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	20	20	20	20	20	20	20	20	20
Run No.									
Altitude - Ft	47020	47630	48040	48515	49005	49730	50265	50660	50725
IAS - Knots	231.0	227.0	227.5	228.0	226.5	221.5	210.5	207.0	209.0
OAT - °C	-28	-30	-30	-30	-32	-32	-34	-35	-34.5
RPM (high press)	9045	9020	8990	8990	8990	8990	8990	8990	8960
RPM (low press)	5765	5735	5740	5735	5755	5775	5785	5805	5820
Exhaust Gas Temp - °C	563	567	567	570	570	570	575	583	575
Fuel Used (Eng) - Gal	127.6	128.4	129.3	130.4	131.4	132.5	133.5	134.5	135.1
Fuel Used (A/B) - Gal	249.6	252.0	254.6	257.6	260.8	263.8	266.8	269.6	271.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	5.75	6.80	7.85	8.40	9.00	10.70	11.75	13.70	15.95
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	367.2	379	390.8	405.2	420.2	435	450	465	475

Test	CHECK CLIMB ~ MAX POWER								
Flight No.	21	21	21	21	21	21	21	21	21
Run No.									
Altitude - Ft	3090	4095	4985	5935	6975	7915	8960	9980	10950
IAS - Knots	485.25	485.0	477.0	474.5	462.0	455.5	449.0	448.0	446.80
OAT - °C	66	66	67	62	58.5	56	53	50	48
RPM (high press)	9480	9500	9505	9505	9495	9485	9480	9465	9480
RPM (low press)	5720	5740	5745	5760	5765	5780	5515	5515	5575
Exhaust Gas Temp - °C	485	485	490	570	515	515	520	520	520
Fuel Used (Eng) - Gal	52.4	54.7	56.1	57.5	58.8	60.1	61.7	63.4	65.0
Fuel Used (A/B) - Gal	35.6	102.8	107.4	111.8	116.2	120.2	125.2	130.6	134.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	0	0	0	0	0	0	0	0	0
Time - Sec	0	10	17	23.5	30	36	44	52.5	60

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	11965	12950	13970	14920	15930	16955	17935	18970	20005	
IAS - Knots	495.86	492.0	497.5	491.0	429.5	416.75	409.5	404.75	397.25	
OAT - °C	46.0	45.5	40.0	37.0	37.0	30.5	29.5	27.5	25.0	
RPM (high press)	3780	3780	3765	3760	3760	3760	3770	3765	3760	
RPM (low press)	5550	5560	5570	5580	5605	5635	5660	5670	5690	
Exhaust Gas Temp - °C	522.5	522.5	530	530	540	540	540	540	540	
Fuel Used (Eng) - Gal	65.9	68.2	69.8	70.9	72.1	73.2	74.2	75.1	76.3	
Fuel Used (A/B) - Gal	140.6	145.2	149.8	153.7	156.6	159.8	162.6	165.6	168.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	68	76	83.5	89.5	95	100.5	105	111	117	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	21035	21980	22985	23990	25025	26050	27020	28020	29030	
IAS - Knots	397.25	393.75	388.0	379.5	376.75	370.25	364.0	355.5	350.25	
OAT - °C	24.0	22.0	20.0	18.0	16.5	14.0	12.0	9.0	6.0	
RPM (high press)	3760	3760	3760	3760	3760	3760	3760	3760	3745	
RPM (low press)	5690	5705	5725	5755	5770	5790	5800	5825	5835	
Exhaust Gas Temp - °C	540	540	540	540	540	540	550	550	555	
Fuel Used (Eng) - Gal	77.6	78.7	77.6	80.7	81.8	82.8	83.8	84.0	85.6	
Fuel Used (A/B) - Gal	172.6	175.6	178.7	181.2	184.7	187.2	189.8	192.2	194.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	127	130	135	140.5	147	153	159	167	169.5	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	30010	31005	31990	33010	33955	34970	35940	36960	37910	
IAS - Knots	343.25	333.75	327.0	323.0	318.5	314.75	303.5	300.0	286.5	
OAT - °C	2.5	-0.5	-4.0	-6.0	-8.0	-10.0	-12.0	-14.0	-15.5	
RPM (high press)	3740	3700	3665	3610	3555	3500	3435	3375	3210	
RPM (low press)	5840	5835	5810	5790	5765	5760	5760	5745	5760	
Exhaust Gas Temp - °C	555	555	555	542.5	542.5	540	540	540	540	
Fuel Used (Eng) - Gal	86.5	87.3	88.3	89.5	90.3	91.8	92.7	93.7	97.5	
Fuel Used (A/B) - Gal	197.0	199.7	202.0	205.2	208.0	211.6	214.4	217.0	219.4	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	175	180.5	187	195	203	212.5	220	228	236	

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	38930	39945	40930	41970	42860	43730	44710	45785	46715	
IAS - Knots	281.5	271.5	266.5	259.5	256.0	255.0	248.0	237.25	225.7	
OAT - °C	-16.0	-18.0	-20.	-21	-22	-22.5	-24	-26	-28	
RPM (high press)	9190	9175	9110	9120	9035	9090	9065	9065	9070	
RPM (low press)	5740	5745	5715	5735	5730	5750	5730	5745	5750	
Exhaust Gas Temp - °C	570	572.5	572.5	555	555	555	555	560	565	
Fuel Used (Eng) - Gal	95.5	96.7	97.4	98.6	99.9	101.1	102.5	103.6	104.8	
Fuel Used (A/B) - Gal	222.2	227.6	227.8	231.2	235.0	238.8	242.6	246.0	249.4	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0.3	1.4	2.1	2.6	3.8	4.4	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	245	255	264	277	292	307	322	337	352	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	47155	47465	47975	48840	49630	49980	50235	50255	50310	
IAS - Knots	228.0	230.5	231.0	225.2	213.5	209.5	208.0	210.5	211.5	
OAT - °C	-29	-29	-29	-30	-32	-34	-34	-34	-33	
RPM (high press)	9010	9010	9010	9000	9010	8990	8970	8970	8970	
RPM (low press)	5735	5730	5740	5745	5790	5765	5745	5740	5735	
Exhaust Gas Temp - °C	560	565	565	582.5	582.5	582.5	582.5	582.5	582.5	
Fuel Used (Eng) - Gal	105.9	106.9	108.1	109.2	110.3	111.2	112.2	113.2	114.2	
Fuel Used (A/B) - Gal	252.6	255.8	259.0	262.2	265.2	268.2	271.3	274.0	276.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	6.3	7.7	8.4	9.3	10.7	12.2	14.2	15.75	17.9	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	367	382	397	412	427	442	457	472	487	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	50380	50340	50455	50405	50550	50590	50657	50710	50695	
IAS - Knots	210.5	213.5	212.5	217.0	210.0	206.5	201.5	202.5	207.0	
OAT - °C	-32.5	-32	-32	-32	-33	-34	-35	-35	-34	
RPM (high press)	8970	8970	8970	8970	8970	8970	8970	8970	8935	
RPM (low press)	5735	5730	5725	5745	5750	5740	5730	5725	5715	
Exhaust Gas Temp - °C	582.5	582.5	575	575	575	582.5	582.5	582.5	582.5	
Fuel Used (Eng) - Gal	119.8	116.2	117.2	118.1	119.0	120.0	120.9	121.9	122.0	
Fuel Used (A/B) - Gal	279.8	282.5	285.6	288.4	291.2	294.0	296.0	299.6	302.7	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	18.7	18.7	18.7	18.7	18.7	18.7	18.7	18.7	18.7	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	502	517	532	547	562	577	592	607	622	

DATA CORRECTED FOR INSTRUMENT ERROR
 YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21	21	21	21	21	21	21	21	21
Run No.										
Altitude - Ft	50550	50240	50765	51100	51750	52280	52730	53295	53775	
IAS - Knots	207.5	218.0	209.5	207.0	202.5	197.5	195.25	192.15	186.0	
OAT - °C	-34	-32	-32	-32.5	-34	-36	-37	-38	-40	
RPM (high press)	8935	8925	8940	8940	8940	8925	8910	8865	8890	
RPM (low press)	5710	5695	5715	5720	5715	5725	5725	5755	5775	
Exhaust Gas Temp - °C	582.5	582.5	582.5	582.5	572.5	582.5	582.5	585	585	
Fuel Used (Eng) - Gal	123.8	124.8	126.6	128.0	129.9	132.7	137.3	146.0	150.8	
Fuel Used (A/B) - Gal	305.2	308.2	313.6	317.8	323.4	331.6	345.4	371.8	386.2	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4	18.4	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	0	0	0	0	0	0	0	0	0	
Time - Sec	637	652	687	704	724	780	799	949	1032	

Test	CHECK CLIMB ~ MAX POWER									
Flight No.	21	21								
Run No.										
Altitude - Ft	54390	54735								
IAS - Knots	177.5	173.0								
OAT - °C	-42	-42								
RPM (high press)	8870	8765								
RPM (low press)	585	585								
Exhaust Gas Temp - °C	585	585								
Fuel Used (Eng) - Gal	153.1	154.2								
Fuel Used (A/B) - Gal	393.2	396.6								
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	18.4	18.4								
Bleed Valve Position	C	C								
Fuel Weight - lbs/gal										
Nozzle Position	0	0								
Time - Sec	1073	1095								

Test										
Flight No.										
Run No.										
Altitude - Ft										
IAS - Knots										
OAT - °C										
RPM (high press)										
RPM (low press)										
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.										
Bleed Valve Position										
Fuel Weight - lbs/gal										
Nozzle Position										
Time - Sec										

