NACA

RESEARCH MEMORANDUM

ALTITUDE OPERATIONAL CHARACTERISTICS OF A PROTOTYPE MODEL OF THE J47D (RX1-1 AND RX1-3) TURBOJET ENGINES WITH INTEGRATED ELECTRONIC CONTROL

By E. William Conrad, Harry E. Bloomer, and Adam E. Sobolewski

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

January 8, 1952
Declassified October 31, 1958

2130

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

ALTITUDE OPERATIONAL CHARACTERISTICS OF A PROTOTYPE MODEL OF THE J47D (RX1-1 AND RX1-3) TURBOJET ENGINES WITH INTEGRATED ELECTRONIC CONTROL

By E. William Conrad, Harry E. Bloomer, and Adam E. Sobolewski

SUMMARY

The altitude operational characteristics of a prototype model of the J47D (RX1-1 and RX1-3) turbojet engines, which includes an afterburner, a variable-area exhaust nozzle, and an integrated electronic control, were investigated in the NACA Lewis altitude wind tunnel at altitudes from 10,000 to 55,000 feet at a flight Mach number of 0.19 and at flight Mach numbers from 0.19 to 0.89 at an altitude of 25,000 feet. Data obtained with oscillograph recorders and conventional instrumentation are presented to show the following characteristics:

(a) Compressor stall

(b) Combustor blow-out

(c) Acceleration

(d) Deceleration

(e) Altitude starting

(f) Afterburner ignition

(g) Afterburner transients

(h) Afterburner blow-out

For both of the engines investigated (RX1-1 and RX1-3), it was found that the compressor-stall data plotted as single curves on coordinates of compressor pressure ratio and corrected engine speed. Two distinct types of stall appear to exist with the transition occurring at corrected engine speeds between 5250 and 6250 rpm. The compressor unstall characteristics are shown on the same coordinates.

For corrected engine speeds above 5800 rpm, the pressure ratio necessary to unstall the compressor was lower than the pressure ratio for either stall or steady-state operation. As the altitude was increased, the compressor unstalled at slightly higher pressure ratios; however, flight Mach number had no apparent effect within the range investigated. Combustor blow-out data for all flight conditions investigated with the RX1-1 engine were plotted as a single curve on the same coordinates and coincided almost exactly with the compressor stall curve.

Acceleration time from the idle to the rated thrust condition without afterburning increased from 14 to 22 seconds as the altitude was increased from 15,000 to 45,000 feet at a flight Mach number of 0.19. At an altitude of 25,000 feet, an increase in flight Mach number from 0.19 to 0.75 reduced the acceleration time from idle to rated thrust without afterburning from 14.4 to 6.0 seconds. When a minimum fuel limit of 450 pounds per hour was used, lean combustor blow-out could not be obtained during deceleration for the range of flight conditions covered by the investigation.

Ignition in all combustors was obtained at windmilling speeds from 1300 to 1500 rpm at an altitude of 50,000 feet using MIL-F-5624 (AN-F-58) fuel at a temperature of about 70° F and inlet-air temperatures of 0° to -6° F. At 35,000 feet, ignition occurred in all combustors up to 3500 rpm, the highest windmilling speed obtainable. With the use of MIL-F-5624 fuel (treated to give a 1-pound Reid vapor pressure) at a temperature of approximately 90° F and inlet-air temperatures from -20° to 30° F, ignition was possible in all combustors at 49,000 feet up to a windmilling speed of about 1500 rpm. At 25,000 feet, however, starts were possible up to 2200 rpm. At an altitude of 40,000 feet, the optimum fuel flow for starting appeared to be about 650 pounds per hour for MIL-F-5624 fuel with a 7-pound vapor pressure.

Afterburner starts by autoignition were obtained at altitudes up to 53,000 feet at a flight Mach number of 0.19 using MIL-F-5624 fuel. The tail-pipe fuel-air ratios required for autoignition increased with altitude and at 35,000 feet decreased as the burner-inlet temperature was raised. The tail-pipe fuel-air ratio at which lean blow-out of the afterburner occurred was increased as altitude was raised. The width of the blow-out band also increased with altitude.

2180

INTRODUCTION

The improvement in performance characteristics of turbojet engines effected by the application of afterburning and a continuously variable-area exhaust nozzle have been well established during the past few years. Manual control of a turbojet engine equipped with an afterburner and variable-area exhaust nozzle would, however, place a heavy burden on the pilot or flight engineer. To relieve the pilot of the extreme complexity of engine operation and the need for constant surveillance of many engine operation limits, a completely automatic control system is required.

Accordingly, a prototype model of the J47D (RX1-1 and RX1-3) turbojet engines was provided with an afterburner, a variable-area nozzle and an integrated electronic control. The engine was installed in the NACA Lewis altitude wind tunnel to obtain the performance characteristics and insight into the engine's operational problems. The steady-state performance of the engine without afterburning is presented in references 1 and 2. The operational characteristics are presented herein.

The integrated electronic control used with the prototype model of the J47D (RX1-1 and RX1-3) turbojet engines was designed to (1) provide single-lever thrust control over the full range of operation from starting to full afterburning condition, (2) schedule all services required, and (3) give the maximum acceleration and deceleration rates possible without exceeding engine speed and temperature limits or causing combustor blow-out or compressor stall. To attain the latter objective, the compressor stall and combustor blow-out regions were investigated with the control inoperative. The limits maintained by the control were then adjusted as necessary and the transient performance of the controlled engine was evaluated.

Oscillograph traces are presented herein to show the typical behavior of these variables during compressor stall, complete and partial combustor blow-out during acceleration, controlled acceleration and deceleration, and afterburner ignition. Compressor stall and combustor blow-out limits have been correlated to show the effects of altitude, flight Mach number and corrected engine speed. The effects of altitude and flight Mach number on engine acceleration and deceleration are also given, as well as steady-state operational characteristics such as engine and afterburner ignition and afterburner lean blow-out. Data were obtained at altitudes from 10,000 to 55,000 feet at a flight Mach number of 0.19 and at flight Mach numbers from 0.19 to 0.89 at an altitude of 25,000 feet.

DESCRIPTION OF ENGINE

The J47D RX1-1 and RX1-3 engines used in this investigation were aerodynamically the same as the J47D engine. The manufacturer's guaranteed static sea-level performance of the J47D engine is 5670 pounds thrust at 7950 rpm and an exhaust-gas temperature of 1275° F. The main components include a 12-stage axial-flow compressor with a pressure ratio of about 5.1 at rated conditions, eight cylindrical direct-flow combustors, a single-stage turbine, a diffuser, an afterburner combustion chamber, a variable-area exhaust nozzle, and an integrated electronic control. The over-all length of the engine including the afterburner is about 217 inches, the maximum diameter is approximately 37 inches, and the total weight is about 3000 pounds. A view of the turbojet engine installed in the test section of the altitude wind tunnel is given in figure 1 and a schematic drawing of the engine is presented in figure 2.

Two combustor configurations were used with the RX1-1 engine. The original configuration included conventional-type spark plugs, a 20,000-volt coil and vibrator unit, and cross-fire tubes $1\frac{3}{16}$ inches in diameter. The second configuration, referred to as the "modified combustor," had cross-fire tubes that were 2 inches in diameter. The modified combustor liners also had semicylindrical shrouds projecting 1/4 inch radially inward from the downstream half of each of the first six rows of holes. These changes, as well as other minor differences between the original and modified combustor liners are shown in figure 3. The ignition systems used with the modified configuration were (1) opposite polarity spark plugs mounted as shown in figure 4 and (2) an ignition system which produced a potential difference approximately twice that of the standard system. Only the modified configuration was used with the RX1-3 engine. For both combustor configurations, two sets of spark plugs, located in diametrically opposite combustors, were used.

The afterburner shown schematically in figure 2 was comprised of a diffuser 43 inches in length, a combustion chamber 50 inches in length which tapered from a $32\frac{1}{8}$-inch diameter at the flame holder to a 29-inch diameter at the exhaust-nozzle section, and a variable-area exhaust nozzle which was 16 inches in length in the open position. Flame-holding surfaces were provided with 2-ring, V-gutter flame holders used in conjunction with a $13\frac{1}{2}$-inch diameter pilot cone. The location of the flame holders was varied from 2 to 7 inches downstream of the pilot cone. Fuel was injected by means of 24 radial-spray bars (two sets of 12) located $9\frac{3}{4}$ and 13 inches upstream of the pilot cone.

2180

INTEGRATED ELECTRONIC CONTROL

The function of the integrated electronic control system is to cause the engine to operate at an optimum point determined from performance, operational, and safety considerations at any given operating condition. A comprehensive study of the control is given in reference 3. A block diagram from this reference is given in figure 5 to show the relation of the control components. Detailed functions, some of which are discussed more fully in appendix A, may be listed as follows:

1. Maintain a set engine speed irrespective of changes in altitude or flight Mach number and maintain rated engine speed within close limits during steady-state operation

2. Prevent serious overshoot above rated engine speed as a result of transients

3. Prevent serious loss of engine speed when ignition occurs in afterburner

4. Maintain rated turbine-outlet temperature (1275° F) at the rated thrust position, irrespective of flight conditions

5. Prevent excessive turbine-outlet temperatures during afterburning operation when the exhaust nozzle is fully open or if the exhaust nozzle locks in a partially open position

6. Provide a minimum idle speed at a value above the blow-out speed and at a value from which satisfactory accelerations can be made

7. Permit the most rapid accelerations possible without combustor blow-out, compressor stall, or exceeding temperatures above the transient limit

8. Give near optimum specific fuel consumptions at the steady-state conditions

9. Permit the maximum rate of deceleration possible without combustor blow-out

10. Provide a schedule of fuel flows during the starting cycle to avoid overtemperature

11. Schedule the necessary services, such as cooling air to the exhaust nozzle, before afterburning begins

12. Fail safely so that failure of any part of the control system will permit engine control to continue uninterrupted under the remaining components of the main system or emergency system

Engine speed, exhaust-nozzle area, and afterburner fuel flow during steady-state operation are scheduled as functions of thrust-selector position, as shown in figure 6. Exhaust-nozzle area (in the nonafterburning region only) and afterburner fuel flow are scheduled by potentiometers linked directly to the thrust selector lever. In the afterburning region, the exhaust nozzle is adjusted by the control to maintain limiting turbine-outlet temperature. Although the area is affected by flight condition, a typical curve of exhaust-nozzle area is given by the broken line. Engine speed is governed by changes in fuel flow. The relation between engine speed and exhaust-nozzle area during steady-state operation is given in figure 7. It will be noted that a large reduction in exhaust-nozzle area occurs at rated engine speed as the thrust selector changes from 70° to 90°. This large change in exhaust-nozzle area was used to permit a relatively large thrust change to be made very rapidly without the necessity of changing engine speed.

During large accelerations, the exhaust nozzle remains in the initial position until the approximate final speed or until a speed of 7800 rpm is reached, whereupon it returns to the steady-state schedule. A maximum fuel limit, scheduled as a function of the compressor-outlet pressure, is imposed during accelerations to prevent both compressor stall and combustor blow-out. A transient temperature limit of about 1500° F at the turbine outlet is also imposed; excessive temperatures will cause the fuel valve to close.

INSTALLATION

The engine was mounted on a wing spanning the test section of the altitude wind tunnel. Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct connected to the engine inlet (fig. 1). Manually controlled butterfly valves in this duct were used to adjust total air pressures at the engine inlet. A slip joint with a frictionless seal was used in the duct, thereby making possible the measurement of thrust and installation drag with the tunnel scales and with strain gages on the engine supports.

Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (fig. 2) to determine the steady-state performance and to calibrate the transient performance data. Instrumentation (dynamic response and design features of the transient instrumentation are given in reference 4) for measuring transient performance was also installed, as given in the following table:

Measured quantity	Transient instrumentation (calibrated from steady-state instrumentation)			Steady-state instrumentation
	Sensor	Recorder	Dynamic lag (equivalent time constant) (sec)	Instrument
Engine speed	Tachometer generator, direct current	Multiple-channel direct-inking oscillograph with associated amplifiers	0.04	Tachometer generator, alternate current
Compressor-outlet pressure	Aneroid-type pressure sensor		0.02	Bourdon-type gage
Tail-pipe temperature (station 6)	Unshielded loop thermocouple		0.25 at average sea-level mass flow	Six thermocouples in parallel on self-balancing Brown potentiometer
Engine mount force (function of jet thrust)	Strain gage on main engine support		Less than 0.002	
Engine fuel-valve position (also reheat fuel-valve position)	Wire-wound potentiometer		Less than 0.002	Selsyns
Exhaust-nozzle area	Wire-wound potentiometer		Less than 0.002	Selsyns
Ram pressure ratio	Aneroid-type pressure sensor		0.02	Bourdon-type gage

2180

PROCEDURE

The air flow through the make-up air duct was throttled from approximately sea-level pressure to a total pressure at the engine inlet corresponding to the desired flight Mach number at a given altitude. A list of symbols is given in appendix B. Inasmuch as the throttling valves were manually controlled, it was impossible to maintain constant engine-inlet pressure during all transients, particularly at high altitudes where the valves were almost closed; however, an attempt was made to maintain the desired pressure as nearly as possible. The static pressure in the tunnel test section was maintained to correspond to the desired altitude. Because of the large tunnel volume, the tunnel test-section pressure did not vary appreciably during transients. The engine-inlet-air temperatures were held at approximately NACA standard values corresponding to the simulated flight conditions, except for high altitudes and low flight Mach numbers. No inlet-air temperatures below -20° F were obtained. In addition, three data points were obtained at an inlet-air temperature of 140° F.

Compressor stall and combustor blow-out during accelerations were investigated by imposing step-function increases in fuel flow to the engine with the control inoperative. Successively larger steps were used until either stall or blow-out occurred. In general, after a stall was obtained, the fuel flow was reduced, permitting the compressor to unstall. Stall data were obtained in the range of corrected engine speeds from approximately 4000 to 8000 rpm. Combustor blow-out data during accelerations were obtained at corrected engine speeds from about 5000 to 7800 rpm. This phase of the investigation covered a range of altitudes from 10,000 to 55,000 feet and flight Mach numbers from 0.19 to 0.89.

With the exception of a brief study to determine the fuel flows required for starting at an altitude of 40,000 feet, the remainder of the investigation discussed herein was conducted with the control in operation. The terms "throttle burst" and "throttle chop" are used to denote accelerations and decelerations, respectively, wherein the thrust-selector position was changed as rapidly as possible. In addition to these data, several runs were made wherein the throttle was first chopped and then a throttle burst was made while the engine speed was decreasing rapidly.

A large number of starts were attempted with the control operative over a wide range of windmilling speeds at altitudes from 25,000 to 45,000 feet using MIL-F-5624 fuels having Reid vapor pressures of 1 and 7 pounds. These starting attempts were made by setting windmilling speed at the desired value and then advancing the thrust selector to the idle range (from 10° to 20°). Whether the control was operative or inoperative during starts, if ignition was not obtained within 30 to 40 seconds, the attempt was considered unsuccessful.

2180

Afterburner ignition and blow-out limits were obtained during the normal course of the investigation of afterburner performance. As the afterburner fuel flow was gradually increased, the fuel flow at which autoignition occurred was noted and similarly the lean blow-out data were obtained as the afterburner fuel flow was gradually decreased prior to shutdown.

RESULTS AND DISCUSSION

Many of the results to be discussed are in the form of oscillograph traces. To familiarize the reader with the typical behavior of several important engine variables, an oscillograph record is given in figure 8 for a throttle-burst acceleration of the controlled engine from idle to full dry thrust. Arrows are shown with each trace to indicate the direction of increase of each variable. The fuel flow increased almost instantaneously to the maximum fuel limit imposed by the control at point A and thereafter followed the maximum fuel-limit curve until, at point B, the flow was reduced by the control because of a progressive reduction in the transient temperature limit with engine speed after an engine speed of about 7200 rpm was reached. This limit, generally set at 1300° F, was set at 1250° F during the transient shown. Turbine-outlet temperature followed the trend of the fuel flow and compressor-outlet pressure was affected by both fuel flow and engine speed. Engine speed increased at an almost uniform rate until the fuel flow was reduced at point B, after which the rate of acceleration was reduced. No appreciable speed overshoot occurred. The exhaust-nozzle area remained locked in the initial open position until a speed of about 7700 rpm was reached. The nozzle then closed to the final position in about 0.9 second. As mentioned previously, the ram pressure ratio could not be held constant during transients of this magnitude and rapidity. As will be shown, the effect of a decrease in ram pressure ratio on the engine during an acceleration is an increase in the acceleration time as compared to the time taken with a constant ram pressure ratio. In general, variations in ram pressure ratio of the magnitude encountered in this investigation had little effect on the operational characteristics of the engine; however, this variation was taken into account in all correlations of the stall, unstall, and combustor blow-out data.

2180

Compressor Stall

Compressor stall in turbojet engines is usually encountered when excessively fast accelerations are attempted, although some engines encounter stall within the normal steady-state operating region. Compressor stall in a turbojet engine is characterized by a sudden reduction and severe fluctuation of pressure throughout the engine, a decrease of air flow, and excessively high turbine-outlet temperatures. Acceleration of a turbojet engine requires that the enthalpy drop through the turbine be increased to exceed the enthalpy rise through the compressor, thereby providing the excess power required for acceleration. This is, of course, accomplished by increasing the fuel flow and consequently the turbine-inlet temperature. Inasmuch as the compressor pressure ratio is a function of the turbine-inlet temperature, the compressor-outlet pressure immediately increases to some value above the steady-state value at the start of an acceleration and, unless stall occurs, remains higher throughout most of the transient. The pressure ratio which can be tolerated by the compressor without flow breakdown (stall) is limited by the effects of the increased adverse pressure gradient on boundary-layer flow and by the angle of attack of the blades. As a result, the rate of acceleration of a turbojet engine is limited by the stall characteristics of the compresser.