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB - MILITARY POWER									
Flight No.	1	1	1	1	1	1	1	1	1	1
Run No.										
Altitude - Ft	3999	4957	5974	6977	7991	8987	9989	10978	11978	
IAS - Knots	404.3	401.8	401.8	397.3	391.3	396.3	384.9	382.9	376.7	
OAT - °C	56.0	54.0	52.0	48.0	46.0	43.0	39.0	36.0	36.0	
RPM (high press)	9545	9540	9545	9545	9545	9545	9545	9545	9555	
RPM (low press)	5330	5550	5660	5580	5600	5690	5655	5670	5660	
Exhaust Gas Temp - °C	523	523	527	527	525	540	540	540	543	
Fuel Used (Eng) - Gal	72.0	79.0	89.0	96.0	105	111	115	121	126	
Fuel Used (A/B) - Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.°C	0.1	0.1	0.1	0.1	0.1	0.1	0.35	0.35	0.35	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	0	30.5	74.6	102.6	146.6	177.5	195.4	223.6	246.6	

Test	CHECK CLIMB - MILITARY POWER									
Flight No.	1	1	1	1	1	1	1	1	1	1
Run No.										
Altitude - Ft	12968	13990	14970	15989	16993	17948	18972	19976	20975	
IAS - Knots	364.2	378.3	366.2	367.7	360.7	357.7	354.2	350.7	346.2	
OAT - °C	34.0	36.0	32.5	31.5	28.5	26.0	23.5	21.5	19.5	
RPM (high press)	9555	9545	9565	9540	9555	9545	9540	9480	9545	
RPM (low press)	5680	5670	5710	5720	5740	5765	5790	5780	5840	
Exhaust Gas Temp - °C	543	546	543	540	540	540	543	519	540	
Fuel Used (Eng) - Gal	130	146	143	149	152	157	161	183	187	
Fuel Used (A/B) - Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.°C	0.55	1.55	1.80	2.2	2.2	2.2	2.5	2.3	8.4	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	267.4	311.9	326.2	358.8	376.2	398.6	420.6	566.8	589.6	

Test	CHECK CLIMB - MILITARY POWER									
Flight No.	1	1	1	1	1	1	1	1	1	1
Run No.										
Altitude - Ft	21970	22985	23954	24957	25944	26929	27925	28896	29850	
IAS - Knots	391.2	338.8	334.3	330.3	328.9	324.0	320.0	320.5	313.3	
OAT - °C	16.5	14.5	12.0	10.0	6.5	4.0	1.0	0	-4.0	
RPM (high press)	9545	9530	9510	9475	9455	9410	9380	9350	9325	
RPM (low press)	5855	5870	5870	5840	5840	5830	5810	5800	5795	
Exhaust Gas Temp - °C	546	550	546	543	543	540	540	540	545	
Fuel Used (Eng) - Gal	192	177	200	205	209	212	216	222	225	
Fuel Used (A/B) - Gal	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	76.6	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.°C	8.4	8.4	7.95	7.95	7.95	7.45	7.95	7.95	7.95	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	619.9	629.7	658.9	682.9	707.8	728.6	752.5	788.1	809.9	

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER									
Flight No.	1	1	1	1	1	1	1	1	1	1
Run No.										
Altitude - Ft	30818	31783	32760	32729	32696	35750	36603	37581	38523	
IAS - Knots	308.4	306.5	302.0	297.0	292.2	285.6	281.8	276.7	269.5	
OAT - °C	-6.0	-8.5	-12.0	-14.0	-15.5	-18.0	-19.5	-20.0	-21.5	
RPM (high press)	9280	9260	9205	9200	9175	9150	9130	9100	9095	
RPM (low press)	5780	5770	5745	5750	5755	5740	5745	5755	5750	
Exhaust Gas Temp - °C	535	531	519	519	519	519	519	523	527	
Fuel Used (Eng) - Gal	228	234	238	242	248	251	258	264	270	
Fuel Used (A/B) - Gal	76.8	76.8	77.0	77.0	77.2	77.4	77.6	77.8	78.0	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.°C	7.45	7.45	7.45	7.45	7.7	7.7	8.8	10.2	11.6	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	834.9	873.6	910.8	938.1	982.9	1014.4	1070.4	1125.6	1184.9	

Test	CHECK CLIMB ~ MILITARY POWER									
Flight No.	1	1	1	1		7	7	7	7	
Run No.										
Altitude - Ft	39515	40492	41475	42437		3050	4065	7360	5175	
IAS - Knots	246.0	263.0	256.0	249.0		407.5	405.5	398.75	395.5	
OAT - °C	-22.5	-23.5	-25.0	-27.5						
RPM (high press)	9285	9140	9035	8990		9530	9535	9535	9535	
RPM (low press)	5730	5740	5745	5740		5510	5535	5545	5555	
Exhaust Gas Temp - °C	535	535	540	535						
Fuel Used (Eng) - Gal	278	307	313	325		59.5	70.2	72.8	80.3	
Fuel Used (A/B) - Gal	78.4	78.6	78.6	78.6		66.8	66.8	66.8	66.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.°C	14.0	18.4	16.4	18.4		0	0	0	0	
Bleed Valve Position	C	C	C	C		C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C		C	C	C	C	
Time - Sec	1266.6	1568.4	1634.8	1875.9		0	48.5	60.5	95.5	

Test	CHECK CLIMB ~ MILITARY POWER									
Flight No.	7	7	7	7	7	7	7	7	7	7
Run No.										
Altitude - Ft	5330	6030	6535	7025	8015	8530	8975	9350	9915	
IAS - Knots	400.5	394.5	394.0	394.5	391.0	387.5	390.0	385.0	382.5	
OAT - °C										
RPM (high press)	9535	9535	9535	9535	9535	9535	9535	9535	9535	
RPM (low press)	5555	5575	5585	5590	5610	5615	5625	5645	5665	
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal	83.3	87.5	91.5	95.9	101.7	104.7	108.0	110.7	114.5	
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	0	0	0	0	0	0	0	0	0	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	114	128	176.5	167.5	194.5	207.5	223	236	253.5	

DATA CORRECTED FOR INSTRUMENT ERROR

YF-100 USAF No. 52-5754

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	10770	11690	12740	13905	14300	15100	15935	16855	17605
IAS - Knots	381.0	379.0	376.0	368.5	365.5	366.0	363.0	360.5	359.0
OAT - °C									
RPM (high press)	9525	9525	9525	9525	9525	9510	9510	9510	9510
RPM (low press)	5660	5685	5715	5725	5740	5770	5785	5795	5800
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	119.7	125.2	131.3	135.8	137.9	171.6	175.4	199.7	152.3
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8	66.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.3	0.3	0.65	0.55	0.55	0.55	0.55	0.55	0.75
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	278	303.5	333.5	354	364.5	382.4	401	421.5	435.5

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	18740	18985	19455	19965	20700	21005	21995	22840	23275
IAS - Knots	354.5	354.0	352.0	349.0	343.5	344.0	344.5	338.5	335.5
OAT - °C									
RPM (high press)	9510	9515	9515	9515	9515	9505	9500	9485	9460
RPM (low press)	5800	5815	5815	5840	5855	5855	5855	5845	5845
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	156.2	158.1	159.5	161.3	164.7	166.4	169.9	172.5	174.6
Fuel Used (A/B) - Gal	66.8	66.8	66.8	66.8	66.8	66.8	66.8	67.0	67.0
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	0.75	1.0	1.0	1.0	1.0	1.25	1.25	1.25	1.5
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	454.5	464.5	471.5	480.5	493	507.5	526.5	541	552

Test	CHECK CLIMB - MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	24025	24970	25300	26035	26455	27020	27470	27950	28855
IAS - Knots	336.5	331	328.5	327	326.5	325.5	323.5	321.5	317
OAT - °C									
RPM (high press)	9450	9435	9415	9395	9390	9360	9350	9340	9315
RPM (low press)	5835	5830	5830	5820	5815	5805	5800	5795	5785
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	177.4	180.5	181.7	184.7	186.5	189.1	191.1	193.3	197
Fuel Used (A/B) - Gal	67	67	67	67	67	67	67	67.2	67.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	2.05	1.9	2.3	2.6	2.6	2.9	3.3	3.3	3.8
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	568.5	586.5	593.5	611.5	623.5	639.5	652.5	671.5	682

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	29505	29960	30685	30965	31535	31550	31820	32310	32975
IAS - Knots	313.5	313.0	309	306.5	304.5	307.5	305	307.5	307
OAT - °C									
RPM (high press)	9295	9275	9265	9250	9240	9230	9220	9220	9225
RPM (low press)	5765	5760	5760	5750	5735	5745	5715	5740	5750
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	199.4	201.5	203.9	204.8	207.1	208.9	205.1	213.9	215.9
Fuel Used (A/B) - Gal	67.2	67.4	67.4	67.4	67.4	67.4	67.8	68	68.2
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	3.9	4.4	4.75	4.75	5.5	5.7	5.7	6.6	6.8
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	708.5	723.5	741.5	744.5	765	778.5	791	818	834.5

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	33555	34085	34465	35105	35310	35715	36235	36730	37050
IAS - Knots	297.5	296.5	294.5	287	289	287	281.5	277.5	275.5
OAT - °C									
RPM (high press)	9215	9190	9190	9170	9155	9145	9140	9120	9115
RPM (low press)	5735	5730	5745	5740	5725	5720	5735	5725	5725
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	218.1	220.2	221.3	224	225.8	2278	229.8	232.1	233.9
Fuel Used (A/B) - Gal	68.4	68.1	68.6	68.8	69	69	69.4	69.6	69.8
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	6.9	6.9	7.1	7.7	8.0	8.35	8.6	9.0	9.75
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	852	862.5	851.5	90.3	919.5	931.5	956.5	978	996