As explained under PROCEDURE, progressively larger steps in fuel flow were made with the control inoperative until stall was encountered. Two such runs are shown in figure 9. In the first run, (fig. 9(a)) the fuel flow was increased from 800 to 3900 pounds per hour, producing a smooth acceleration until the fuel valve was retracted to prevent overspeed of the uncontrolled engine. In the second run (fig. 9(b)), a slightly larger fuel-valve step from 800 to 4150 pounds per hour was used, resulting in compressor stall. The effects of stall may be clearly seen by comparing the traces of figure 9(a) with figure 9(b). The stall is evidenced by the sudden reduction (point A) followed by a rapid fluctuation in the compressor-outlet pressure trace, by a large increase in the turbine-outlet temperature, and by a break in the slope of the engine-speed trace. It will be noted that the engine speed continued to increase during the period of stall; however, the acceleration could not be completed because of the high turbine-outlet temperature (2160° F). The manner in which stall affects engine air flow is indicated by the traces of ram pressure. In the first run, the increase in air flow associated with the engine acceleration decreased the ram pressure until manual adjustment by the tunnel operator could be made. In the second run, a sudden increase in ram pressure occurred at the stall point, indicating a marked reduction in air flow. Analysis of other traces not shown indicates that during stalled operation the air flow gradually decreases despite the fact that engine speed slowly increases. A few runs were made with a chart speed ten times as fast as the traces shown. These data show that the frequency of the compressor-outlet pressure pulsations was from 37 to 55 cycles per second, however, the response of the instruments was not sufficiently fast to permit accurate determination of the amplitude. It should be noted that the compressor unstalled at point B following a decrease in fuel flow and temperature.

The compressor stall curve for the J47D (RX1-3) engine is given in figure 10(a) for altitudes from 10,000 to 35,000 feet at a flight Mach number of 0.19, for flight Mach numbers up to 0.89 at 25,000 feet, and for high (140° F) inlet-air temperatures at 15,000 feet. Inasmuch as air-flow measurements were not available during transients, corrected engine speed is used as the abscissa in figure 10 instead of corrected air flow. It should be noted that the compressor stall data reduce to a single curve for the entire range of flight conditions investigated. A limited amount of data were also obtained at an altitude of 45,000 feet but are not included in figure 10(a) because data for correction of the variation in ram pressure ratio were unavailable. When considered on an uncorrected basis, however, these data for an altitude of 45,000 feet correlated to a single curve with the data at lower altitudes. The stall-limit curve appears to be divided into two distinct segments, with a transition occurring at corrected engine speeds between 5250 and 5500 rpm. Similar data for the RX1-1 engine (fig. 10(b)) corroborate the trends shown in figure 10(a), however, the transition occurs at slightly higher corrected engine speeds. Although the compressor design was the same for the two engines, there may have been slight differences in the tip clearance of the various stages because of manufacturing tolerances. Various types of compressor stall are discussed in reference 5 and explained in terms of the blade velocity triangles. According to this reference, the stall at speeds above 5500 rpm is probably due to excessive angle of attack (positive stall) of the latter compressor stage whereas at lower speeds it is due to positive stall of the early stages.

Compressor Unstall

The measures necessary to unstall a compressor are of interest and accordingly the unstall data have been correlated on the same coordinates as the stall data. It might be expected that a slight reduction in compressor pressure ratio below the stall limit would cause the compressor to unstall; however, from figure 9 it may be seen that during the stall the pressure ratio dropped far below the stall limit and the compressor did not unstall. A furthur reduction in pressure ratio obtained by a reduction in fuel flow was required. Unstall characteristics of the RX1-3 engine are given in figure 11 to show the effects of altitude at a flight Mach number of 0.19 and engine-inlet temperature at 15,000 feet. As the altitude increased from 10,000 to 25,000 feet, the compressor unstalled at slightly higher pressure ratios. Although the amount of data at 15,000 feet with engine-inlet temperature of 140° F is limited, two of the three data points indicate no effect of engine-inlet temperature on the unstall characteristic of the engine. Similar data in figure 12 show that changing the flight Mach number from 0.19 to 0.89 at an altitude of 25,000 feet had no effect on the unstall characteristics. At high corrected engine speeds, the compressor unstalled at lower pressure ratios than were encountered at stall or during steady-state operation. This is shown in figure 13, where both the stall and unstall data are represented by single curves and compared with the steady-state region of operation. The shaded area represents the maximum region of steady-state operation possible by variations in exhaust-nozzle area or flight condition. Inasmuch as the unstall curve lies below this region over most of the practical range of engine speeds, it is obvious that the compressor cannot in general be unstalled by opening the exhaust nozzle and that a reduction of fuel flow is usually required.

2130

The distance between the operating line (which lies somewhere within the shaded region) and the stall-limit curve is indicative of the margin of excess power available for acceleration. It will be noted that this margin increases abruptly as the corrected engine speed is raised above 5250 rpm. The path of a typical throttle-burst acceleration with the control operative to restrict operation to the region below the stall limit is denoted by a broken line. The rapid decrease in compressor pressure ratio at 8000 rpm is due to a reduction in fuel flow called for by the control to prevent exceeding the transient temperature limit.

Combustor Blow-Out During Acceleration

In turbojet-engine operation, combustor blow-out during transients, like compressor stall, is usually encountered during rapid acceleration. It is likely that the blow-out during acceleration is caused by excessively rich regions in the primary zone, resulting from the sudden increase in fuel flow and a reduction in air flow associated with the negative slope of compressor characteristic curves at high pressure ratios. For the engines investigated and for a flight Mach number of 0.19, stall was prevalent at altitudes below 35,000 feet and blow-out was prevalent at higher altitudes; however, a change in compressor design, combustor design, or flight Mach number would probably shift this transition altitude. A few stalls (not shown) were encountered at 45,000 feet, and occasionally stalls at 35,000 feet were immediately followed by blow-out.

Oscillograph traces of two runs are given in figure 14 to permit comparison of a successful acceleration and an attempted acceleration using a slightly larger step increase in fuel flow which resulted in combustor blow-out. The blow-out point is obvious on both turbine-outlet temperature and compressor-outlet pressure traces (fig. 14(b)). During the first run (fig. 14(a)), a turbine-outlet temperature of 1300° F was obtained, however, during the second run, (fig. 14(b)) blow-out occurred at 1320° F. The fuel flow was reduced manually shortly after the blow-out was obtained. Engine speed and turbine-outlet temperature decreased very rapidly and went below the limit of pen travel on the recorder. Because of the reduction in turbine-inlet temperature and the flow conditions in the turbine nozzle diaphragm, the compressor-outlet pressure decreased markedly at the blow-out point and thereafter decreased slowly as the engine speed decreased. Because of differences in instrument response time, the compressor-outlet pressure trace indicated the blow-out approximately 0.1 second before the turbine-outlet temperature trace.

On several occasions, combustor blow-out appeared to be incomplete. Oscillograph traces of one such case of partial blow-out are given in figure 15. Following the partial blow-out, the average turbine-outlet temperature decreased rapidly from 1254° F to about 830° F and then increased about 30° F in about 6.5 seconds during which time the fuel flow was constant. Approximately 1/2 second after the fuel flow was reduced and about 9 seconds after the partial blow-out, full combustion was restored and the average turbine-outlet temperature increased to 1160° F for about 4 seconds after which it decreased toward a value commensurate with the fuel-air ratio. It will be noted that complete combustion resumed at approximately the same fuel flow as that required for steady-state operation. Motion-picture records of individual thermocouples located just downstream of the turbine behind each combustor revealed that during the partial blow-out, temperatures behind three combustors became very high whereas the temperatures behind the remaining five were between 500° and 600° F. The phenomenon occurring in five of the combustors providing a low temperature rise is believed to be due to cessation of flame propagation away from the recirculating region in the combustor primary zone, which is due to an excessive fuel-air ratio, and the resumption of flame propagation following the restoration of a more favorable fuel-air ratio. The inordinate temperature value following complete combustion is probably the result of fuel

2180

accumulated in the combustors during the period of partial blow-out. It should be noted that complete combustion was restored at an engine speed of about 5800 rpm in contrast to a maximum windmilling speed for ignition (to be discussed later) of about 2000 rpm. From these observations it may be concluded that some cases of apparent combustor blow-out during rapid accelerations are not actually a complete failure of combustion but instead the occurrence of a combustion process in which only part of the fuel is burned. In this event, complete combustion may be restored by a gradual throttle retraction which will reduce the fuel-air ratio to more favorable values.

2180

Another interesting point is shown by the data of figure 15. Immediately following the fuel step, a quenching action occurred which reduced the turbine-outlet temperature as much as 110° F. This quenching action occurred over a period of more than 1 second and was of sufficient magnitude to cause a reduction of 0.9 inch of mercury in compressor-outlet pressure and 50 rpm in engine speed. Inasmuch as this effect becomes more pronounced as altitude is increased and engine speed decreased, the flame may well be quenched completely at some flight conditions following a rapid increase in fuel flow.

Maximum values of fuel flow which may be used without causing combustor blow-out are defined by the data of figure 16 as a function of corrected initial engine speed for both original and modified combustor configurations and a flight Mach number of 0.19 at an altitude of 45,000 feet. Steady-state fuel-flow requirements and the margin of fuel flow available for acceleration are indicated. It will be noted that the margin for acceleration was not greatly different for the two configurations and varied from about 100 percent of the steady-state requirement at a corrected engine speed of 5500 rpm to 145 percent at 6900 rpm. At 8200 rpm, the margin was 69 percent for the original combustors and 65 percent for the modified combustors.

Stall and Blow-Out Protection

It has been shown here and elsewhere (reference 5) that compressor stall is a function of compressor pressure ratio and either engine speed or air flow. Also, to a close approximation, air flow is directly proportional to the compressor-outlet pressure and inversely proportional to the turbine-inlet temperature. By relating turbine-inlet temperature to fuel flow, air flow, and compressor-outlet temperature and in turn relating compressor-outlet temperature to pressure ratio and inlet-air temperature, relations may be derived (appendix C) which may be used to provide protection against both compressor stall and combustor blow-out using only two measurements, compressor-outlet pressure and engine-inlet-air temperature. The manufacturer designed the control accordingly to prevent stall or blow-out by limiting the maximum fuel flow for any given value of compressor-outlet pressure corrected for inlet-air temperature. The data obtained at 140° F inlet-air temperature (fig. 10) indicate, however, that the correction for inlet-air-temperature variations can be neglected.

2180

Because of the uncertainty of the assumptions made in the derivations, it was necessary to establish experimentally the absolute values of the maximum fuel-limit curve. This was done by plotting fuel flow against compressor-outlet pressure for all stall and blow-out points for which data were available. These data are given in figure 17 for both engines and both combustor configurations. The highest permissible setting of the maximum fuel-limit curve is also shown as well as a steady-state operating line and the path of a typical throttle-burst acceleration. Although the stall and blow-out data correlate reasonably well, the maximum permissible fuel flow appears to increase with flight Mach number. The data are not sufficient to establish this trend definitely; however, most of the data indicate that at an altitude of 25,000 feet and a flight Mach number of 0.75 the margin available for acceleration could be increased perhaps 35 percent over that obtained using the setting of the maximum fuel limit required to prevent stall at a Mach number of 0.19. Much of the scatter shown is probably due to variations in component efficiencies which were assumed constant in the derivations. Although meager, the data obtained at an inlet-air temperature of 140° F at 15,000 feet indicate no apparent trend with inlet temperatures. Also, the data show no difference between combustor configurations (denoted by different symbols) or between the two engines.

By making adjustments to the control, it was possible to shift the position of the maximum fuel-limit curve, and considerable effort was devoted to obtaining the optimum position. The position of the curve for the acceleration shown is given by the line BCD, the acceleration starting at point A. The fuel flow increased immediately from point A to the limit curve, followed the limit curve BCD as the compressor-outlet pressure increased with engine speed, and then was reduced suddenly at D because the turbine-outlet temperature reached the transient limit value. Excessively rapid action of the fuel valve in response to the overtemperature signal caused the fuel flow to undershoot the steady-state value in the region E before equilibrium running was reached at F.

Combustor blow-out data are commonly correlated in terms of the pressure, temperature, and velocity at the combustor inlet and the fuel-air ratio. Because of the relations discussed previously, combustor blow-out data may be correlated in terms of compressor pressure ratio and corrected engine speed. Such a correlation is shown in figure 18 for both original and modified combustor configurations. Compressor stall data, denoted by the solid symbols, are superimposed, and it will be noted that the stall and blow-out data fall around a single curve. An alternative method of providing protection against stall and blow-out is thus afforded by the use of this correlation. A comparison of the stall and blow-out protection correlations as established by the manufacturer's control (fig. 17) and the alternative method of protection derived from the analysis of the data (fig. 18) indicates that a better correlation is obtained with the alternative method. Consequently, with the alternative method instead of the manufacturer's method, more rapid acceleration of the engine can be made over a wider range of operating conditions. However, with the alternative method of protection, four measurements (compressor-inlet pressure, compressor-outlet pressure, engine speed, and inlet-air temperature) are required as compared with one (compressor-outlet pressure) used by the control.

Acceleration Characteristics

After the stall and blow-out regions had been determined, the maximum fuel-limit curve was adjusted to skirt the lower side of these regions, thereby permitting the maximum safe margin for acceleration. The acceleration and deceleration characteristics of the RX1-3 engine were then evaluated over a wide range of flight conditions and engine speeds. Results of this phase of the investigation are summarized in figures 19 to 23.

An oscillograph record of a typical throttle-burst acceleration from idle speed to full dry thrust is given in figure 19. Fuel flow increased to the maximum fuel limit in about 0.1 second and thereafter followed the maximum fuel limit as determined from compressor-outlet pressure (fig. 17) until the acceleration was almost completed. After an initial reduction due to the quenching effect, turbine-outlet temperature reflected the changes in fuel flow and increased rapidly for about 1 second after which it increased gradually to the final value of 1280° F. Compressor-outlet pressure decreased slightly and then increased at a more or less uniform rate for about 9 seconds after which the rate of acceleration was reduced. Engine speed decreased slightly due to the quenching effect and thereafter increased smoothly until rated engine speed was reached 16.5 seconds after the thrust selector was moved. The exhaust-nozzle area remained open until an engine speed of about 7800 rpm was reached. It then moved in about 0.8 second to the area required for limiting turbine-outlet temperature. It should be noted that the turbine-outlet temperature was not drastically affected by the large area change which occurred at approximately constant fuel flow. Following a small initial reduction, the thrust increased gradually until the exhaust nozzle closed. The decrease in nozzle area resulted in a rapid thrust increase. The final engine speed and thrust were attained approximately 16 and 21 seconds, respectively, after the thrust selector was moved.

Inasmuch as thrust is the variable of prime importance during accelerations, the succeeding figures will be discussed in terms of thrust acceleration time, which is defined as the time required to change the engine mount force (a function of jet thrust) from the initial to the final value where the final value is defined as the point where thrust becomes approximately constant or where it begins to vary about a mean line. The effect of altitude on the time required to change both thrust and engine speed is shown in figure 20 for throttle bursts from idle speed to full unaugmented thrust for altitudes from 15,000 to 45,000 feet at a flight Mach number of 0.19. Both engine speed (fig. 20(a)) and engine mount force (fig. 20(b)) are expressed as percent of the rated values at the flight condition under consideration. It should be noted that the control caused the idle engine speed to increase with altitude. The time required for increasing both thrust and engine speed from idle to rated conditions increased with altitude. The thrust acceleration time (fig. 20(b)) required varied from 14 seconds at an altitude of 15,000 feet to 22 seconds at 45,000 feet, a ratio of 1.57. If these accelerations had been started at the same engine speed or percent of rated thrust, the ratio of acceleration time would have been much larger. It will be noted that a very slight speed overshoot occurred at 15,000 and 25,000 feet, resulting in an immediate reduction in fuel flow and thrust. Both the speed overshoot and thrust reduction were of very short duration.

A trend, which becomes more apparent with increasing altitude, is exhibited by the engine-mount-force curve for 45,000 feet about 9 seconds after the start of the acceleration. The decrease in mount force shown was due to a change in ram pressure at the face of the engine resulting from increased sensitivity of the tunnel make-up air throttle valve at high altitudes where the valve is almost closed. If it had been possible to maintain constant ram pressure ratio, the engine mount force would probably have increased smoothly along the broken line.

Because of the high energy of the air at the engine inlet due to ram, and the corresponding variation in idle speed it would be expected that acceleration characteristics would be improved with

2180

an increase in flight Mach number. The data of figure 21 show this to be true. At an altitude of 25,000 feet, an increase in flight Mach number from 0.19 to 0.75 decreased the thrust acceleration time from idle to full unaugmented thrust from 14.4 to 6 seconds. The thrust reductions occurring after 3.5 seconds at a Mach number of 0.75 and after 13.4 seconds at a Mach number of 0.19 are due to reduction of fuel flow made by the control as a result of overtemperature in the first case and overspeed in the second.

Throttle-burst accelerations from various initial thrusts to rated unaugmented thrust are compared in figures 22(a) and 22(b) for altitudes of 15,000 and 45,000 feet at a flight Mach number of 0.19. For the throttle burst from 10° to 90° on the thrust selector at 15,000 feet (fig. 22(a)), acceleration was extremely slow below 30 percent of rated thrust and required 14 seconds. An acceleration from 42 percent to rated thrust required only 3 seconds, and only 1.8 seconds was required to change thrust from 69 percent to rated thrust. At 45,000 feet (fig. 22(b)), the time required varied directly with the increase in thrust. The very rapid thrust changes associated with closure of the exhaust nozzle are apparent, and also it will be noted that thrust overshoots occur which are as much as 15 percent of rated thrust.