Test	CHECK CLIMB ~ MILITARY POWER								
Flight No.	7	7	7	7	7	7	7	7	7
Run No.									
Altitude - Ft	37060	37675	38100	38235	38520	38670	39020	39265	39465
IAS - Knots	275	278.5	274.5	273.5	273	273	270.5	270	268.5
OAT - °C									
RPM (high press)	9100	9095	9095	9095	9070	9070	9065	9040	9040
RPM (low press)	5730	5705	5725	5735	5725	5730	5730	5725	5730
Exhaust Gas Temp - °C									
Fuel Used (Eng) - Gal	234	240.3	241.9	242.7	244.4	246	248.5	249.7	251.9
Fuel Used (A/B) - Gal	69.8	70	70	70.2	70.2	70.2	70.4	70.4	70.4
Fuel Flow (total) - GPH									
Oil Cooler GAP - in.	9.75	11.4	11.4	11.7	12.5	12.5	13	13.4	14
Bleed Valve Position	C	C	C	C	C	C	C	C	C
Fuel Weight - lbs/gal									
Nozzle Position	C	C	C	C	C	C	C	C	C
Time - Sec	996.5	1058.5	1074.5	1082	1099.5	1117.5	1141.5	1154.5	1177.5

DATA CORRECTED FOR INSTRUMENT ERROR
YF-100 USAF No. 52-5754

Test	CHECK CLIMB ~ MILITARY POWER									
Flight No.	7	7	7	7	7	7	7	7	7	7
Run No.										
Altitude - Ft	39660	40060	40215	40420	40840	40855	40845	40975	40865	
IAS - Knots	266.5	263	260	259	256.5	257	257.5	253.5	254	
OAT - °C										
RPM (high press)	9040	9020	9020	9015	9020	9020	8995	8995	8995	
RPM (low press)	572.5	572.5	572.5	572.0	572.5	572.5	570.5	573.0	572.0	
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal	253.5	256.9	258.2	259.5	264.5	265	269.9	271.3	278.5	
Fuel Used (A/B) - Gal	70.4	70.4	70.4	70.4	70.4	70.4	70.4	70.4	70.4	
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	14.3	15.7	16.2	16.7	18.4	18.4	18.4	18.4	18.4	
Bleed Valve Position	C	C	C	C	C	C	C	C	C	
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C	C	C	C	C	C	
Time - Sec	1194	1231	1245.5	1259.5	1315.5	1322	1377	1393.5	1474.5	

Test	CHECK CLIMB ~ MILITARY POWER									
Flight No.	7	7	7	7						
Run No.										
Altitude - Ft	41360	41510	41655	41890						
IAS - Knots	247.5	246	245.5	243						
OAT - °C										
RPM (high press)	899.5	896.0	896.0	897.0						
RPM (low press)	572.5	572.0	572.0	571.5						
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal	280.8	282.3	283.7	286.2						
Fuel Used (A/B) - Gal	70.4	70.4	70.4	70.4						
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.	18.4	18.4	18.4	18.4						
Bleed Valve Position	C	C	C	C						
Fuel Weight - lbs/gal										
Nozzle Position	C	C	C	C						
Time - Sec	1501	1519.5	1535	1564.5						

Test										
Flight No.										
Run No.										
Altitude - Ft										
IAS - Knots										
OAT - °C										
RPM (high press)										
RPM (low press)										
Exhaust Gas Temp - °C										
Fuel Used (Eng) - Gal										
Fuel Used (A/B) - Gal										
Fuel Flow (total) - GPH										
Oil Cooler GAP - in.										
Bleed Valve Position										
Fuel Weight - lbs/gal										
Nozzle Position										
Time - Sec										

FIGURE No. /
TAKE OFF
YF-100A USAF No 52-5754

FLT NO	4	N ₂ @ T.O	9,500	RPM
GROSS WT	25,000 LBS	N ₁ @ T.O	5,630	RPM
CG	31 %MAC	F ₆ @ T.O	8,760	LBS
PRESS ALT	2,500 FT	V ₆ @ T.O	172	KNOTS
VAS @ T.O	146 KNOTS	V ₆ @ 50'	192.5	KNOTS
VAS @ 50'	NO DATA	EGT	560	°C
AFTER BURNER OPERATION	ON	FAT	28.5	°C

WIND CALM

TIME IN SECONDS

HEIGHT in FT

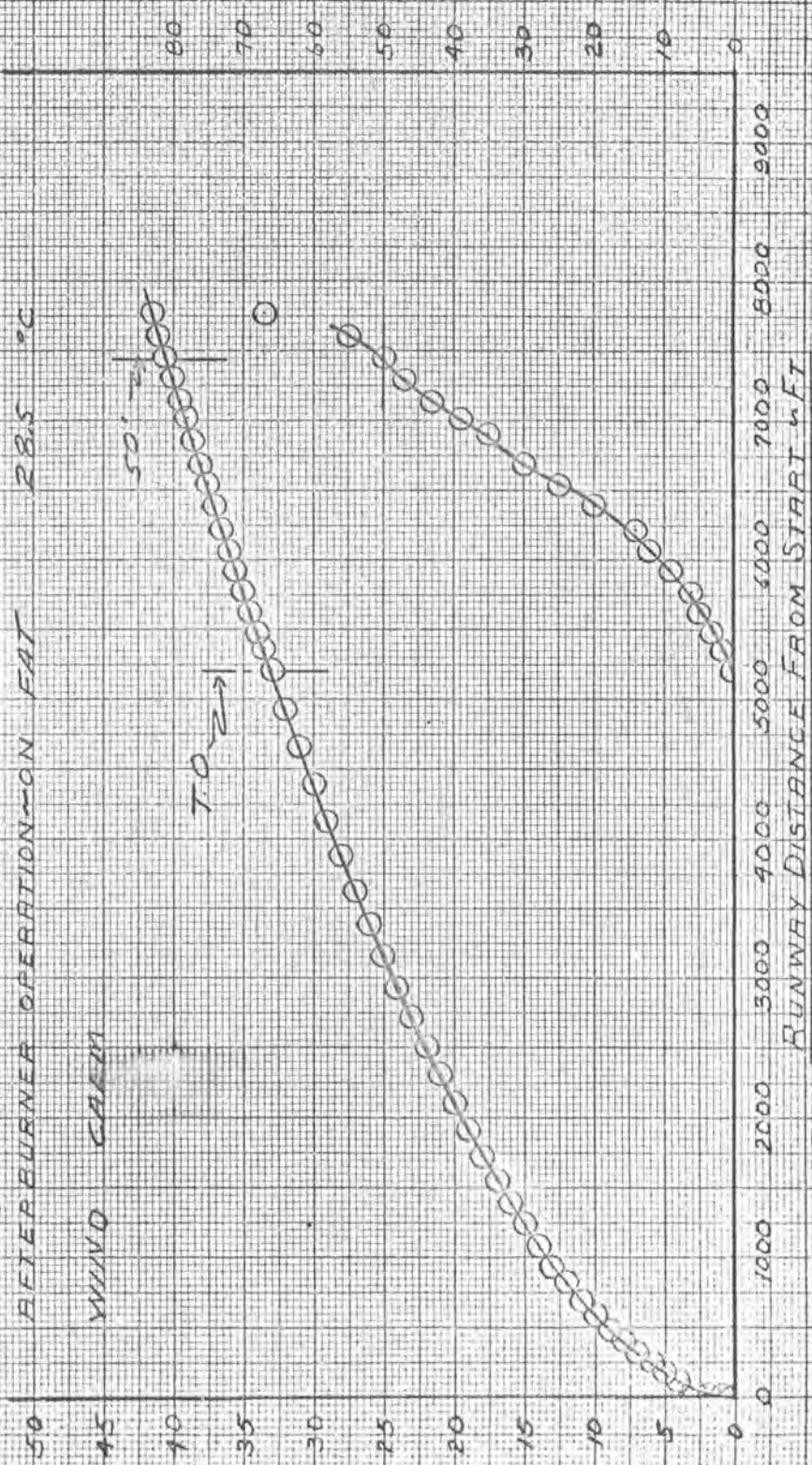


FIGURE NO. 2
TAKE OFF
YF-100A USAF No 52-5754

FLT NO	5	N ₂ @ TO	9500	RPM
GROSS WT	25,000	N ₁ @ TO	5600	RPM
CG	31	GROSS THRUST	8310	LEBS @ TO
PRESS ALT	2,330	V _L @ TO	166	KNOTS
IAS @ TO	148	V _C @ 50'	NO DATA	KNOTS
IAS @ 50'	NO DATA	EGT	560	°C
AFTERBURNER OPERATION	NO	FAT	34	°C