2180

The effect of exhaust-nozzle area on acceleration time is given in figure 23 for altitudes of 15,000 and 45,000 feet at a flight Mach number of 0.19. For the accelerations at constant area, the exhaust nozzle was locked at a position giving limiting turbine-outlet temperature at rated engine speed, thereby simulating the performance of an engine equipped with a fixed-area exhaust nozzle. In view of the increase in turbine back pressure with the smaller fixed-area nozzle, and the corresponding increase in compressor pressure ratio and turbine work, it would be expected that acceleration characteristics would be penalized by a reduction in exhaust-nozzle area inasmuch as the fuel margin available for acceleration is limited. This was found to be the case at both 15,000- and 45,000-foot altitudes. At 15,000 feet, thrust acceleration time was 13.5 seconds for the variable-area nozzle as compared with 18 seconds for the fixed-area nozzle, a decrease of 25 percent. At 45,000 feet, rated thrust was obtained in 22 seconds with the variable-area nozzle as compared with 35 seconds for the fixed-area nozzle. From a tactical point of view, these acceleration times at 45,000 feet are misleading because during almost the entire time required for acceleration with the variable-area nozzle the thrust was higher with the fixed-area nozzle. It should be noted, however, that at 15,000 feet the variable-area nozzle was superior with respect to both thrust level and acceleration time.

The effect of an improper setting of the maximum fuel limit which will permit compressor stall is shown in figure 24. Throttle-burst accelerations from idle to rated thrust at an altitude of 10,000 feet and a flight Mach number of 0.19 are compared. The solid line denotes the acceleration during which first stall and then unstall occurred owing to the excessively high maximum fuel limit. The broken curve denotes an acceleration with the maximum fuel limit as high as possible without stall. For the first 5.4 seconds up to the stall point the higher maximum fuel limit resulted in higher values of both thrust and engine speed. After the stall occurred, the thrust fluctuated violently but the general thrust level remained almost constant until unstall occurred 14.9 seconds after the acceleration started. The amplitude of thrust fluctuations is of course attenuated. Upon unstall, the engine thrust increased rapidly; however, stable operation had not been obtained at the end of an 18-second period. In contrast, the proper setting of the maximum fuel limit permitted the successful acceleration to be completed in 14.4 seconds. It will be noted that during the stall, the rate of engine acceleration was reduced; however, the speed did increase gradually. Also, the compressor pressure ratio (not shown) was initially reduced when stall was encountered and thereafter increased slowly as engine speed increased. As a result, the operating point dropped from the stall limit (fig. 13) to a point above the unstall line and moved to the right and up until the unstall limit was reached, whereupon the compressor unstalled. In many of the throttle-burst accelerations, the compressor did not unstall and rated speed could not be reached without exceeding turbine temperature limitation.

Deceleration Characteristics

A series of throttle chops from rated thrust to idle thrust were made at altitudes from 15,000 to 45,000 feet and a flight Mach number of 0.19 and flight Mach numbers up to 0.75 at 25,000 feet to determine the suitability of the minimum fuel limit of 450 pounds per hour for preventing lean combustor blow-out. At no time during the entire investigation was lean combustor blow-out encountered. The effects of altitude and flight Mach number on deceleration characteristics are shown in figures 25 and 26, respectively. Neither engine speed nor thrust decreased as rapidly at an altitude of 45,000 feet as at 15,000 feet because the minimum fuel flow was a larger percentage of the steady-state fuel flow requirement at 45,000 feet than at 15,000 feet and because the density of the working fluid was reduced while the rotor inertia remained constant. Although the final thrust and speed levels are different, the effect of flight Mach number on deceleration characteristics is slight (fig. 26).

2182

As a more severe test of both maximum and minimum fuel limits, a series of runs was made over a wide range of flight conditions in which throttle bursts to rated thrust were made during rapid decelerations and throttle chops to idle were made during rapid acceleration. With the final setting of the maximum and minimum fuel limits, no stall or blow-out was encountered and no appreciable time delay occurred in changing from acceleration to deceleration or deceleration to acceleration.

Altitude Starting Characteristics

The superior altitude starting characteristics of the modified combustor configuration as compared with the original is shown in reference 6. Only the starting data obtained with the RX1-3 engine using the modified combustor configuration are presented.

The maximum windmilling speed and the corresponding flight Mach number at which ignition and flame propagation are possible using the fuel flow scheduled by the control are shown in figure 27(a) as a function of altitude. MIL-F-5624 (AN-F-58) fuel with a Reid vapor pressure of 7 pounds per square inch was used. The fuel was at a temperature of about 70° F and the engine-inlet-air temperature varied from 0° to -6° F. Engine fuel flow varied from an average of 400 pounds per hour at an altitude of 50,000 feet to 650 pounds per hour at 25,000 feet. At an altitude of 50,000 feet, ignition occurred in all combustors at windmilling speeds from 1300 to 1500 rpm. Propagation was poor at 1550 rpm, and no ignition was obtained at windmilling speeds of 1800 rpm or above. As the altitude was reduced the maximum starting speed increased to 2300 rpm at 38,000 feet, and at 35,000 feet starts could be made up to 3500 rpm, the maximum windmilling speed obtainable in the tunnel.

The starting limits obtained with MIL-F-5624 (AN-F-58) fuel, which was treated to give a 1-pound Reid vapor pressure, are shown in figure 27(b). The inlet-air temperature varied from -20° to 30° F and the fuel temperature was about 90° F. At an altitude of 49,000 feet, starts were possible at windmilling speeds up to about 1500 rpm; however, at 25,000 feet starts were not possible above 2200 rpm. The reduction in vapor pressure from 7 pounds to 1 pound appears to have had little effect at 49,000 or 50,000 feet but at 25,000 and 35,000 feet the maximum starting speed was reduced considerably.

It was observed, however, that at altitudes above 40,000 feet, once ignition was obtained it was not possible to accelerate the engine above about 3100 rpm without increasing the ram pressure ratio. This may be explained by the existence of a combustor dead band similar to that shown in reference 7. This dead band results from the fact that the temperature rise required by the engine is greater than the temperature rise obtainable in the combustors. As the dead band is approached, the rate of acceleration drops to zero and some of the combustors blow-out whereas others begin to emit flame through the turbine. An increase in ram pressure ratio decreases the temperature rise required by the engine and permits the acceleration to continue.

Previous work (reference 6) has shown the importance of fuel flow on starting characteristics. Accordingly, a large number of starts were attempted with MIL-F-5624 (AN-F-58) fuel with a vapor pressure of 7 pounds at a 40,000-foot altitude using various values of fuel flow which were set manually. These data, shown in figure 28, define a range of fuel flows in which starts were consistently obtained. This range becomes narrower as the windmilling speed is increased; the maximum speed at which ignition was possible in all combustors was approximately 2410 rpm. The region of certain ignition is bounded by a region in which ignition and flame propagation was possible in some combustors and another region is shown in which no ignition was possible. These data indicate an optimum fuel flow at 40,000 feet of about 650 pounds per hour as contrasted to a value of about 450 pounds per hour scheduled by the control.

Temperature histories of six of the eight combustors, as obtained from motion picture records of individual thermocouples located just downstream of the turbine behind each combustor, are given in figure 29 to show the time required for flame propagation and also the rate of temperature rise for starts at an altitude of 35,000 feet and windmilling speeds of 1600 and 1200 rpm with MIL-F-5624 (AN-F-58) 7-pound vapor pressure fuel. Thermocouples for two combustors were burned out when these data were obtained. Propagation occurred in less than 5 seconds for the runs shown.

The time required at various altitudes to obtain ignition in one combustor (denoted by circles) and in all combustors (denoted by squares) is defined by the shaded areas shown in figure 30. The numbers adjacent to each data point refer to windmilling speed. As the altitude is increased, the time required for ignition in one combustor and the time required for flame propagation to the remaining combustors also increased. There appears to be no consistent effect of windmilling speed on the time required for either ignition or flame propagation.

2180

Afterburner Operational Characteristics

During most of the previous afterburner investigations at the NACA Lewis laboratory, fixed conical exhaust nozzles were used. When exhaust-nozzle sizes were sufficient to permit large thrust augmentation ratios, the burner-inlet temperature was only about 800° F at the time of ignition. With such low burner-inlet temperatures, afterburner ignition was accomplished by the use of the "hot-shot" ignition system discussed in reference 8.

For the more recent investigations, variable-area nozzles were available, permitting burner-inlet temperatures of 1200° to 1300° F at the time of ignition. On two occasions in the previous investigations with burner-inlet temperatures of 1200° to 1300° F, autoignition was obtained with MIL-F-5572 (AN-F-48) grade-80 fuel at high tail-pipe fuel-air ratios. Because of the high tail-pipe fuel-air ratio required for autoignition with this fuel, the starts were violent; one start caused the engine combustors to blow-out and the other resulted in damage to the afterburner. Subsequent experience with MIL-F-5624 (AN-F-58) fuel, which has a slightly lower surface ignition temperature than MIL-F-5572 fuel, has shown that autoignition may be obtained at reasonably low tail-pipe fuel-air ratios without excessive violence. Starting the afterburner by autoignition has therefore become commonplace, although slightly higher fuel-air ratios are required than are needed using the hot-shot system.

For this investigation, afterburner starts, with either autoignition or using the hot-shot ignition system, were obtained at all flight conditions investigated, including an altitude of 53,000 feet where the absolute pressure in the afterburner was 388 pounds per square foot.

The band of tail-pipe fuel-air ratios in which autoignition occurred while the fuel flow was gradually increased is shown in figure 31 as a function of altitude. The tail-pipe fuel-air ratio is defined as the ratio of the afterburner fuel flows to the unburned air entering the afterburner. Different symbols are used to denote various fuel-injection-system and flame-holder configurations. A study of the symbols shows that the fuel-air ratio required for autoignition with a given configuration and altitude does not reproduce exactly. This lack of reproducibility is attributed to variations of as much as 50° F in the burner-inlet temperature. A comparison of these data with preliminary calculations of tail-pipe combustion efficiency indicate that, in general, autoignition occurs at leaner mixtures for configurations having the best steady-state performance.

Data are presented in figure 32 for one typical configuration to show the effect of burner-inlet temperature on autoignition characteristics using MIL-F-5624 (AN-F-58), 7-pound vapor-pressure fuel. At an altitude of 25,000 feet and a flight Mach number of 0.19, the effect of temperature was negligible over the range investigated. At 35,000 feet, however, the tail-pipe fuel-air ratio required for autoignition increased from about 0.008 at 1760° R to 0.038 at 1660° R, or increase of almost 5 times for a 100° F reduction in burner-inlet temperature.

Oscillograph traces are given in figures 33 to 35 to show a throttle burst from full dry thrust to full afterburning, full dry thrust to partial afterburning, and a throttle chop from full afterburning to full dry thrust at an altitude of 25,000 feet and a flight Mach number of 0.19. In figure 33, which shows a throttle burst from full dry thrust to full afterburning, a period of about 9 seconds elapsed before autoignition occurred, as noted from a comparison of the traces of afterburner fuel flow and engine mount force. When ignition occurred in the afterburner, the increase in turbine back pressure (not shown) caused the engine speed to decrease about 160 rpm even though the engine fuel flow was increasing to maintain constant speed. As a result of the increase in fuel flow and the reduction in engine speed, the turbine-outlet temperature became excessive, reaching a value of 1690° F before the overtemperature, underspeed condition was alleviated by the opening of the exhaust nozzle. The exhaust nozzle required 1.6 seconds to open. A reduction in the opening time of the exhaust nozzle and a reduction in thermocouple response time would reduce the amount of temperature overshoot. After the exhaust nozzle had opened fully, the turbine-outlet temperature was still excessive. Accordingly, the afterburner fuel flow was reduced by the control in response to the turbine-outlet temperature signal to prevent damage to the turbine. At the time these data were obtained the response of the afterburner fuel valve to overtemperature conditions was excessively rapid, resulting in the afterburner fuel flow being reduced almost to one-third of the original value. The reduction in afterburner fuel flow caused the turbine-outlet temperature to drop below the limiting value, thereby permitting the afterburner fuel valve to open. The afterburner fuel again increased to an excessive value causing overtemperature, and as a result the cycle of events repeated with little attenuation.

A throttle burst from full dry thrust to partial afterburning shown in figure 34 was accompanied by a few cycles in which turbine temperature became excessive and the engine speed was reduced as a result of the simultaneous action of the exhaust nozzle, which tries to maintain constant turbine-outlet temperature, and engine fuel flow,

2180

which attempts to maintain constant engine speed. Because the turbine-outlet temperature was controlled by the exhaust nozzle (which was not fully open) rather than the afterburner fuel flow, the instability exhibited by the data of figure 33 was not encountered. Equilibrium running conditions were restored at point A, 12.5 seconds after ignition occurred.

A throttle chop from the full afterburning condition to the full dry thrust condition is shown in figure 35. The time required for most of the variables to return to equilibrium was about equal to the time required for the exhaust nozzle to close, approximately 8.5 seconds. When the afterburner fuel flow was reduced, the turbine-outlet temperature dropped from 1320° to 1000° F. Simultaneously the engine speed increased about 100 rpm. Engine fuel flow was modified by the control to restore speed and at the same time, the exhaust-nozzle area was reduced to restore the turbine-outlet temperature.

The relatively slow closure of the exhaust nozzle is probably the result of continued burning of a small amount of fuel in the afterburner, inasmuch as the fuel flow to the afterburner did not stop completely for about 7 seconds. During most of the transient, the thrust was less than the rated value. Equilibrium turbine-outlet temperature, indicated on the oscillograph trace after the transient, was 30° F lower than the original value at full afterburning, probably as a result of slight discrepancies between the indicating thermocouples and those used by the control.

For operation at a flight Mach number of 0.19, afterburner autoignition delay, measured from oscillograph traces, varied with altitude as follows:

Altitude (ft)	Average ignition lag (sec)
15,000	3.6
25,000	7.0
35,000	15.1
45,000	40.0

Lean blow-out limits for several afterburner configurations, comprised of changes in fuel distribution and flame holders, are shown as a function of altitude in figure 36 for a flight Mach number of 0.19. The configuration changes which were made had no significant effect on the results; however, it should be pointed out that the flame-holder blocked area was not altered. The minimum tail-pipe fuel-air ratio for lean blow-out increased from 0.004 at an altitude of 15,000 feet to about 0.013 at 50,000 feet. The width of the blow-out region was not markedly changed with altitude. Rich blow-out limits were not obtained because at low altitudes the exhaust-nozzle size limited the maximum fuel-air ratio and at high altitudes operation was not attempted beyond the fuel-air ratio producing the maximum exhaust-gas temperature.

2180

SUMMARY OF RESULTS

From an investigation of J47D (RX1-1) and (RX1-3) turbojet engines (with integrated electronic controls) in the NACA Lewis altitude wind tunnel over a range of altitudes up to 55,000 feet at a flight Mach number of 0.19 and flight Mach numbers up to 0.89 at an altitude of 25,000 feet, the following results were obtained:

1. For the complete range of altitudes and flight Mach numbers investigated, compressor stall data reduced to single curves for both engines on plots of compressor pressure ratio against corrected engine speed. The curves for both engines were similar and each showed two distinct segments indicating stall in different portions of the compressor. The transition, however, occurred at slightly different engine speeds for the two engines.

2. On the same coordinates at a given corrected engine speed, the compressor unstalled at slightly higher pressure ratios as the altitude was increased. Flight Mach number had no apparent effect on compressor unstall characteristics within the range investigated.

3. The unstall compressor pressure ratio occurred at a lower value than either the stall or steady-state-operation compressor pressure ratios for given corrected engine speeds above 5800 rpm.

4. A maximum fuel limit scheduled as a function of compressor-outlet pressure provided adequate protection against both compressor stall and combustor blow-out and required the measurement of only one variable; however, the limit appears conservative for operation at high flight Mach numbers.

5. Both combustor blow-out and compressor stall data reduced to a single curve on coordinates of compressor pressure ratio against corrected engine speed, thereby providing a relation which could be used for protection against these difficulties.

2180

6. The time required to accelerate the controlled engine from idle to rated thrust increased from about 14 seconds at 15,000 feet to 22 seconds at 45,000 feet for operation at a flight Mach number of 0.19. At an altitude of 25,000 feet, an increase in flight Mach number from 0.19 to 0.75 reduced the acceleration time from 14.4 to 6 seconds.

7. For the complete range of flight conditions investigated, lean combustor blow-out could not be obtained using a constant minimum fuel limit of 450 pounds per hour.

8. Using MIL-F-5624 (AN-F-58) fuel with a 7-pound Reid vapor pressure at a temperature of about 70° F and inlet-air temperatures from 0° to -6° F, ignition during automatic starts was possible in all combustors at windmilling speeds from 1300 to 1500 rpm at an altitude of 50,000 feet. At these conditions ignition was possible in some combustors up to 2100 rpm. At 35,000 feet, ignition was possible in all combustors up to 3500 rpm, the highest windmilling speed obtainable. At altitudes above 40,000 feet, however, the presence of a combustor dead band prevented acceleration above 3100 rpm at the lower flight Mach numbers.

9. Using MIL-F-5624 (AN-F-58) fuel (treated to give a 1-pound vapor pressure) at a temperature of approximately 90° F and inlet-air temperatures from -20° to 30° F, ignition was possible in all combustors at 49,000 feet up to a windmilling speed of about 1500 rpm. At 25,000 feet, however, starts were not possible above 2200 rpm.

10. At an altitude of 40,000 feet, the optimum fuel flow for starting appeared to be about 650 pounds per hour for MIL-F-5624 (AN-F-58) fuel.