WIND CALM

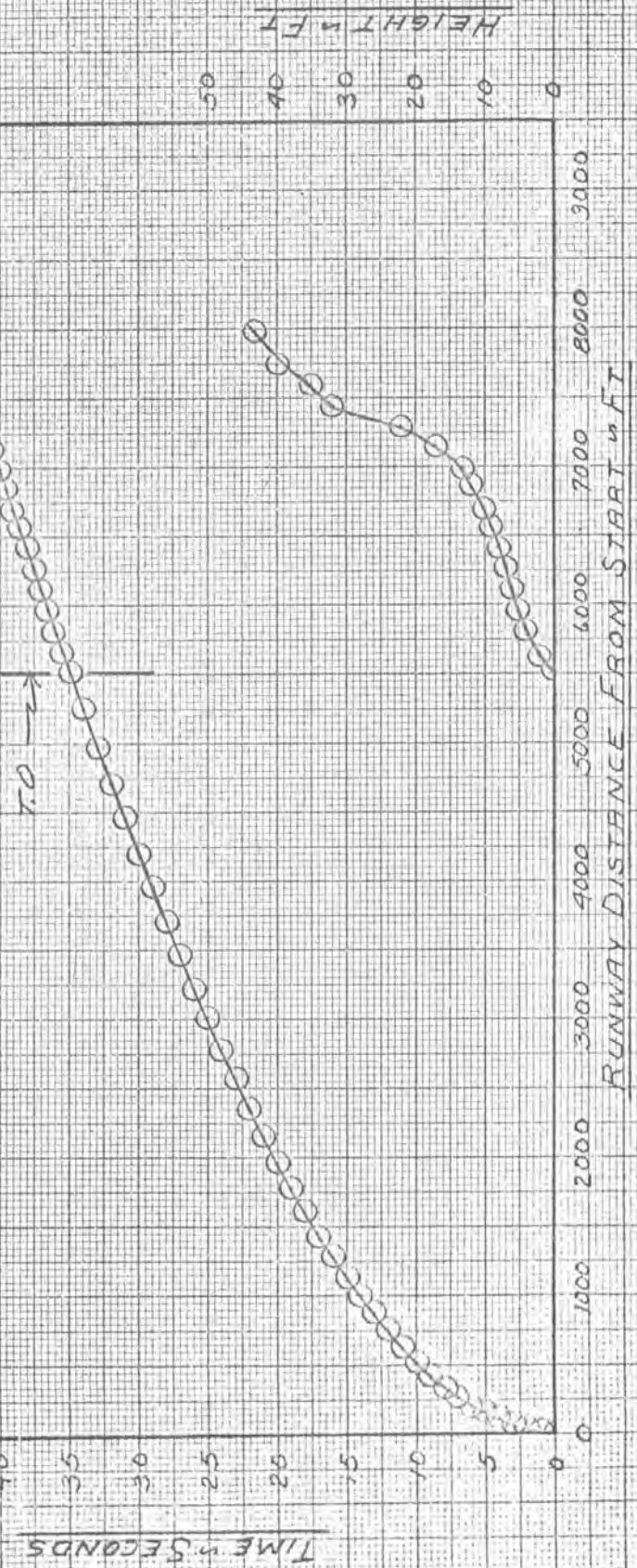


FIGURE No J
TABLE OFF
YF-100A USAF No 52-5754

FLT NO	14(N)	N ₂ @ T ₀	3480 RPM
GROSS WT	25,100 LBS	N ₁ @ T ₀	5,100 RPM
CG	31.1 %MAC	GROSS THRUST	8,950 LBS @ T ₀
PRESS ALT	2,270 FT	V ₆ @ T ₀	157 KNOTS
VAS @ T ₀	147 KNOTS	V ₆ @ 50'	191 KNOTS
VAS @ 50'	NO DATA	EGT	560°C
AFTERBURNER OPERATION	ON	FAT	31°C

WIND CALM

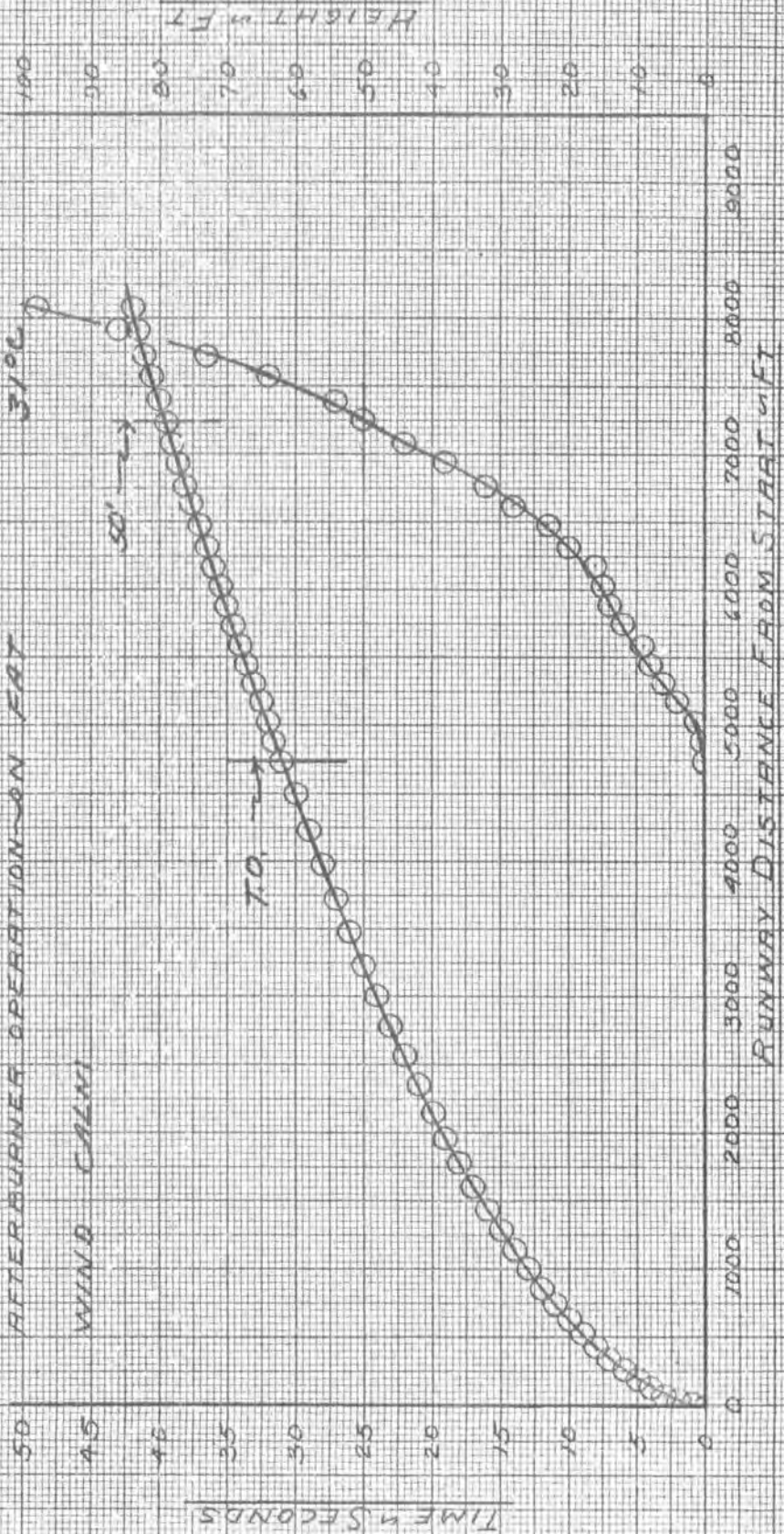


FIGURE No 4
 TAKE OFF
 YF-100A USAF No 52-5754

FLT NO	15	N ₂ @ T.O	9620 RPM
GROSS WT	25,100 LBS	N ₁ @ T.O	5145 RPM
CG	341	GROSS THRUST @ T.O	11,380 LBS
PRESS ALT	2,275 FT	V ₆ @ T.O	160 KNOTS
IAS @ T.O	145 KNOTS	V ₆ @ 50'	180.5 KNOTS
IAS @ 50'	NO DATA	EGT	545 °C
AFTER BURNER OPERATION	NO ON	FMT	38° C

WIND CALM

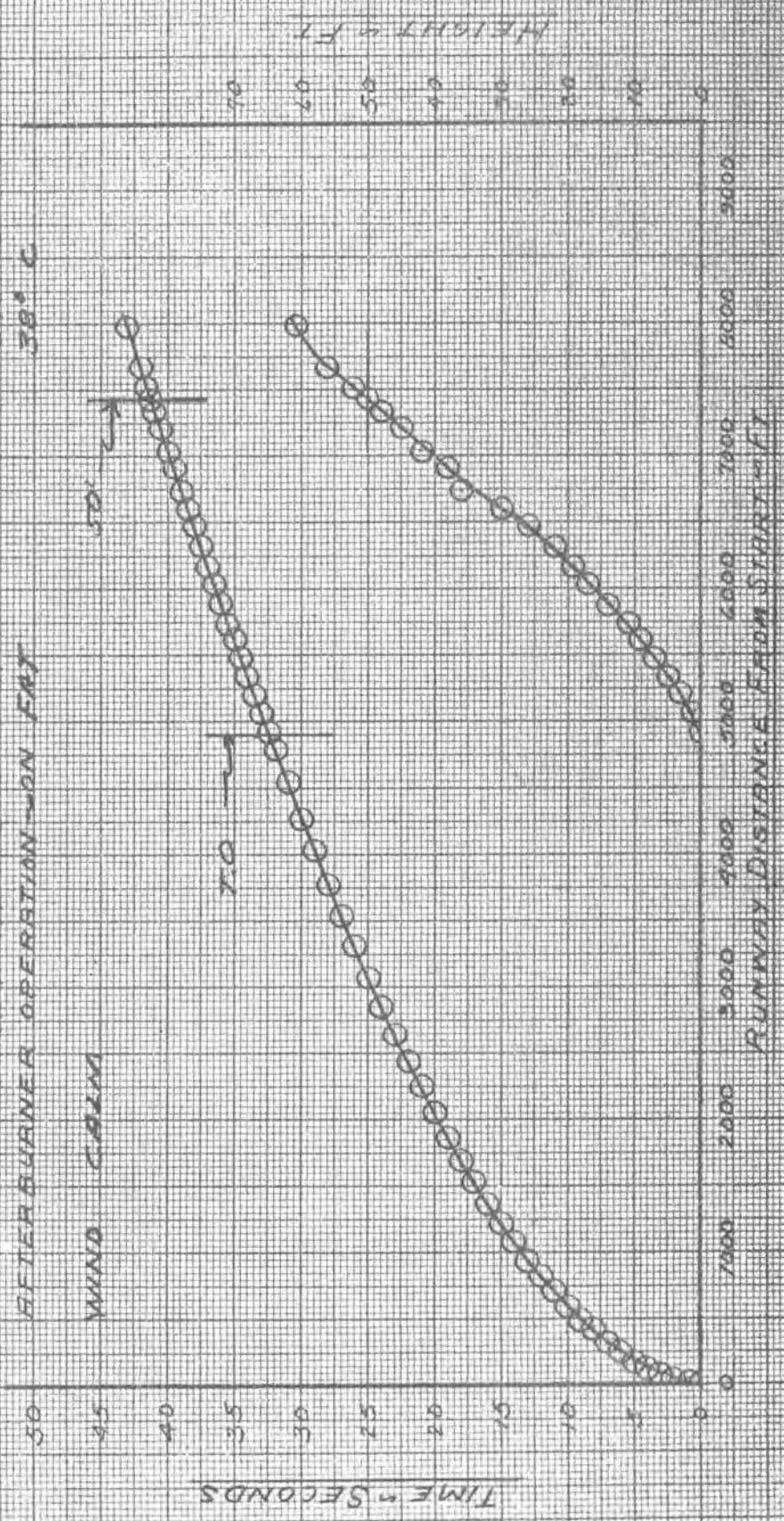


FIGURE No 5
TAKE OFF
YF-100A USAF No 32-5754

FLT NO	20	N ₂ @ T.O	3460 RPM
GROSS WT	24,930 LBS	N ₁ @ T.O	5,160 RPM
CG	31	GROSS THRUST @ T.O	9,560 LBS
PRESS ALT	2,320 FT	V ₅ @ T.O	141.5 KNOTS
IAS @ T.O	148 KNOTS	V ₆ @ 50'	188.5 KNOTS
IAS @ 50'	NO DATA	EGT	555 °C
AFTERBURNER OPERATION - ON		FAT	22 PC

WIND CALM

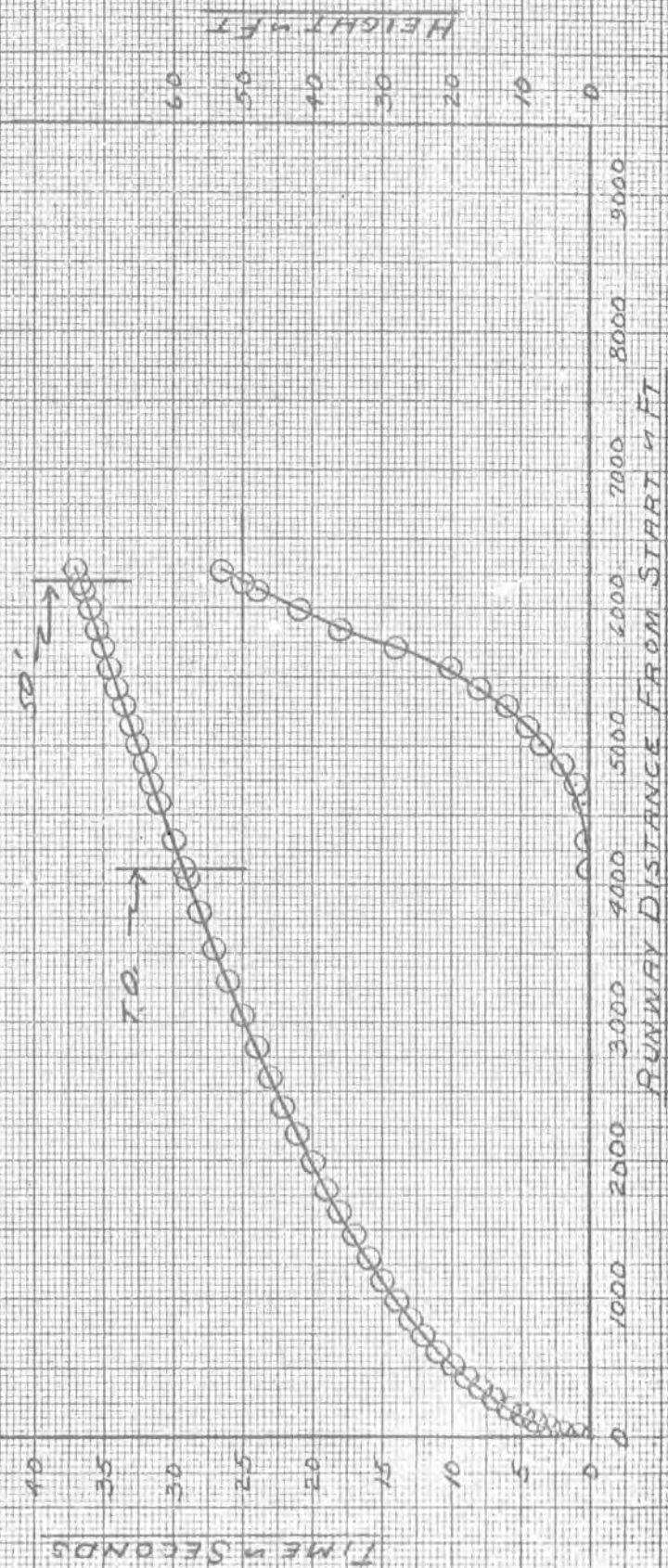


FIGURE No 6
TAKE OFF
YF-100A USAF No 52-5754

FLT NO	21	N ₂ @ TO	3460 RPM
GROSS WT	25,070 LBS	N ₁ @ TO	5120 RPM
CG	31.1 %MAC	GROSS THRUST @ TO	8,980 LBS
PRESS ALT	2,310 FT	V ₆ @ TO	163.5 KNOTS
LBS @ TO	143.5 KNOTS	V ₆ @ 50'	163.5 KNOTS
IAS @ 50'	NO DATA ~ KNOTS	F _{6T}	540 °C
REFUELLER OPERATION	~ ON	F _{6T}	29 °C