11. Afterburner starts by autoignition using MIL-F-5624 (AN-F-58) fuel were obtained at altitudes up to 53,000 feet at a flight Mach number of 0.19. The tail-pipe fuel-air ratio required for autoignition increased with altitude and at 35,000 feet decreased as the burner-inlet temperature was raised.

12. The tail-pipe fuel-air ratio at which lean blow-out of the afterburner occurred was increased as altitude was raised. The width of the blow-out band remained constant over the range of altitudes.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio.

APPENDIX A

DESCRIPTION AND OPERATION OF INTEGRATED ELECTRONIC CONTROL

Principles of Operation

At all operating conditions, the engine is controlled by modulation of fuel flow. As noted previously, steady-state exhaust-nozzle area is scheduled as a function of thrust-selector position (figs. 5 and 6) for operation in the nonafterburning region. Under afterburning conditions, afterburner fuel flow (corrected for altitude and ram) is scheduled against thrust selector position, and limiting turbine-outlet temperature is maintained by modulation of the exhaust-nozzle area. The detailed functions of the control were listed previously; the manner in which the control was designed to accomplish the more important objectives will be discussed in the following paragraphs.

Engine speed control. - Engine speed, scheduled as a function of thrust-selector position, is controlled by suitable modulation of engine fuel flow in response to a speed-error signal. The speed-error signal is the difference between two voltages; one voltage from the speed selector (fig. 5) is proportional to the desired speed, whereas the other voltage is the output of a tachometer unit (fig. 5) driven by the engine and consequently represents the actual speed. This speed-error signal is amplified by the servo amplifier unit (fig. 5) and used to drive a motor-actuated fuel valve located in the main fuel control. If the desired engine speed called for by the thrust selector is higher than the actual engine speed sensed by the tachometer, a positive speed-error signal results. In response to a positive error signal, the valve opens, increasing the fuel flow and producing an acceleration until the speed-error signal is reduced to zero when the desired speed is reached. Similarly, a negative speed-error signal causes the valve to close, reducing the engine speed.

An important requirement of a control is that it maintain constant engine speed under changing flight conditions. This is accomplished by the same system of error signals just discussed. An increase in flight Mach number, for example, will cause the actual speed to increase above the speed set by the thrust selector, producing a negative speed-error signal which reduces the fuel flow and restores or maintains the initial engine speed.

Overspeed protection is afforded by a tuned circuit which produces a negative speed-error signal; the error-signal voltage increases very rapidly in the overspeed region, causing a rapid

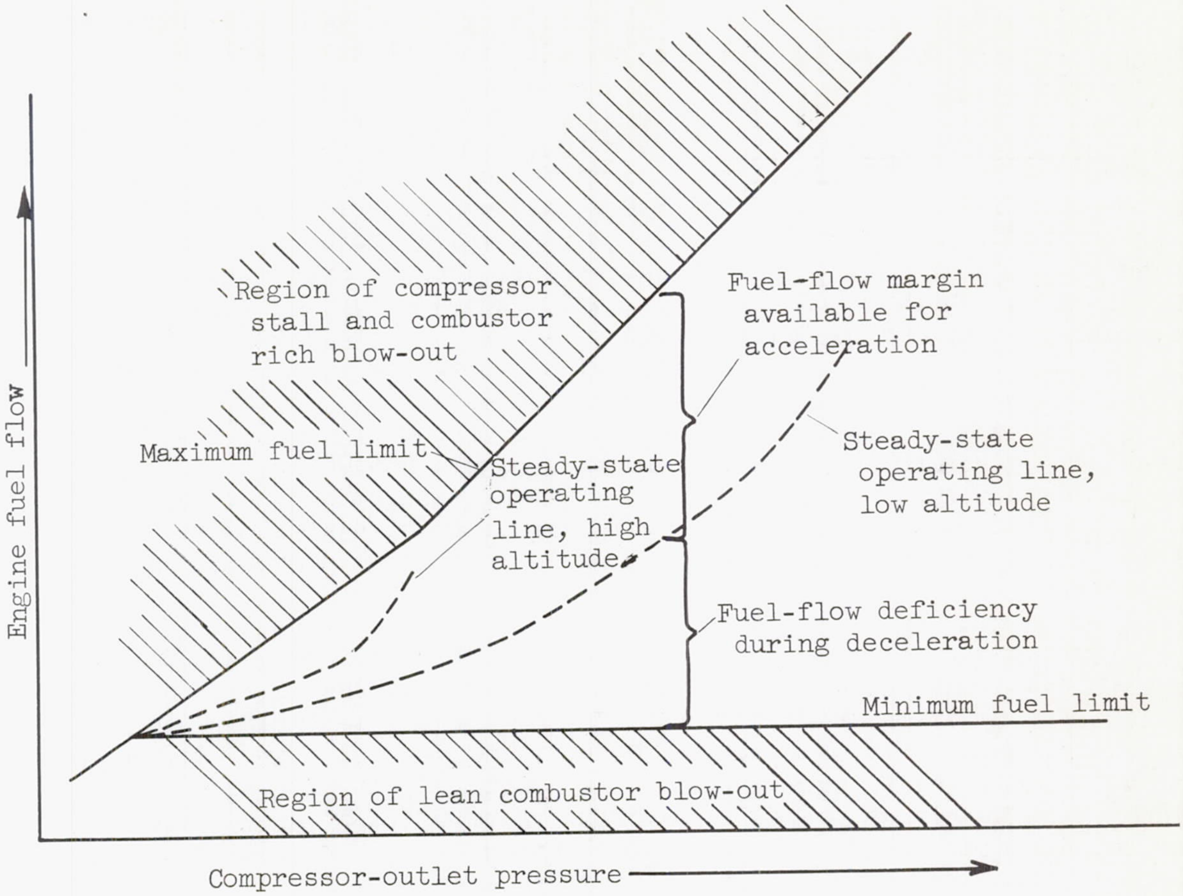
2180

reduction in fuel flow. This protective circuit attempts to hold rated engine speed within 20 rpm. Further protection is provided by an overspeed governor working on the fly-ball principle. This governor is set to bypass fuel back to the pump inlet at 8050 rpm; however, full bypass is not obtained until the engine reaches a speed of
2180 8300 rpm.

Engine temperature control. - With the exhaust-nozzle area schedule used (fig. 6) steady-state turbine-outlet temperatures are well below the limiting value of 1275° F at all engine speeds except rated engine speed. At rated dry thrust (90° thrust selector position) even though a given value of exhaust-nozzle area is scheduled by the nozzle-area selector, the exhaust nozzle, actuated by the nozzle actuator, is permitted to be closed only until limiting temperature is reached. Because of the effects of Reynolds number on component efficiencies, the exhaust-nozzle area producing limiting turbine-outlet temperature increases with altitude. From figure 6 it may be seen, therefore, that rated thrust at high altitudes will be obtained at thrust-selector positions slightly below 90° and that a dead band on the thrust selector will result at high altitudes. To keep this dead band narrow, the scheduled area is changed rapidly in this region.

During engine starts and accelerations at speeds below 7200 rpm, a turbine-outlet temperature limit (as measured by the thermocouple unit (fig. 5) of about 1500° F is imposed. If the temperature tends to exceed this limit, the control reduces engine fuel flow. A smooth transition of the temperature limit is provided from the value of 1500° F at speeds up to 7200 rpm to the value of 1275° F in effect at rated speed. Under afterburning conditions, the exhaust-nozzle area is modulated to maintain limiting temperature. In the event of nozzle failure, or if the nozzle is wide open, an overtemperature of 20° F will cause the afterburner fuel flow to be reduced to a value consistent with the temperature limit.

Acceleration stall and blow-out protection. - As noted previously under RESULTS AND DISCUSSION, it is possible to provide protection against compressor stall and combustor blow-out by scheduling the maximum and minimum fuel flows (corrected for temperature) as functions of compressor-outlet pressure. Experimental data given in figure 17 showed that the maximum fuel-limit curve should be comprised of two straight-line segments. The relation between the fuel limit and steady-state operating lines for both high and low altitudes is given in the following sketch:



2180

The margin of fuel flow in excess of the steady-state requirements available for acceleration is also indicated by the vertical distance between the steady-state operating line and the maximum fuel limit. The optimum acceleration characteristics are of course obtained with the maximum fuel limit set to border the region of stall and rich blow-out. Both the maximum fuel limit and the maximum temperature limit discussed previously are in effect simultaneously and either may impose a restriction on the rate of acceleration. At low engine speeds, the maximum fuel limit is usually encountered first whereas near rated engine speeds, the maximum temperature limit (which decreases near rated speed) is generally the controlling limit.

Fuel-flow deficiency during deceleration represents the difference between the fuel flow required to maintain steady-state conditions and the fuel flow required to prevent lean combustor blow-out. This deficiency is indicative of the forces tending to reduce the engine speed. As noted in the text, lean combustor blow-out was not obtained with the minimum fuel limit set at the value of 450 pounds per hour.

When accelerations of large magnitude are made and speed-error signals equivalent to 400 rpm are present, the exhaust-nozzle area is locked at the initial area until the approximate final speed is reached or until a speed of 7800 rpm is reached. After release, the exhaust nozzle closes to the steady-state exhaust-nozzle schedule.

Provision of services. - All services required in the various regions of operation are scheduled against thrust-selector position or engine speed. The following are some examples. When the thrust selector is advanced from 0° to the idle position of 10°, fuel flow is provided, the engine starter is activated, and ignition is supplied. At an engine speed of about 2000 rpm the starter "cuts out". Just above the 90° thrust-selector position, the afterburner fuel shut-off valve, the valve supplying air to the afterburner fuel pump, and the valve supplying cooling air to the exhaust nozzle open. A relay controlling the afterburner fuel shut-off valve is closed at 7200 rpm, preventing afterburner operation at lower speeds during a burst from low speed into the afterburning region.

2180

Stabilization. - A two-phase motor is used with a gear reduction to move the engine fuel-flow control valve. This motor is driven by the amplified error signal. A tachometer connected to the motor supplies a voltage proportional to motor speed. This voltage modulates the error signal, which is the input to the amplifier. When the error signal is small, the tachometer output tends to prevent small oscillations; however, when the error signal is large, the tachometer output is overpowered and the motor is permitted to operate at full speed.

A potentiometer is also connected to the gear reduction. The output of this potentiometer is used to oppose the error signal. When a speed change is called for by the thrust selector, the motor begins to move the fuel valve to a new position. At the beginning of the transient the error signal is large and the signal from the potentiometer has little effect; however, when the actual engine speed approaches the set speed the error signal becomes sufficiently small so that the signal from the potentiometer is effective. This signal tends to stop the motor before the set speed is reached. The potentiometer feed-back signal is slowly reduced to zero in a short time to permit the set speed to be obtained; however, the anticipatory action provides stability as the engine approaches the set speed.

2180

APPENDIX B

SYMBOLS

The following symbols are used in this report:

A_t	effective area at turbine-nozzle diaphragm
f/a	fuel-air ratio
g	acceleration due to gravity, 32.174 ft/sec^2
K_1, K_2, K_3	constants
M	Mach number
N	engine speed
P	total pressure, lb/sq ft absolute
p	static pressure, lb/sq ft absolute
R	gas constant, ft-lb/(lb)(^{o}R)
T	total temperature, ^{o}R
t	static temperature, ^{o}R
W_a	air flow, lb/sec
W_g	gas flow, lb/sec
W_f	fuel flow, lb/hr
η_c	adiabatic compressor efficiency
η_b	burner efficiency
η_t	adiabatic turbine efficiency
γ	ratio of specific heats
θ	ratio of absolute static temperature at engine inlet to absolute static temperature at NACA standard atmospheric sea-level conditions

Subscripts:

e	engine
0	free air stream
1	engine inlet
3	compressor outlet
4	turbine inlet
6	turbine outlet

APPENDIX C

RELATIONS USED FOR STALL AND BLOW-OUT PROTECTION

A relation which may be used as a basis for stall and blow-out protection may be developed from a consideration of the flow conditions that exist at the turbine nozzle diaphragm. For operation at low power levels, the turbine nozzle diaphragm is not choked, and gas flow is given by the following equation:

$$W_g = A_t\ p_4\sqrt{\frac{2g\gamma}{(\gamma-1)(R)}}\left\{\frac{1}{t_4}\left[\left(\frac{p_6}{p_4}\right)^{\frac{2}{\gamma}} - \left(\frac{p_6}{p_4}\right)^{\frac{\gamma+1}{\gamma}}\right]\right\}^{1/2} \tag{1}$$

For steady-state operating conditions, if the various specific heats are assumed constant and the fuel flow is neglected, the turbine total pressure ratio P_4/P_6 may be shown to be related to compressor total pressure ratio P_3/P_1 by the following expression:

$$\frac{P_6}{P_4} = \left\{\frac{T_1}{\eta_c\ \eta_t\ T_4}\left[1 - \left(\frac{P_3}{P_1}\right)^{\frac{\gamma-1}{\gamma}}\right] + 1\right\}^{\frac{\gamma}{\gamma-1}} \tag{2}$$

As a first approximation, assume that the total temperatures and pressures in equation (2) are equal to the static temperatures and pressures, respectively, and that the turbine-inlet static pressure p_4 is equal to the compressor-outlet static pressure p_3. A study of equations (1) and (2) reveals that for a given engine operating condition ($W_g = K$) and fixed flight conditions (p_1 and t_1 constant) the compressor-outlet pressure is a function of the turbine-inlet temperature.

For most of the normal engine operating conditions, the turbine nozzle diaphragm is choked and the following is true:

$$W_g \propto \frac{p_4}{\sqrt{t_4}} \propto \frac{p_3}{\sqrt{t_4}} \tag{3}$$

During an acceleration, however, equation (3) is valid for lower engine speed than that encountered in steady-state operation, because the turbine pressure ratio must be increased to produce an acceleration.

2180

Turbine-inlet temperature is given by

$$t_4 = \Delta t_b + t_3$$

where the combustor temperature rise Δt_b is

$$\Delta t_b = \eta_b (f/a)_e K_1$$

and

$$t_3 = \frac{t_1\left[\left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]}{\eta_c} + t_1$$

or if η_c is assumed equal to 1 throughout the transient,

$$t_3 = t_1 \left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma}}$$

therefore,

$$t_4 = t_1 \left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma}} + \eta_b (f/a)_e K_1$$

If η_b and W_a are constant and $p_3 = p_4$ from equation (3), the following expression is obtained:

$$p_3^{\,2} \propto K_3 \left[t_1 \left(\frac{p_3}{p_1}\right)^{\frac{\gamma-1}{\gamma}} + K_2 W_f\right]$$

$$\frac{p_3}{p_1} = f\left(\frac{p_3^{\,2} - W_f}{t_1}\right) \tag{4}$$

From equation (4) it may be seen that the fuel flow and the compressor-outlet pressure (corrected for inlet temperature) may be scheduled in such a manner that the compressor pressure ratio will remain below the value producing stall. Stall data obtained at an engine-inlet temperature corresponding to NACA standard condition and inlet temperature of 140° F (fig. 12) indicate that the effect of engine-inlet temperature may be ignored. The imposition of a limit on fuel flow for a fixed compressor-outlet pressure also limits the combustor fuel-air ratio and thereby tends to prevent combustor blowout. Because of the assumptions made and the approximations used, experimental determination of the proper schedule of fuel flow against compressor-outlet pressure was necessary.

2180

REFERENCES

1. Saari, M. J., and Wintler, J. T.: Altitude-Wind-Tunnel Investigation of Performance Characteristics of J47D (RX1-1) Turbojet Engine with Fixed-Area Exhaust Nozzle. NACA RM E51B06,

2. Conrad, E. William, and McAulay, John E.: Altitude-Wind-Tunnel Investigation of Performance Characteristics of J47D (RX1-1) Turbojet Engine with Variable-Area Exhaust Nozzle. NACA RM E51C15, 1951.

3. Wells, B. A., Williams, J. F., and Yates, C. G.: Study of Integrated Electric Control for Turbo-Jet Engines. Rep. TR-55410. Aero. and Ord. Systems Divs., Gen. Elec. Co. (Schenectady, N.Y.)

4. Delio, Gene J., and Schwent, Glennon V.: Instrumentation for Recording Transient Performance for Gas-Turbine Engine and Control Systems. NACA RM E51D27, 1951.

5. Constant, Hayne: Gas Turbines and Their Problems. Todd Publishing Group, Ltd., (London), 1949.

6. Golladay, Richard L., and Bloomer, Harry E.: Investigation of Altitude Starting and Acceleration Characteristics of J47 Turbojet Engine. NACA RM E50G07, 1951.

7. Fleming, William A.: Altitude-Wind-Tunnel Investigation of Westinghouse 19B-2, 19B-8, and 19XB-1 Jet-Propulsion Engines. I - Operational Characteristics. NACA RM E8J28, 1948.

8. Conrad, E. William, and Prince, William R.: Altitude Performance and Operational Characteristics of 29-Inch-Diameter Tail-Pipe Burner with Several Fuel Systems and Flame Holders on J35 Turbojet Engine. NACA RM E9G08, 1949.