WIND CALM

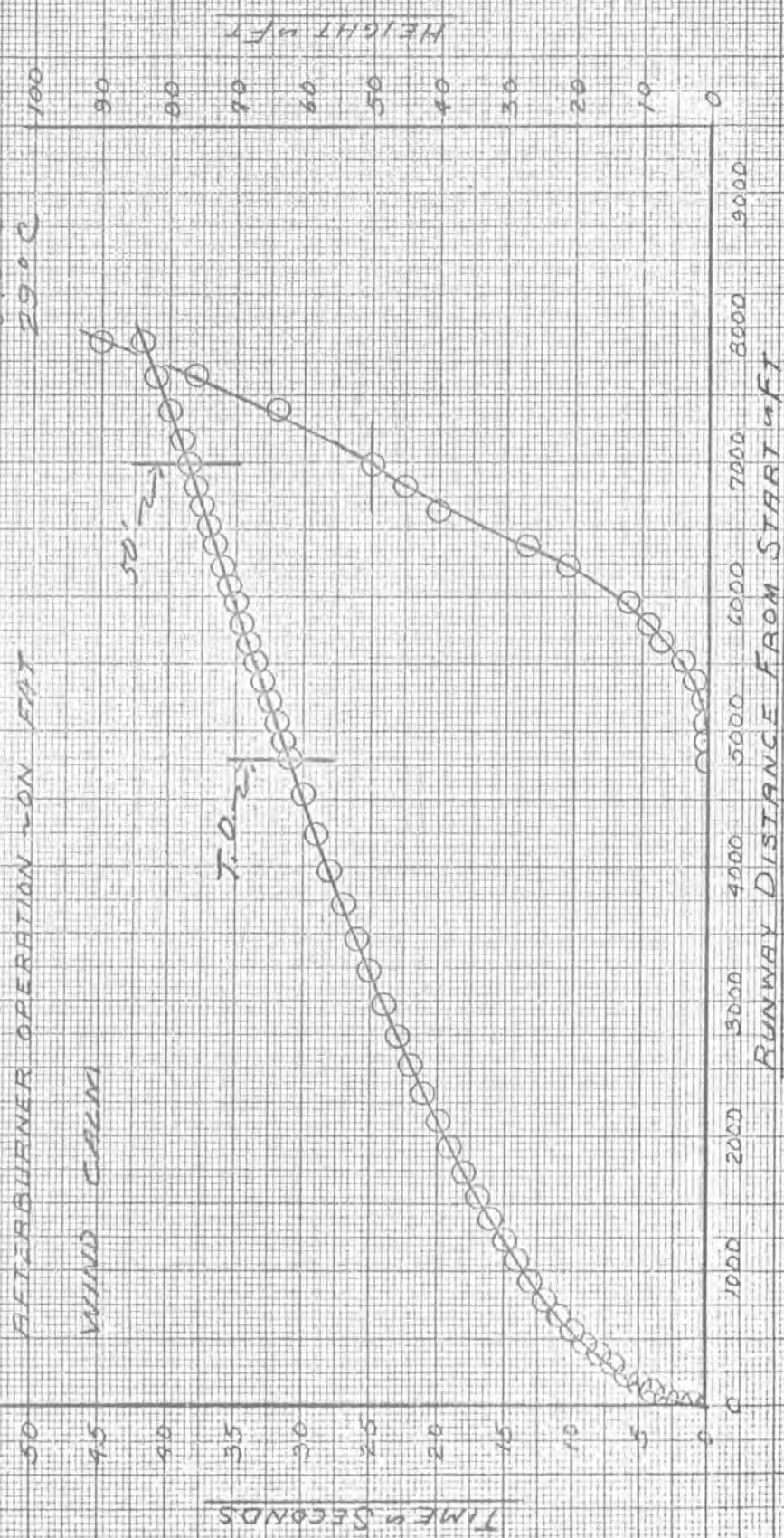


FIGURE No 7
 TAKE OFF
 YF-100A USAF No 52-5754

FLT NO	14(2)	N ₂ @ TO	3480 RPM
GROSS WT	21600 LBS	N ₁ @ TO	3555 RPM
C.G.	2917 %M/MC	GROSS THRUST @ TO	3680 LBS
PRESS ALT	2280 FT	V ₆ @ TO	162.5 KNOTS
IAS @ TO	147 KNOTS	V ₅ @ 50'	~ KNOTS
IAS @ 60'	NO DATA	EGT	535 °C
AFTERBURNER OPERATION	OFF	32°C	

WIND - CALM

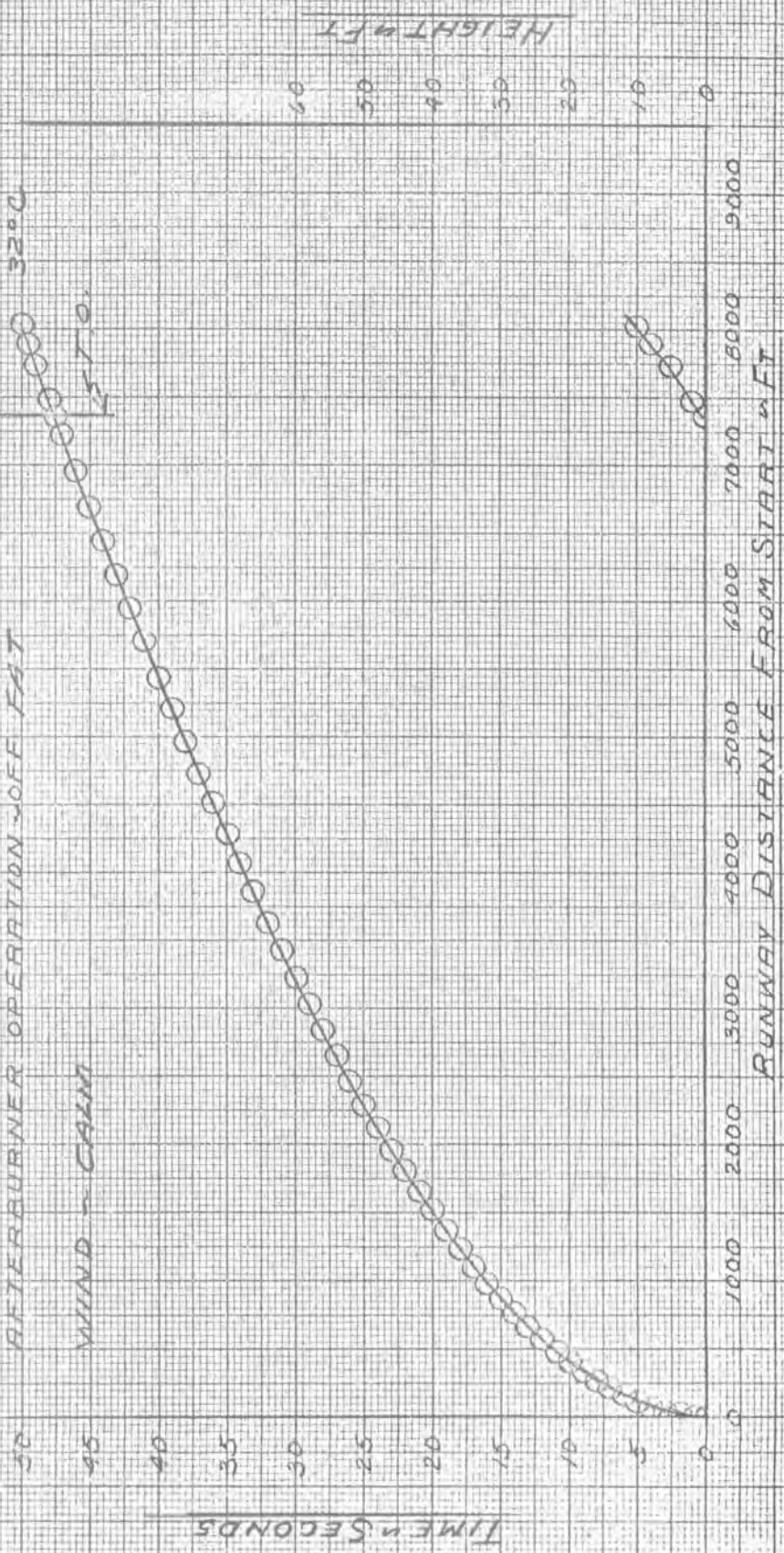


FIGURE No B
 LANDING
 YF-100A USAF No 52-5754

FLT NO	28	IAS @ TD	142	KNOTS
GROSS WT	20,800 LBS	N ₂		IDLF RPM
CG	31.7 %MAC	V ₃ @ TD	139	KNOTS
PRESS ALT	3,480 FT	DRAG CHUTE DEPLOYED	~	YES
IAS @ 60' ~ NO DATA	~ KNOTS	FRT		90 °C



FIGURE No. 9
 LANDING
 YF-100A USAF No. 52-5754

FLT NO.	28	IAS @ TD	—	KNOTS
GROSS WT	23,300 LBS	N ₂	—	RPM
CG	31.7 INAC	V _G @ TD	146.5	KNOTS
PRESS ALT	2,430 FT	DRAG CHUTE DEPLOYED	—	YES
IAS @ 50'	NO DATA	FAT	—	37 ° C



FIGURE NO 10
LANDING
YF-100A USRAF No 52-5754

FLT NO	30	IAS @ T.D		IAS @ T.D	
GROSS WEIGHT	22,700 LBS	N ₂		N ₂	
CG	30.9 %MAC	V ₆ @ T.D		V ₆ @ T.D	
PRESS ALT	2,430 FT	DRAG CHUTE DEPLOYED	YES	DRAG CHUTE DEPLOYED	YES
IAS @ 50'	NO DATA	IAS @ T.D	35 °C	IAS @ T.D	35 °C

→ WIND 15 KNOTS
← L.D.

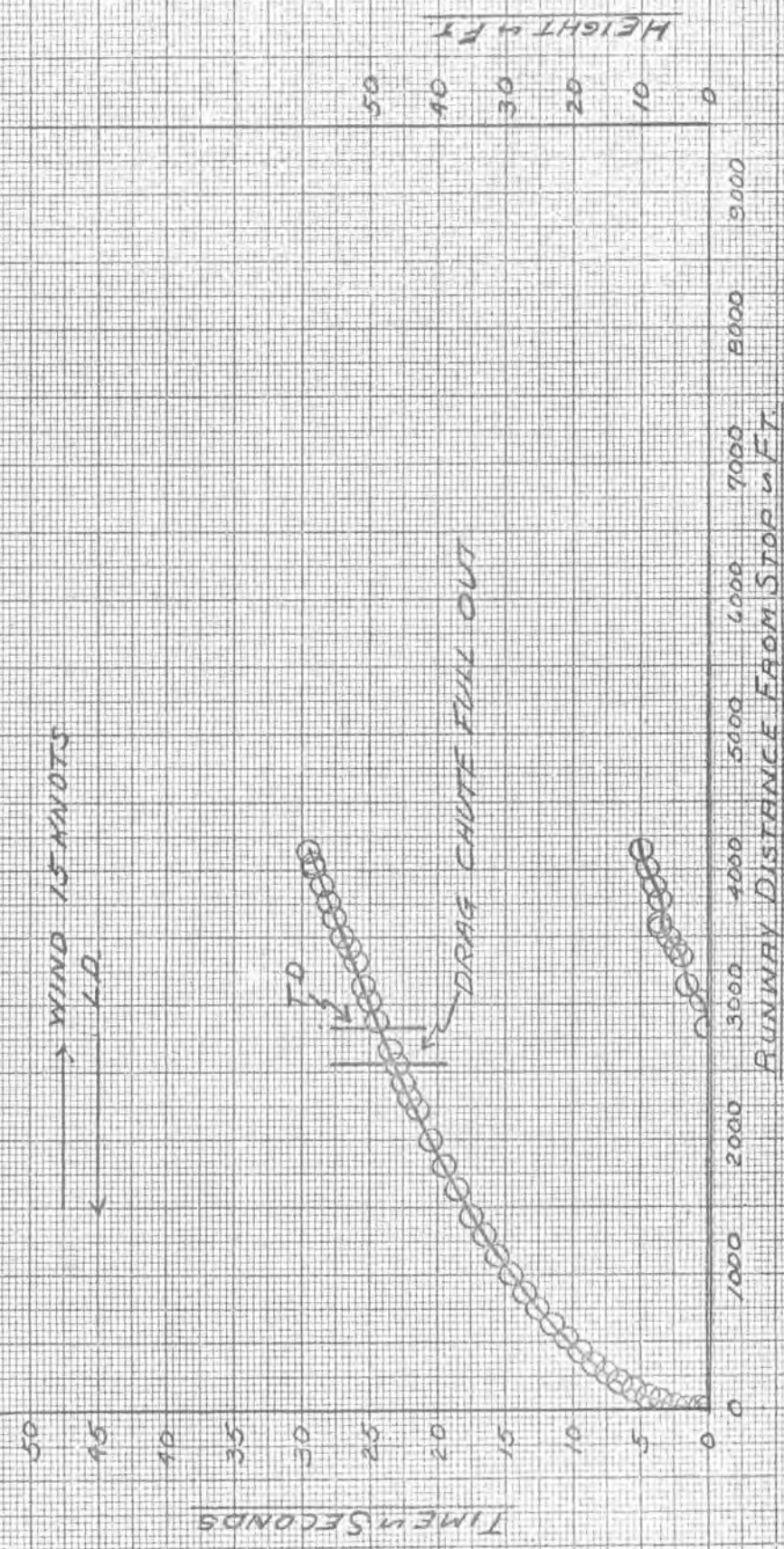


FIGURE No. 11
LANDING
YF-100B USAF No 52-5754

FLT NO	13	IAS @ TD	143	KNOTS
GROSS WT	20,600	N ₂		IDLE RPM
CG	31.3	%MAC	146.5	KNOTS
PRESS ALT	2,300	FT		DRAG CHUTE DEPLOYED NO
IAS @ 50'	NO DATA	KNOTS	147	2400

WIND CALM

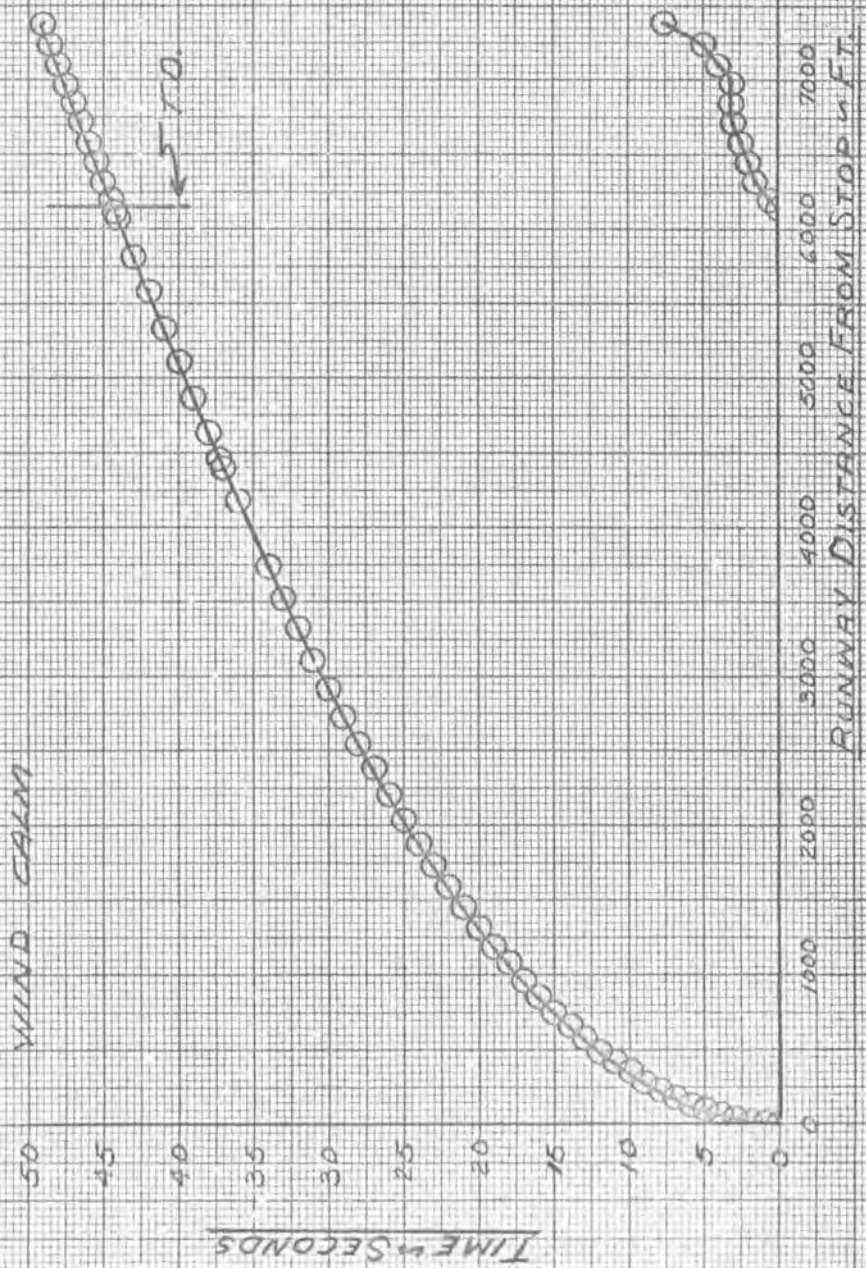


FIGURE No 12
LANDING
YF-100A USAF No 52-5754

FLT NO	14(U)	IAS @ TD	142	KNOTS
GROSS WT	21,700	N ₂	IDLE	
CG	29.4	1/2 @ TD	151	KNOTS
PRESS ALT	2,280	DRAW CHUTE DEPLOYED	NO	
IAS @ 50'	NO DATA	FEET	35	°C

WIND CALM

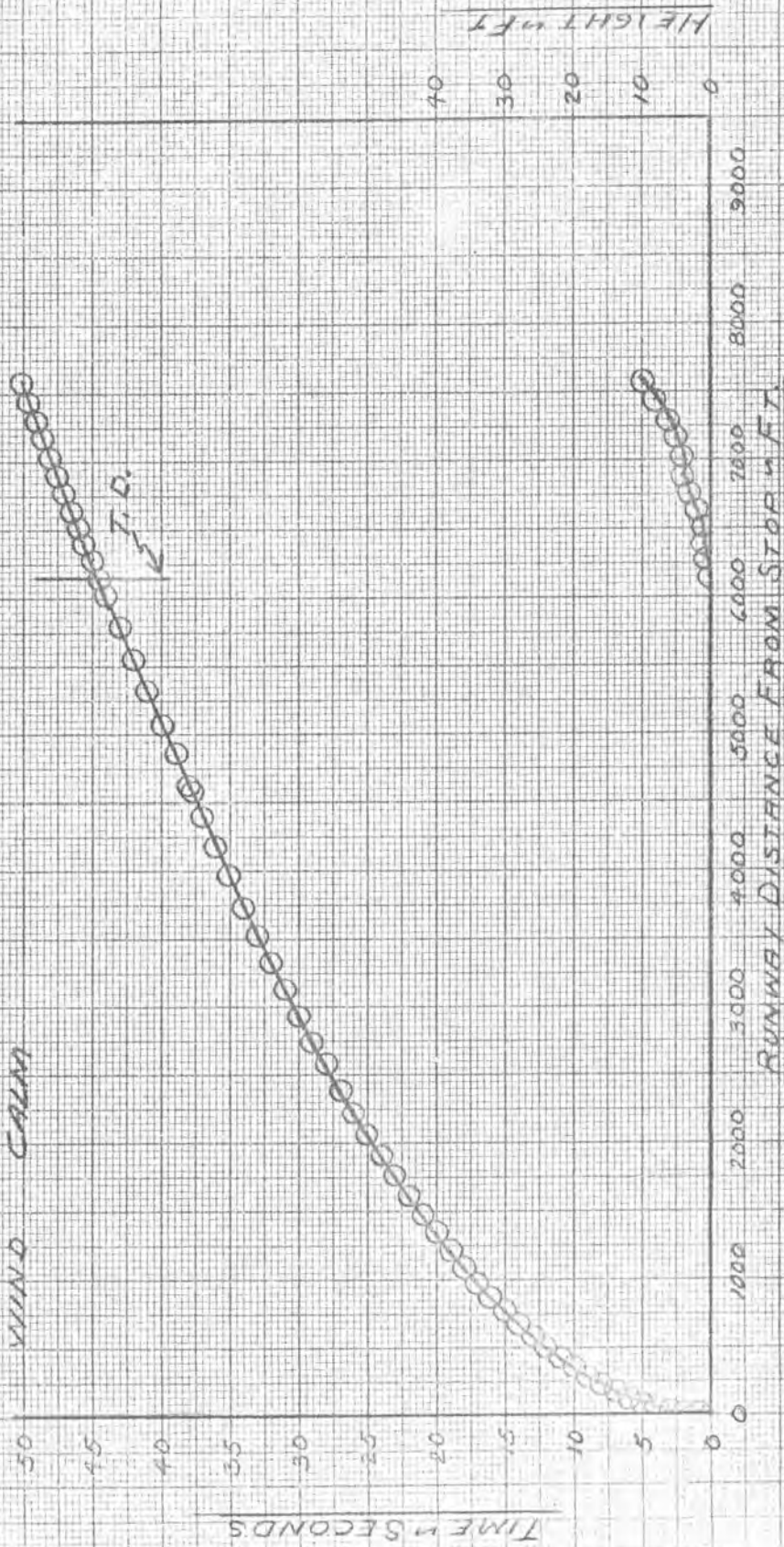
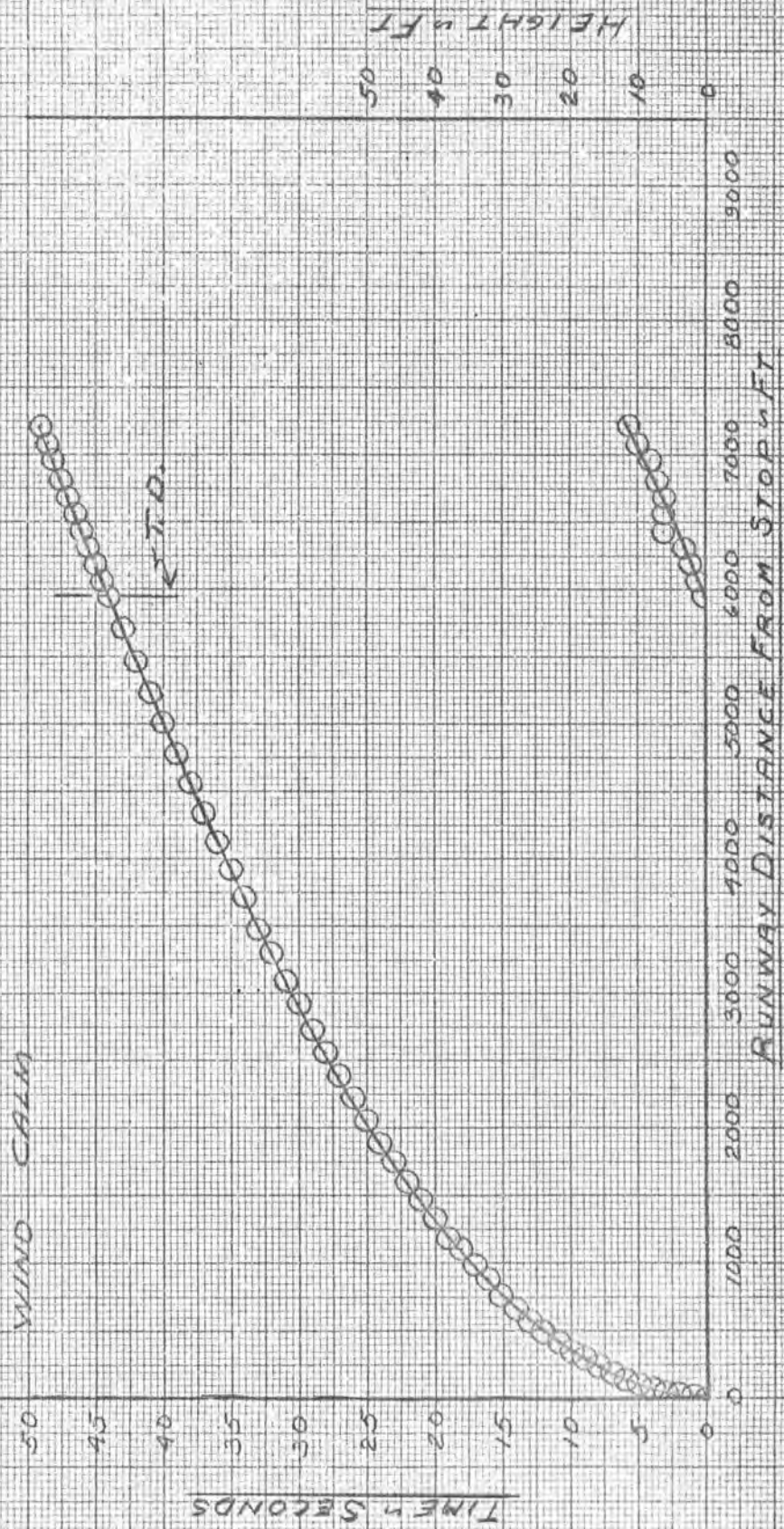