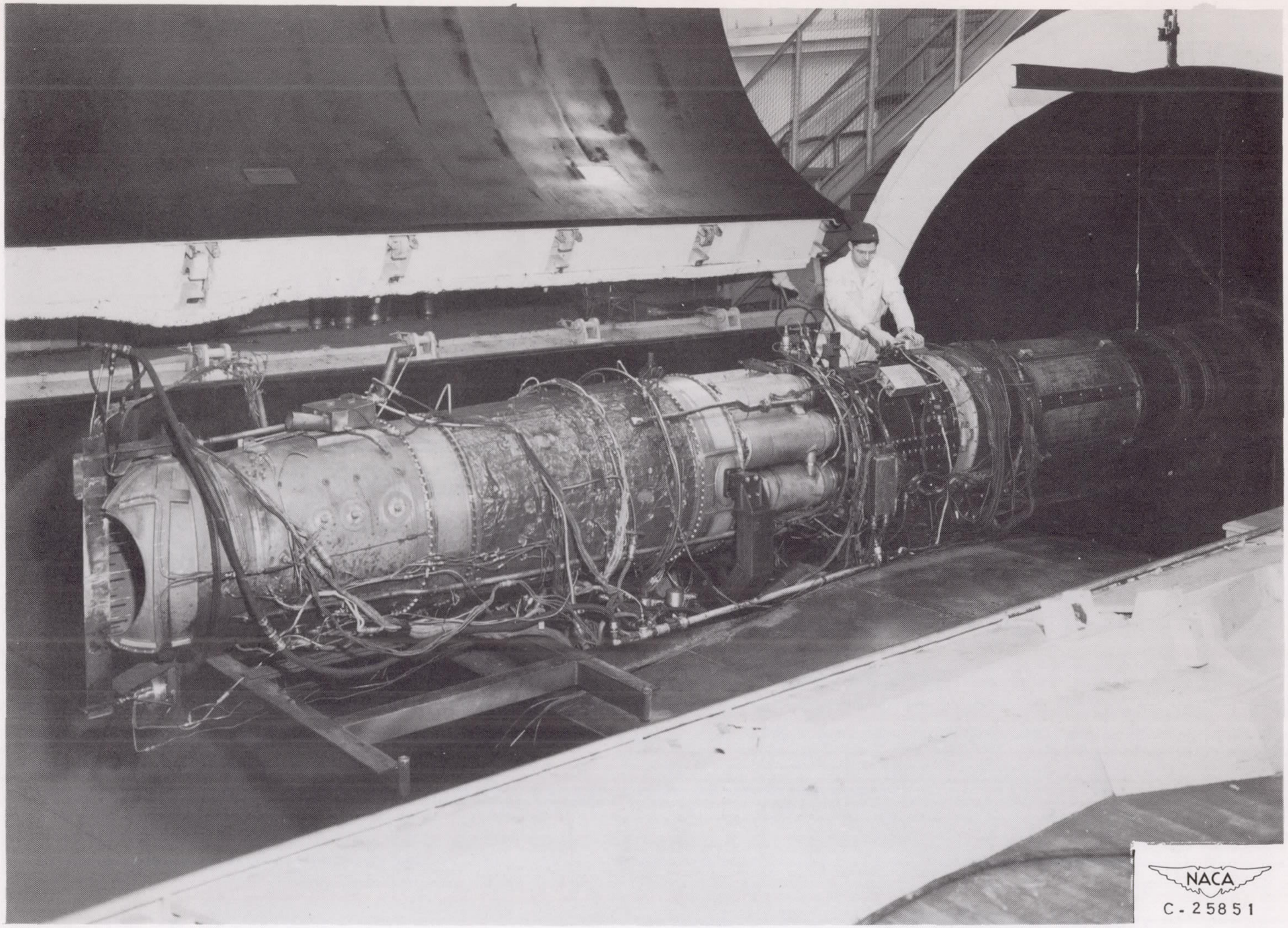
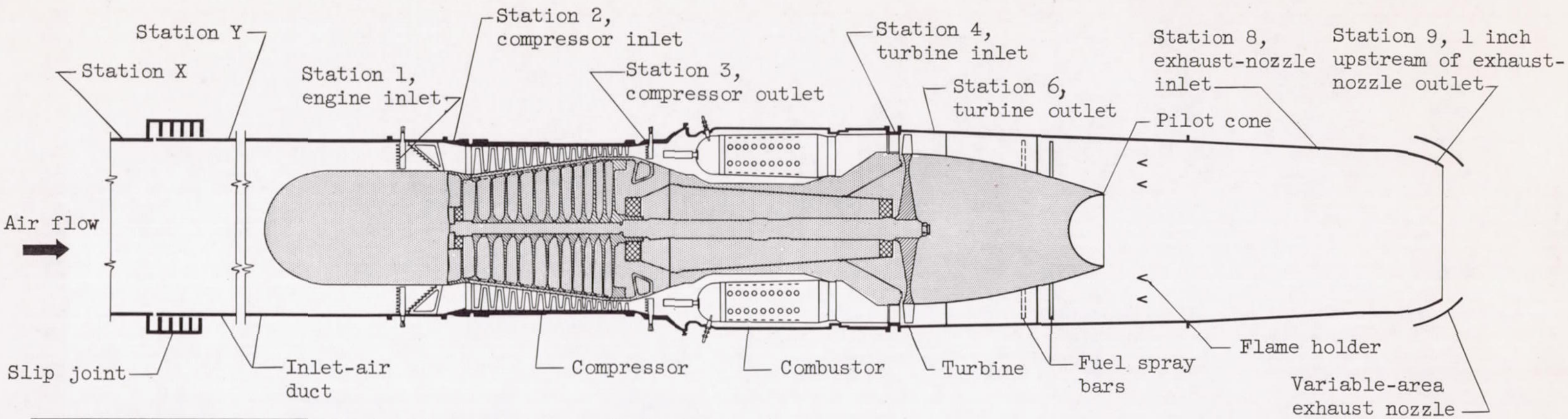


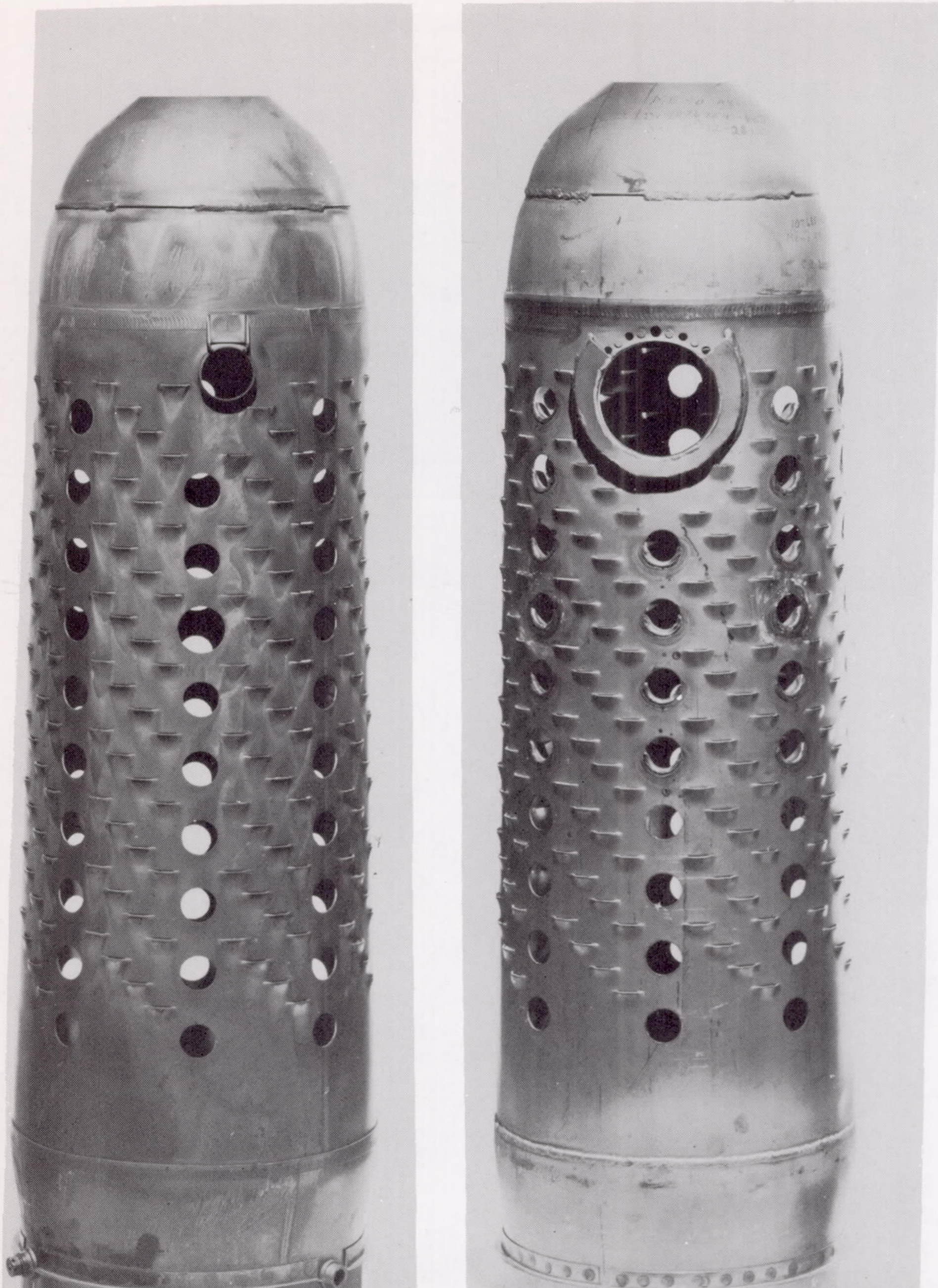
Figure 1. - Installation of turbojet engine in altitude wind tunnel.



Station	Total-pressure tubes	Static-pressure tubes	Wall static-pressure orifices	Thermo-couples
x	0	0	4	0
y	1	2	4	2
1	32	8	5	4
2	6	0	2	0
3	20	0	4	6
4	5	0	0	0
6	30	0	4	24
8	0	0	0	25
9	20	3	4	10

Figure 2. - Cross section of turbojet-engine installation showing stations at which instrumentation was installed.

2180



C-25657

(a) Original combustor liner.

(b) Modified combustor liner.

Figure 3. - Original and modified combustor liners.

2180

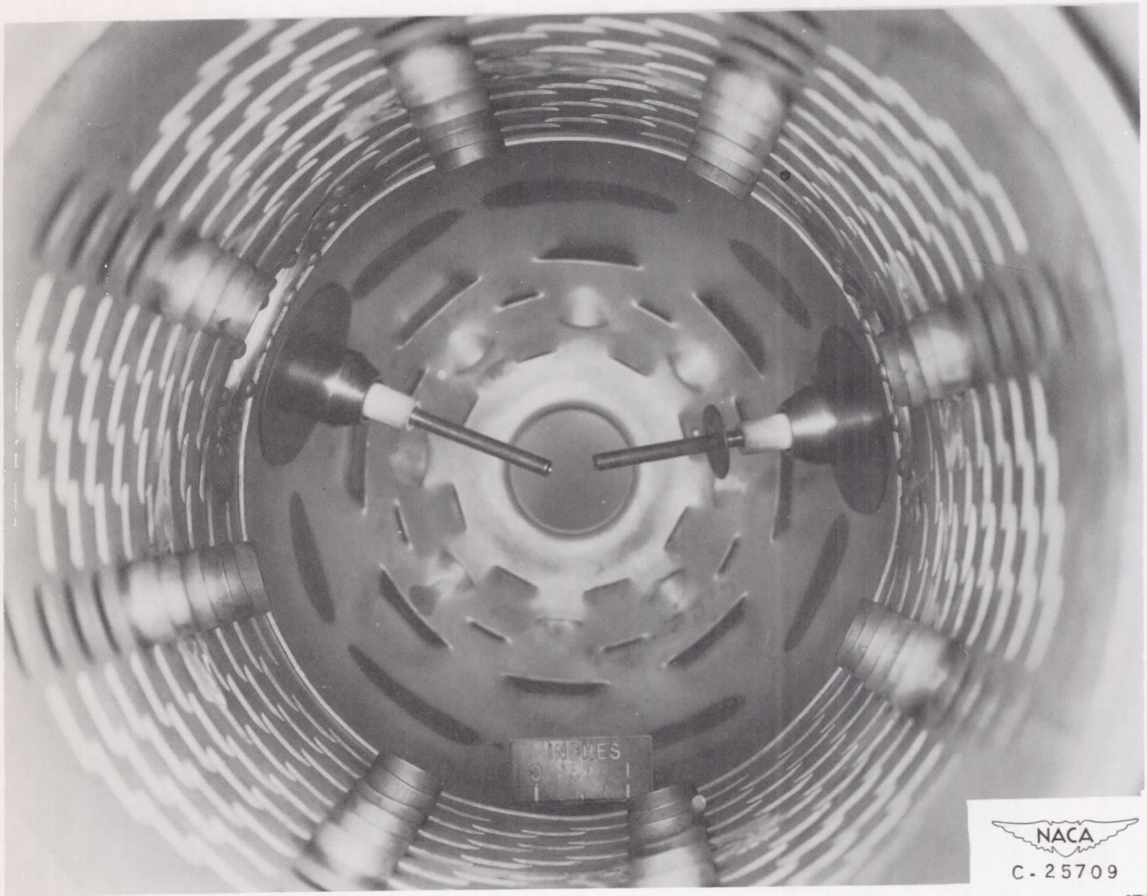


Figure 4. - View looking upstream of modified combustor showing opposite polarity spark plugs and shrouds.

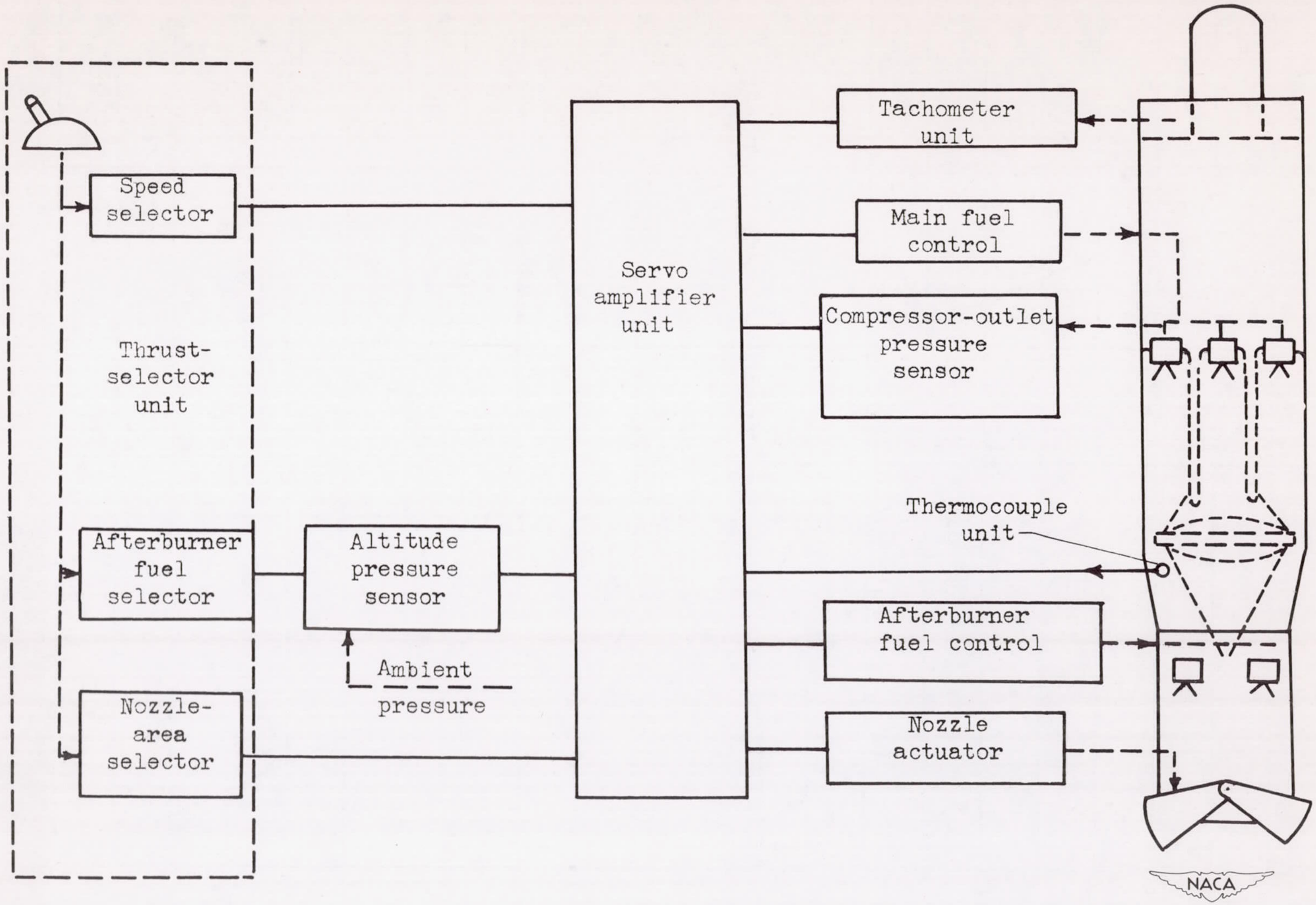


Figure 5. - Block diagram of integrated electronic control.

2180

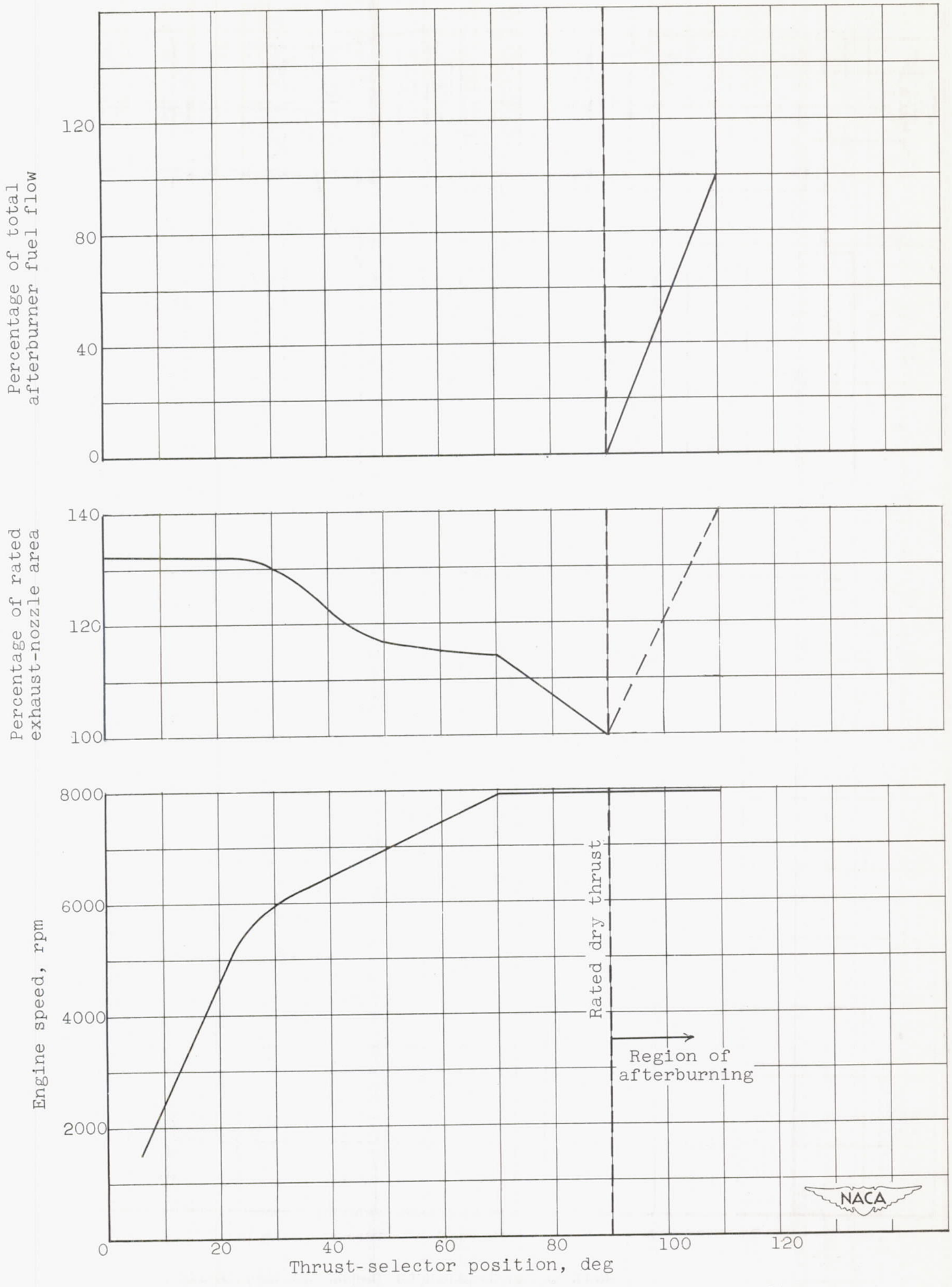


Figure 6. - Schedule of engine speed, exhaust-nozzle area, and afterburner fuel flow with thrust-selector position.

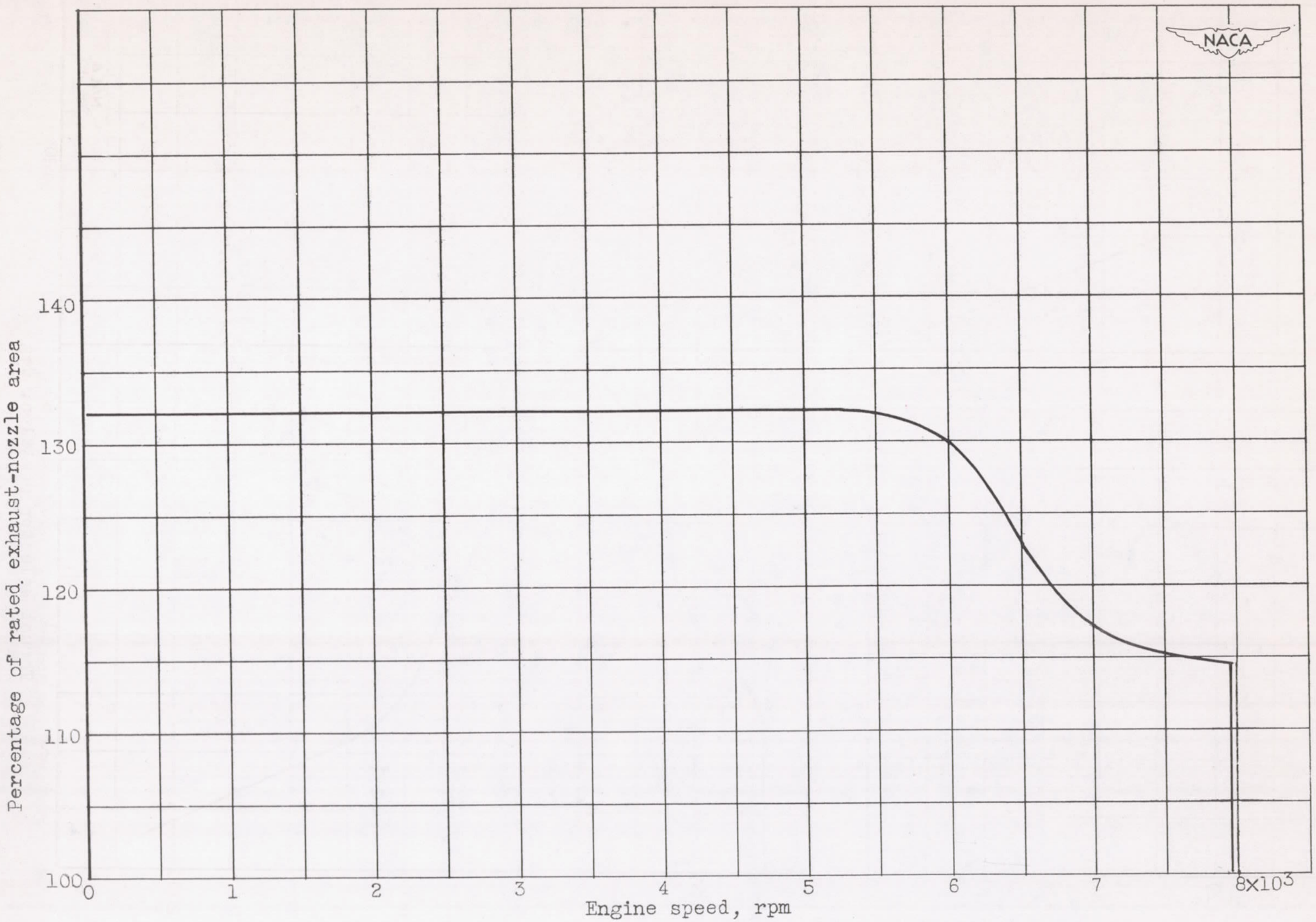


Figure 7. - Schedule of percentage of rated exhaust-nozzle area to engine speed.

2180

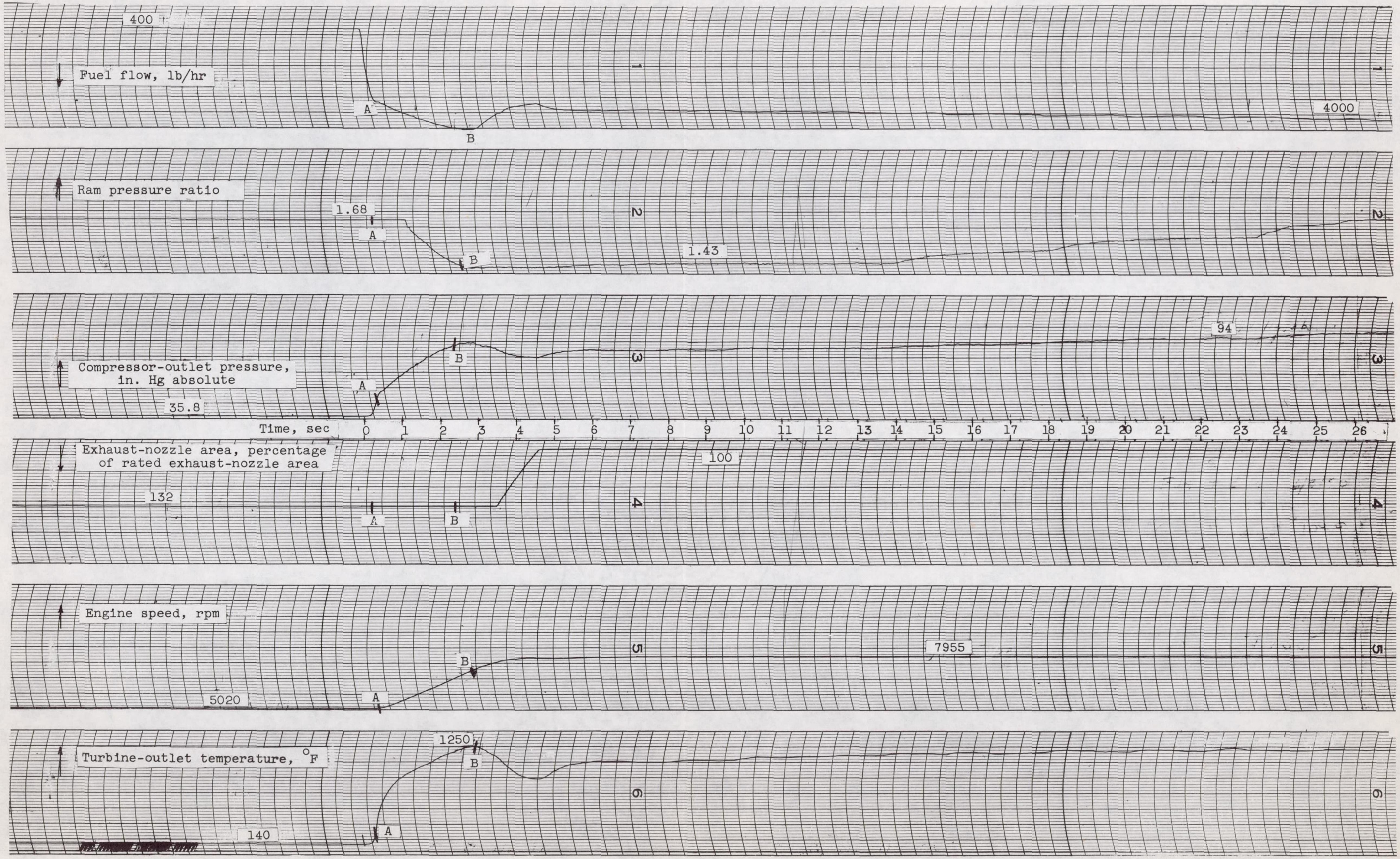
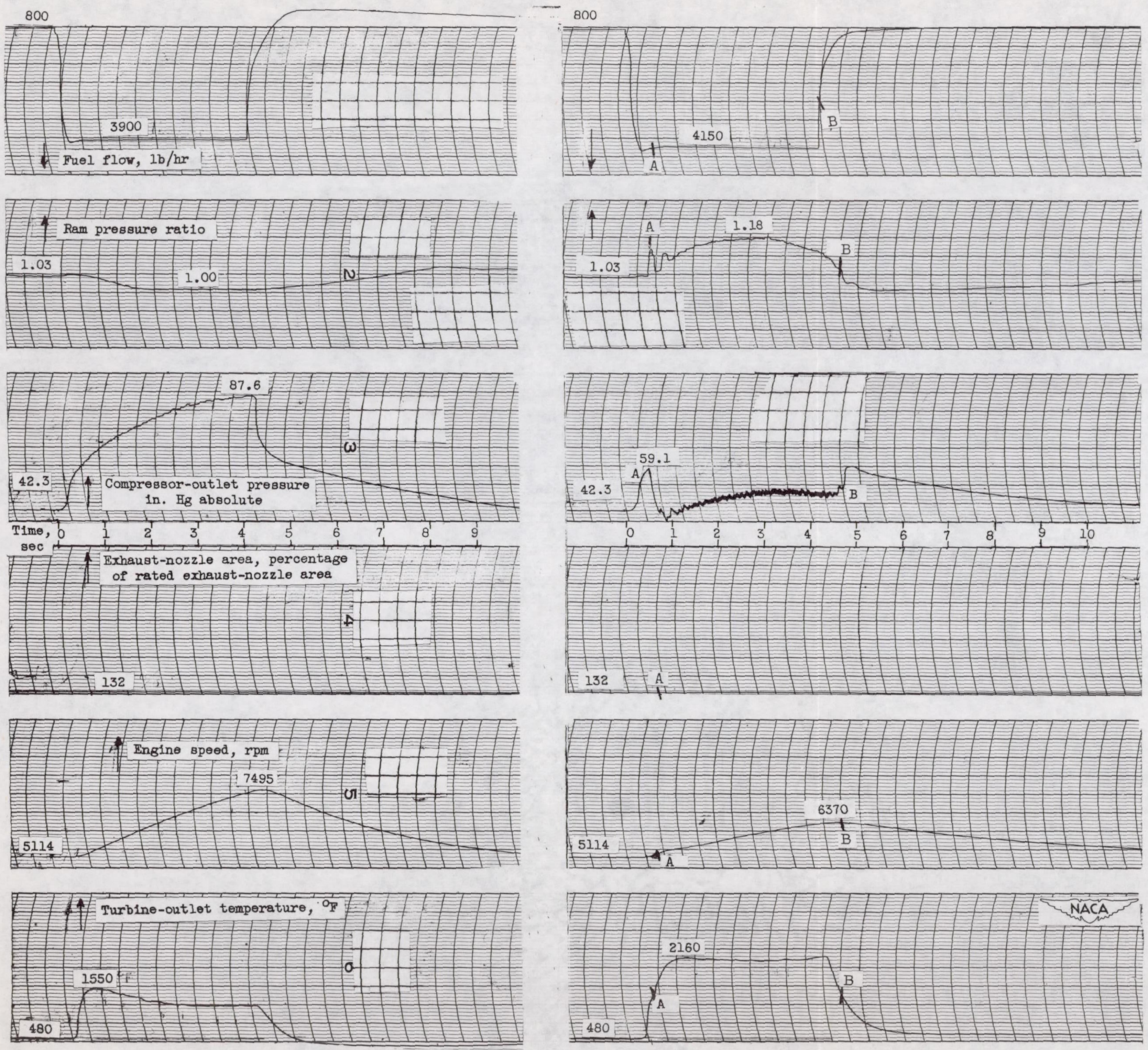


Figure 8. - Behavior of several engine variables with time during automatically controlled acceleration from idle to full dry thrust. Altitude, 25,000 feet; flight Mach number, 0.89.

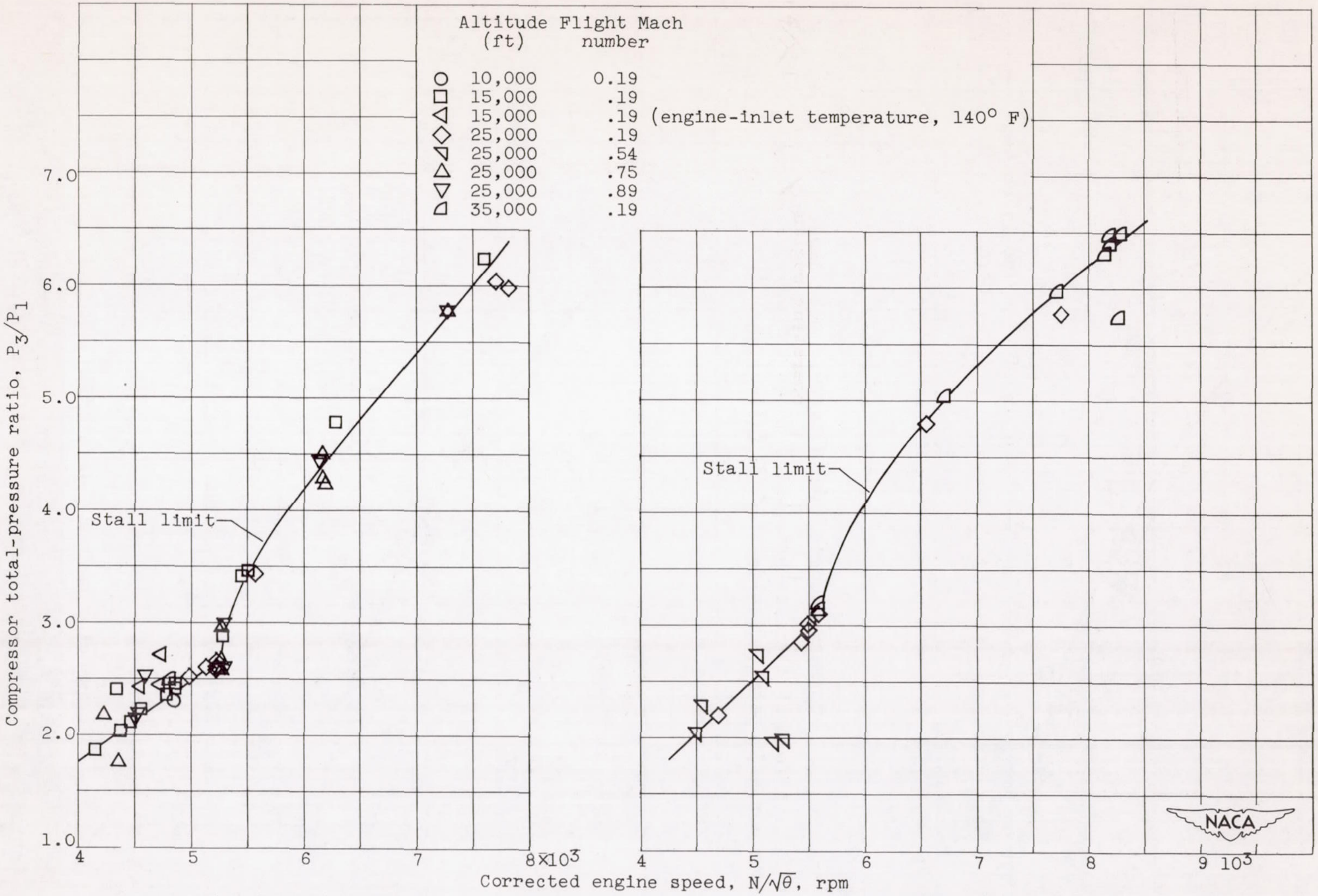
2180



(a) Successful acceleration.