FIGURE No 13
LANDING
YF-100A USAF No 52-5754

FLT NO	14(2)	IAS @ TD	142	KNOTS
GROSS WT	21,100 LBS	N ₂	EDLE	RPM
CG	30.7 %MAC	V ₅ @ TD	142	KNOTS
PRESS ALT	2,280 FT	DRAG CHUTE DEPLOYED	M	NO
IAS @ 50'	NO DATA	FAT		36.0C

WIND CALM



DISTRIBUTION

	No. of cys
Hq, USAF; Washington 25, D. C.	
Attn: Director of Requirements, DCS/O	1
Attn: Asst for Atomic Energy; DCS/O	1
Attn: Asst for Programming; DCS/O	1
Attn: Directorate of Operations; DCS/O	4
Attn: Directorate of Procurement and Production Engineering; Prod. Engineering Division	1
Attn: Directorate of Maintenance Engineering	2
Attn: Aircraft Division	1
Attn: Directorate of Intelligence, DCS/O	
Attn: AFOIN -2B3	1
Attn: AFOIN-2C1	1
Hq. AMC; Wright-Patterson AFB; Ohio	
Attn: MCMRM, Materiel and Flight Safety Deficiency Branch	1
Attn: MCSRB, Directorate of Supply and Services	1
Attn: MXMZ, Programs Monitoring Office Supply, M and S Directorate	1
Attn: MCPBXA, Administration Office, Industrial Resources Division	1
Attn: MCPPA, Aircraft Branch, Procurement Div.	2
Attn: MCPPE, Aeronautical Equipment Section, Procurement Division	1
Attn: MCPPRT, Technical Office, Research and Dev- elopment Branch, Procurement Division	1
Attn: MCQA, Quality Control Section	1
Hq, WADC, Wright-Patterson AFB, Ohio	
Attn: WCY, Assistant Chief of Staff	1
Attn: WCRO, Operations and Plans Office	1
Attn: WCOL-1, Major R. L. Northup	2
Attn: WCS, Weapons System Division	2
Attn: WCOSI, Technical Info Intelligence Branch	1
Attn: WCOST, Test Requirements Branch	4
Attn: WCTE, Engineering Branch	3
Attn: WCLEI-3, Instrumentation Branch, Equipment Laboratory	1
Attn: WCOF, Flight Safety Office	3
Attn: WCLPO, Operations Office (Power Plant Lab)	2
Attn: WCLBO, Plans and Operations Office	2
Attn: WCSF, Fighter Aircraft Branch	2
Attn: WCLSO, Operations Office (Aircraft Lab)	2
Attn: WCOSF, Flight Data Branch	2
Attn: WCLSR, Aero Branch, Aircraft Laboratory	2

(Distribution Con't)

	No. of cys
Hq, ARDC, P. O. Box 1395; Baltimore 3, Maryland	
Attn: RDDAS	3
Attn: RDOS	1
Attn: RDSSD, Capt. Lucas	1
Attn: RDBR, Lt. Col. Diehl	1
Director, Air University Library, Maxwell AFB, Alabama	1
Commandant, USAF Institute of Technology, Wright-Patterson AFB, Dayton, Ohio	1
Armed Services Technical Information Agency, Document Service Center, Knott Building, 4th and Main Street, Dayton 2, Ohio	6
Office of the Inspector General, USAF, Norton AFB, San Bernardino, California	3
Chief, Air Technical Intelligence Center, Wright-Patterson AFB, Dayton, Ohio	1
Bureau of Aeronautics, Department of the Navy, Washington 25, D. C.	2
Comdr, U. S. Naval Air Material Center, Philadelphia, Pa.	1
Comdr, U. S. Naval Air Test Center, Patuxent River, Md.	1
North American Aviation Inc., International Airport, Los Angeles 45, California. Attn: AFPR	1
Attn: Geo. Mellinger	1
North American Aviation Inc., Edwards Air Force Base, Edwards, Calif.	
Attn: Murray O'Toole	1
Lt. Col. G. M. Townsend, % Air Force Plant Representative Boeing Airplane Co., Seattle, Washington	1
Hq, AFFTC	
Attn: FTT,	1
Attn: FTDTP	①
Attn: Alfred D. Phillips, Project Engineer	①
Attn: FTDA	5
Attn: FTAH, Base Historian	1
Attn: FTOT, Technical Information Office	1
Attn: FTDF	1
Attn: FTDF, Lt. Col. Frank K. Everest	1

(Distribution Con't)

No. of cys

Hq, USAF, Attn: Aircraft Division, Washington 25, D.C. DCS/D	1
Comdr, Air Technical Intelligence Center; ATTN: AFOIN- ATISD -1B; Wright-Patterson AFB, Dayton, Ohio	1
Hq, 4925th Test Group (Atomic); Kirtland AFB, New Mexico	1
NACA; Attn: Office of Aeronautical Intelligence 1724 "F" street, N. W. Washington 25, D. C.	5
NACA; Attn: Mr. Walter C. Williams, High Speed Flight Research Station; P. O. Box 273, Edwards, Calif.	1
Comdr, Alaskan Air Command, APO 942, C/O Postmaster, Seattle, Washington	1
AF Engineering Field Representative at the NACA, Ames Aero- nautical Laboratory, Moffett Field, California	1
Comdr, Air Proving Ground Command, Eglin AFB, Florida Attn: Technical Reports Branch, AAG	2
Comdr, MATS, Andrews AFB, Washington 25, D.C. Attn: Maintenance Engineering Div., Technical Publications	1
Washington Air Force Development Field Office, Room 4945 Main Navy Bldg., Washington 25, D. C.	2

~~SECRET~~

~~SECRET~~

~~SECRET~~

~~SECRET~~

~~SECRET~~

~~SECRET~~



~~SECRET~~

~~SECRET~~