(b) Stall.

Figure 9. - The behavior of several engine variables during acceleration which resulted from step increase in fuel flow. Altitude, 15,000 feet; flight Mach number, 0.19.



(a) (RX1-3) engine.

(b) (RX1-1) engine.

Figure 10. - Correlation of compressor stall with compressor total-pressure ratio and corrected engine speed for J47D engines.

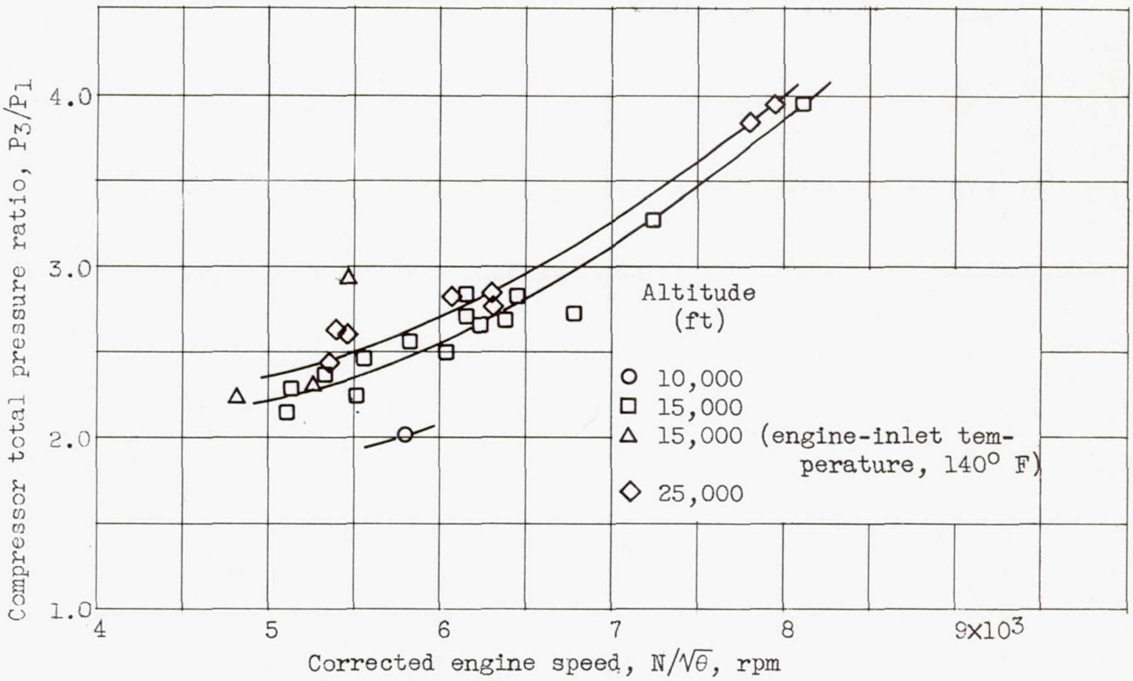


Figure 11. - Effect of altitude and engine-inlet temperature on compressor unstall characteristics of J47D (RX1-3) engine. Flight Mach number, 0.19.

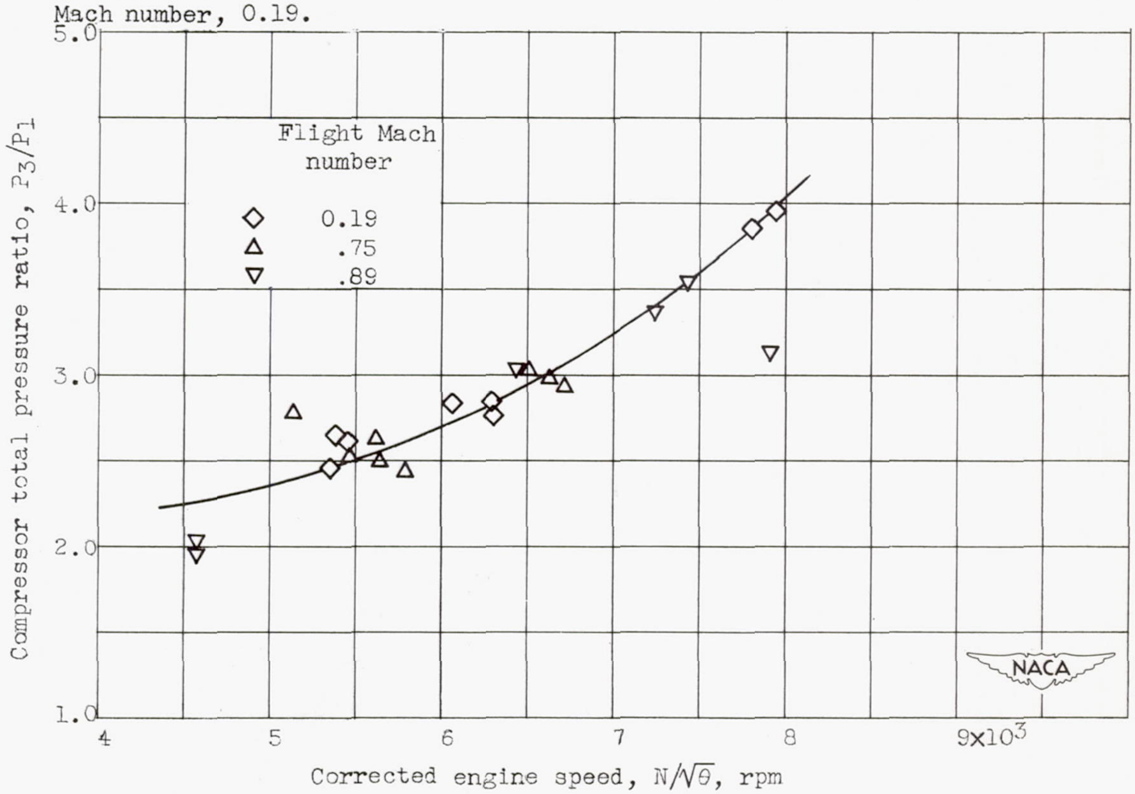


Figure 12. - Effect of flight Mach number on compressor unstall characteristics of J47D (RX1-3) engine. Altitude, 25,000 feet.

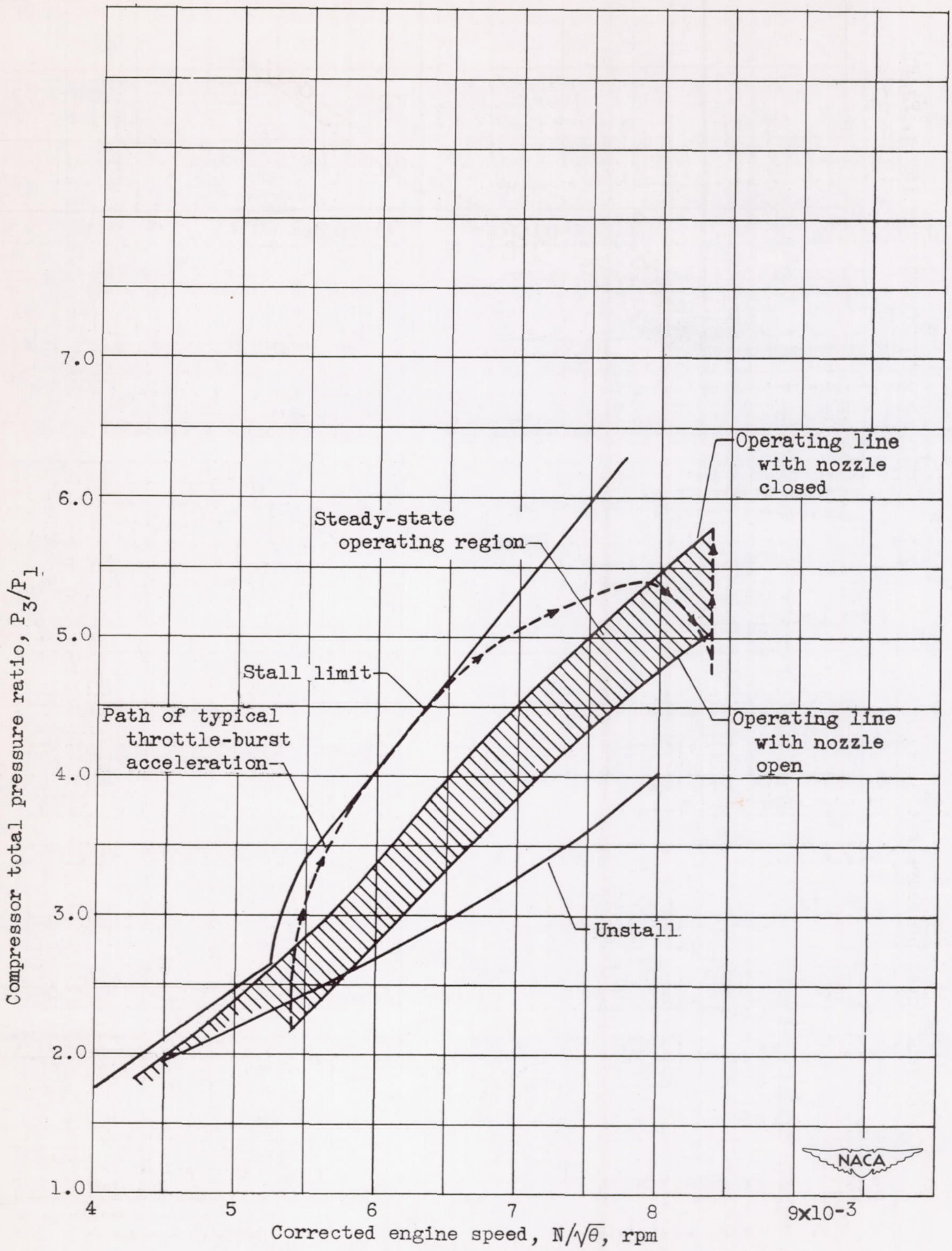
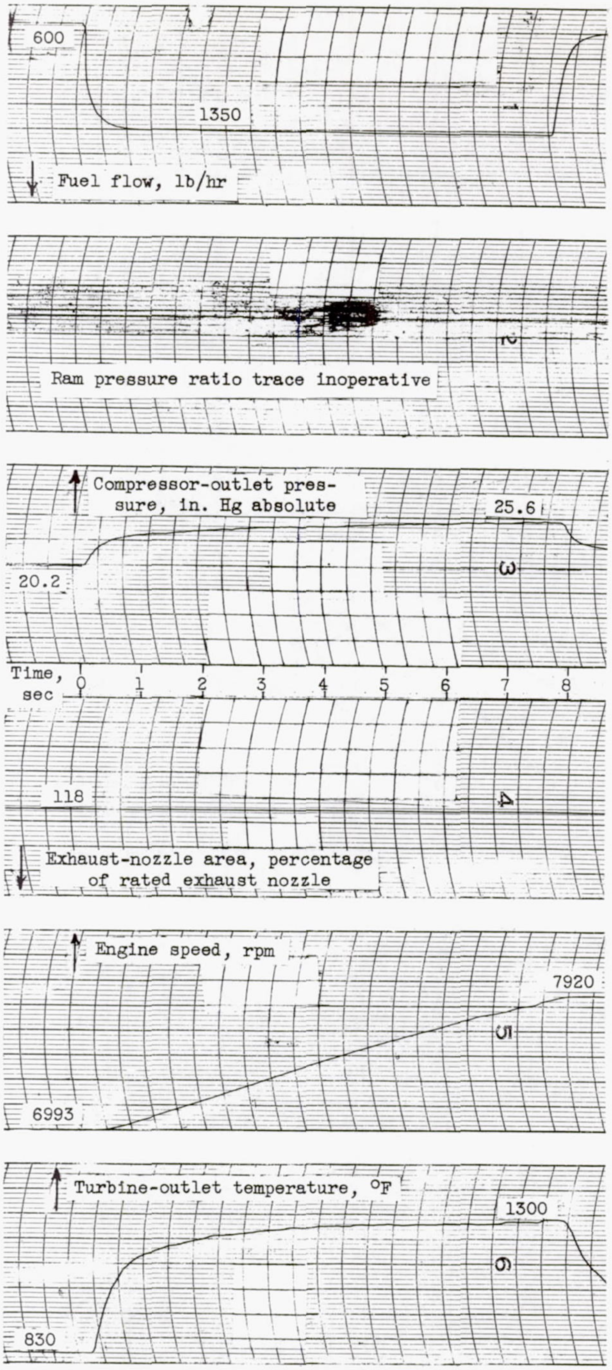
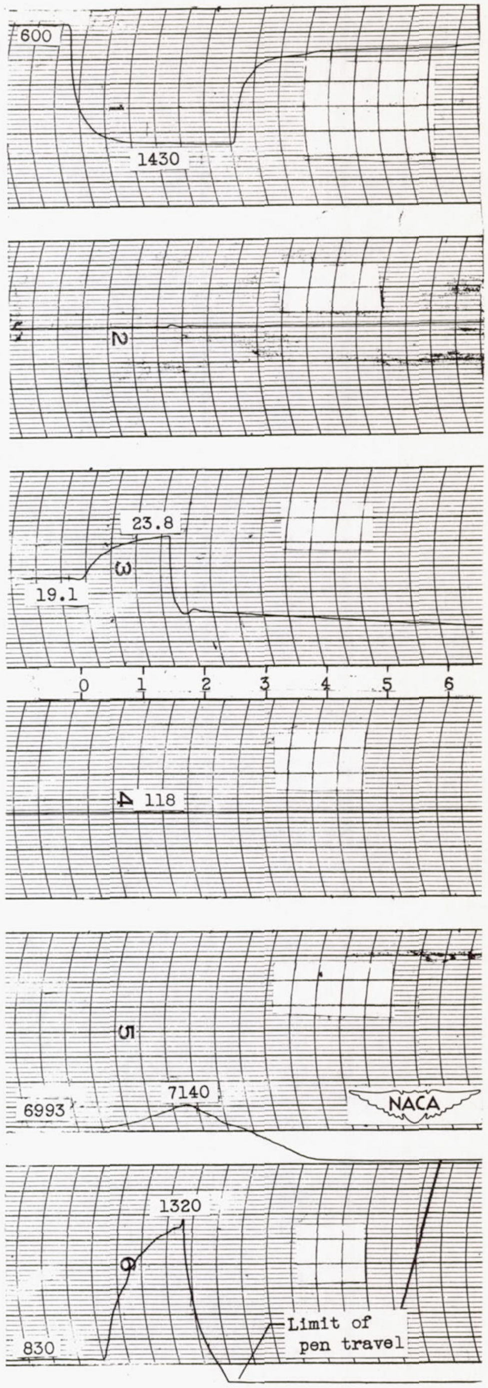


Figure 13. - Relation of compressor steady-state operating region to stall and unstall limits for J47D (RX1-3) engine.



(a) Successful acceleration.



(b) Combustor blow-out.

Figure 14. - Behavior of several engine variables during successful and unsuccessful accelerations which resulted from step increase in fuel flow. Altitude, 45,000 feet; flight Mach number, 0.19.

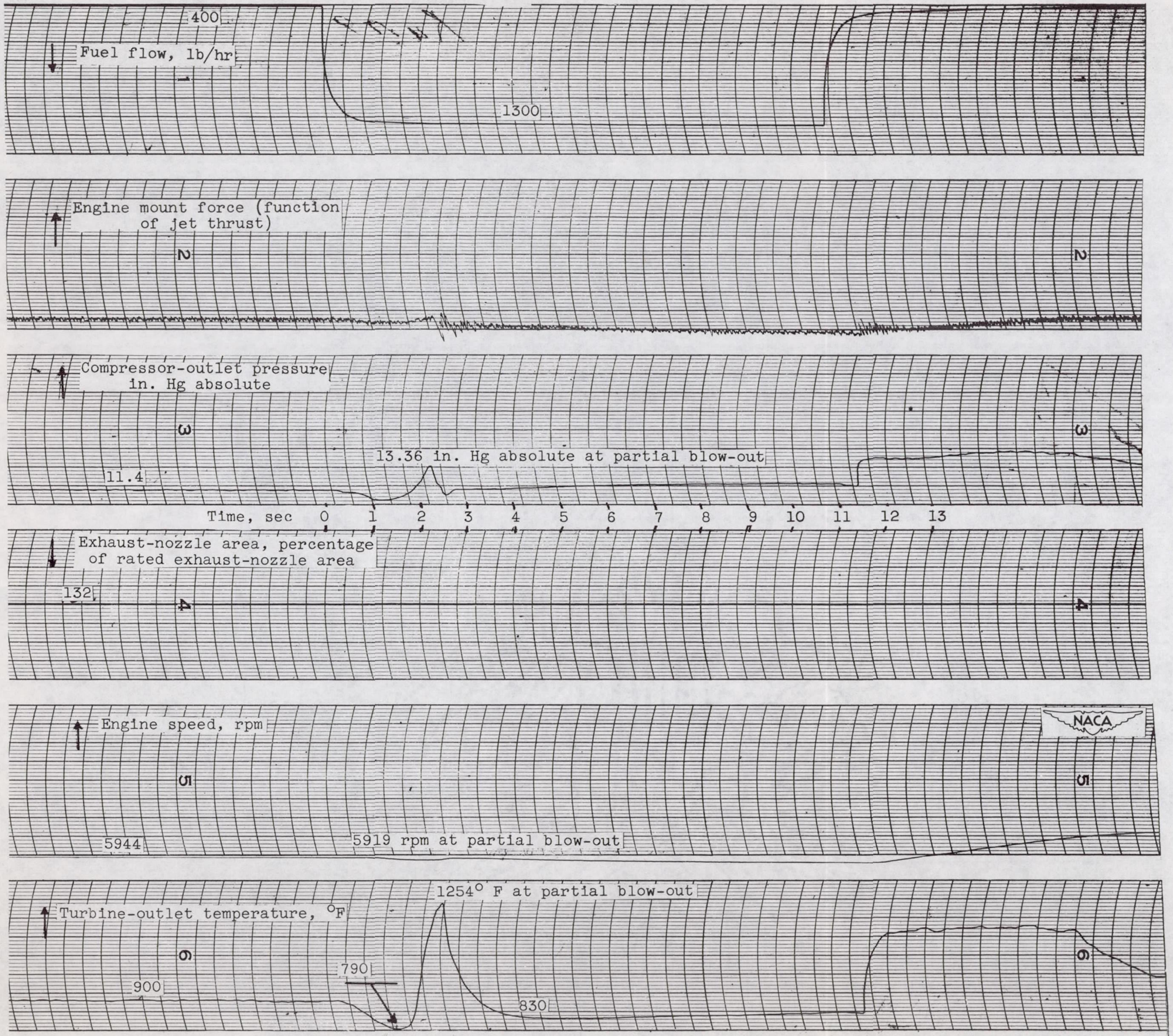


Figure 15. - Behavior of several engine variables during acceleration which resulted from step increase in fuel flow showing phenomenon of incomplete combustor blow-out. Altitude; 45,000 feet, flight Mach number, 0.19.

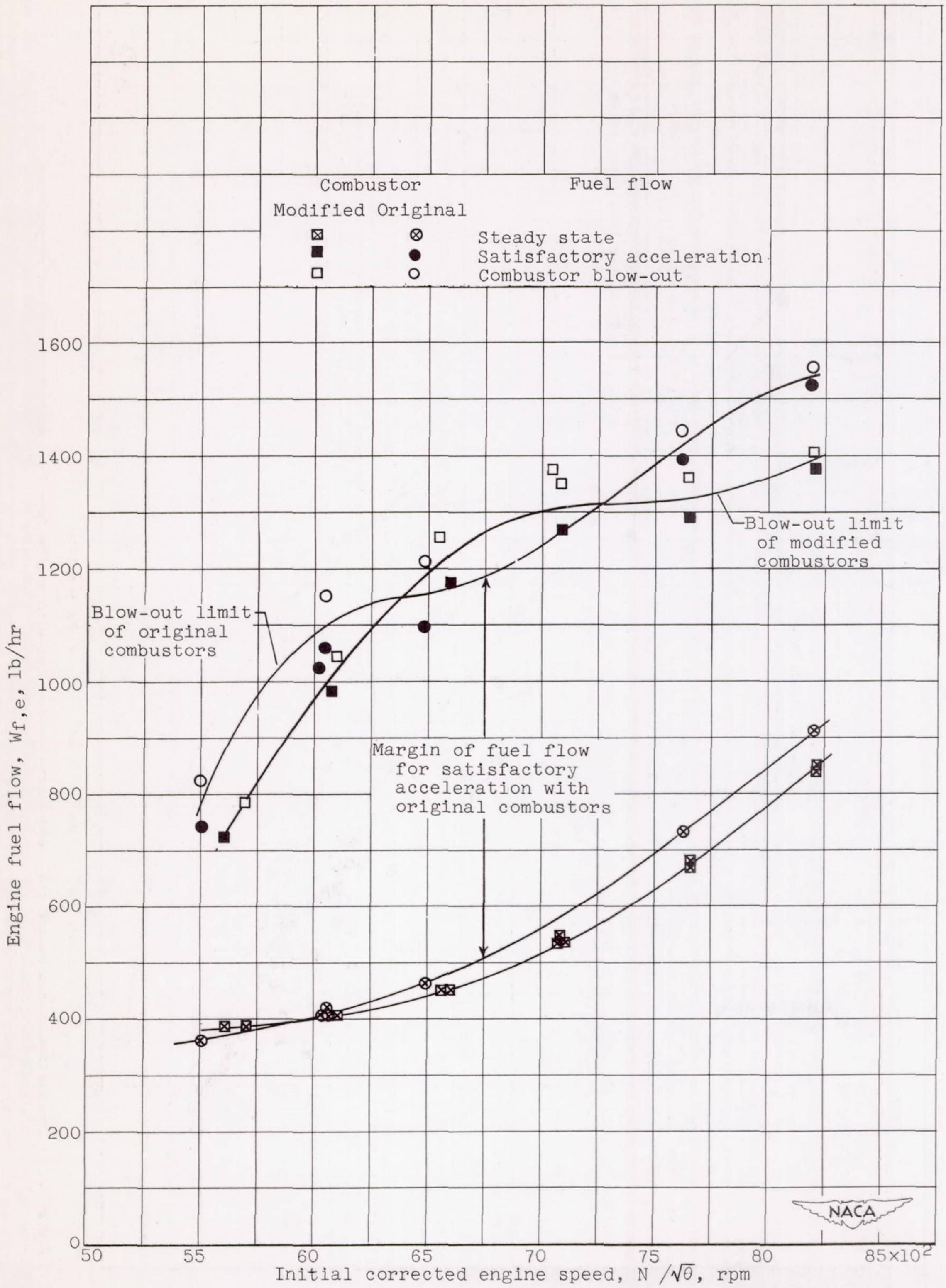


Figure 16. - Limits on engine fuel flow imposed by combustor blow-out. Altitude, 45,000 feet; flight Mach number, 0.19.

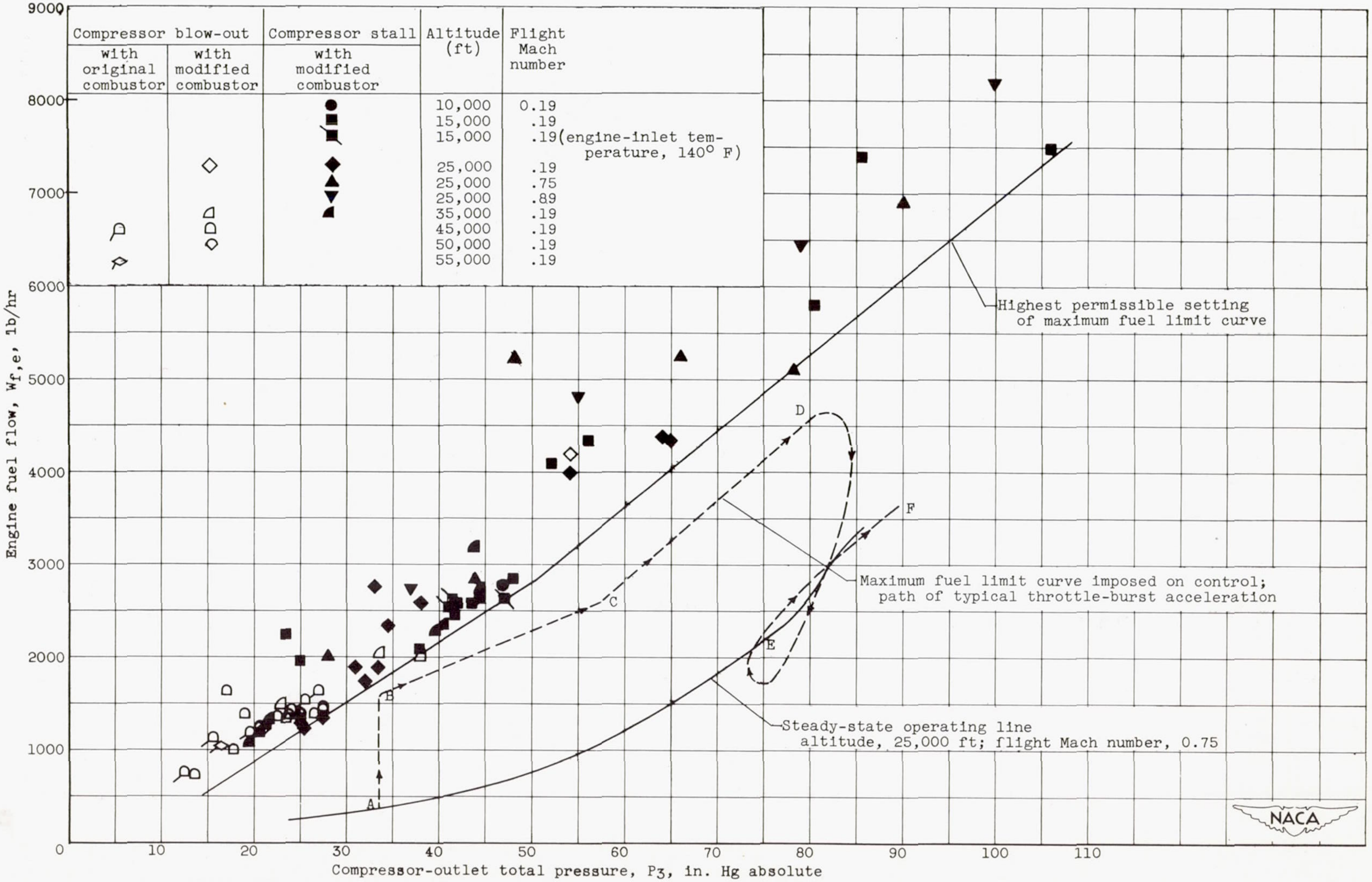


Figure 17. - Correlation of compressor stall and combustor blow-out with engine fuel flow and compressor-outlet pressure.

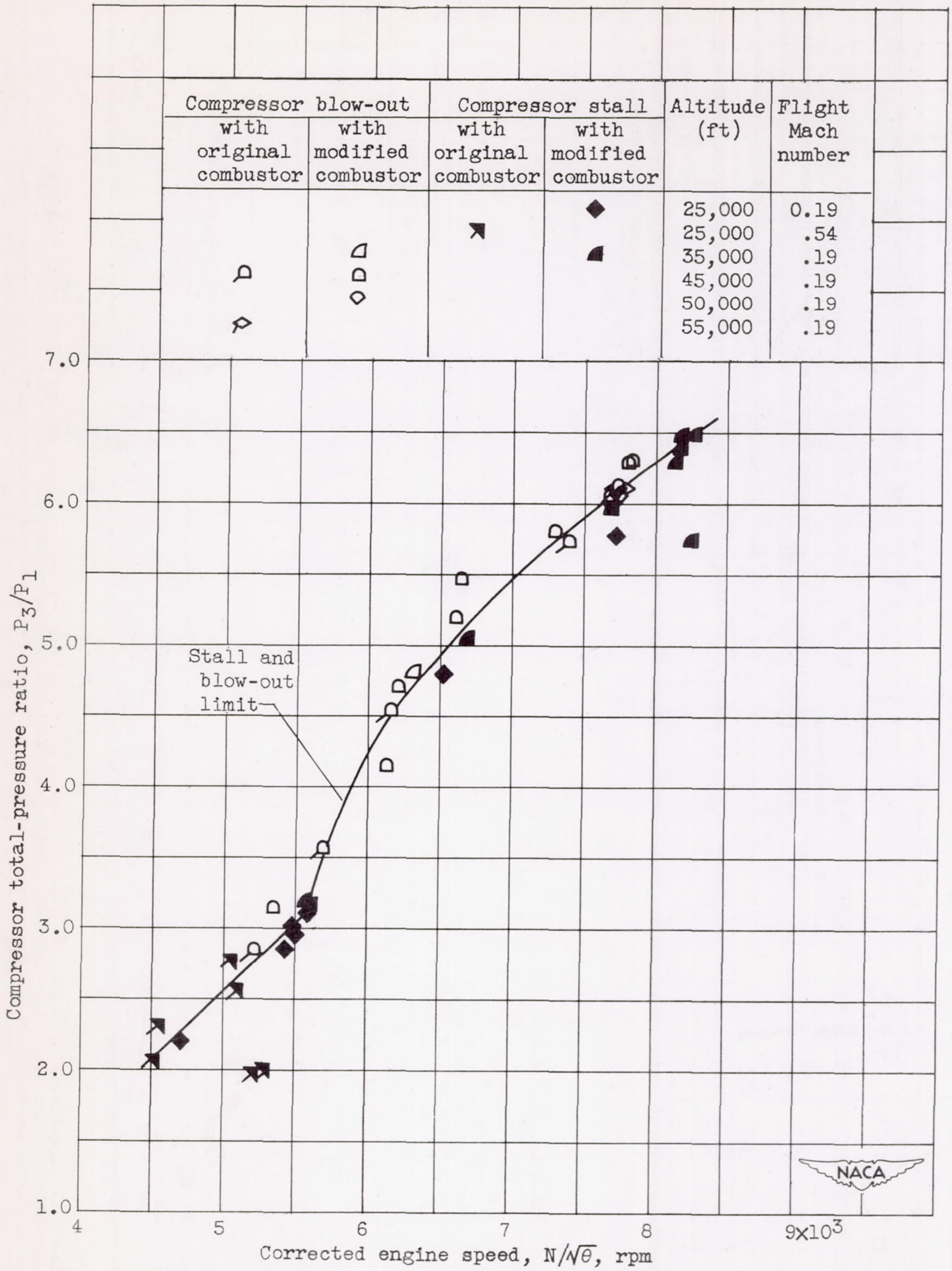


Figure 18. - Correlation of compressor stall and combustor blow-out with compressor pressure ratio and corrected engine speed for J47D (RX1-1) engine.

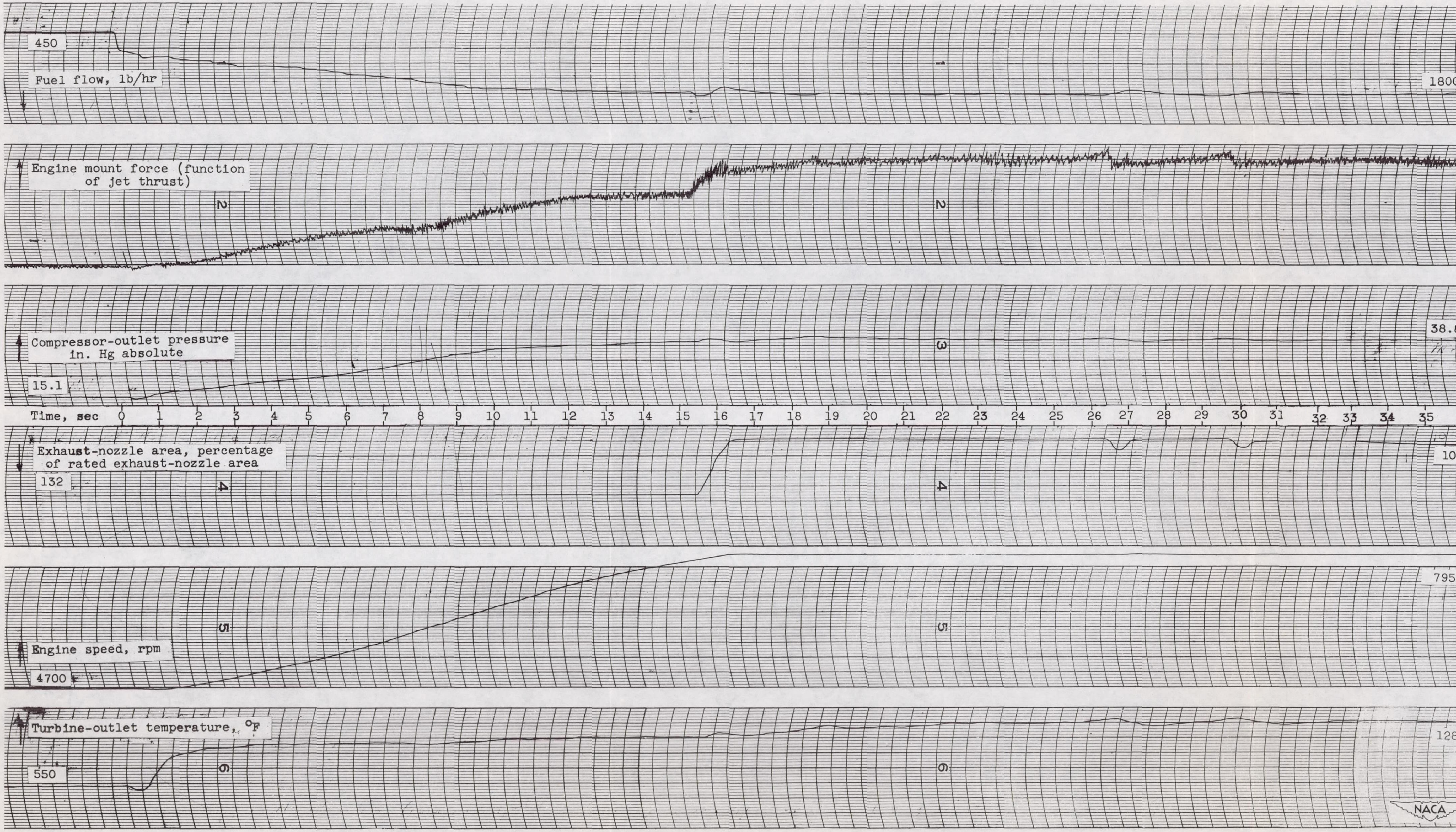
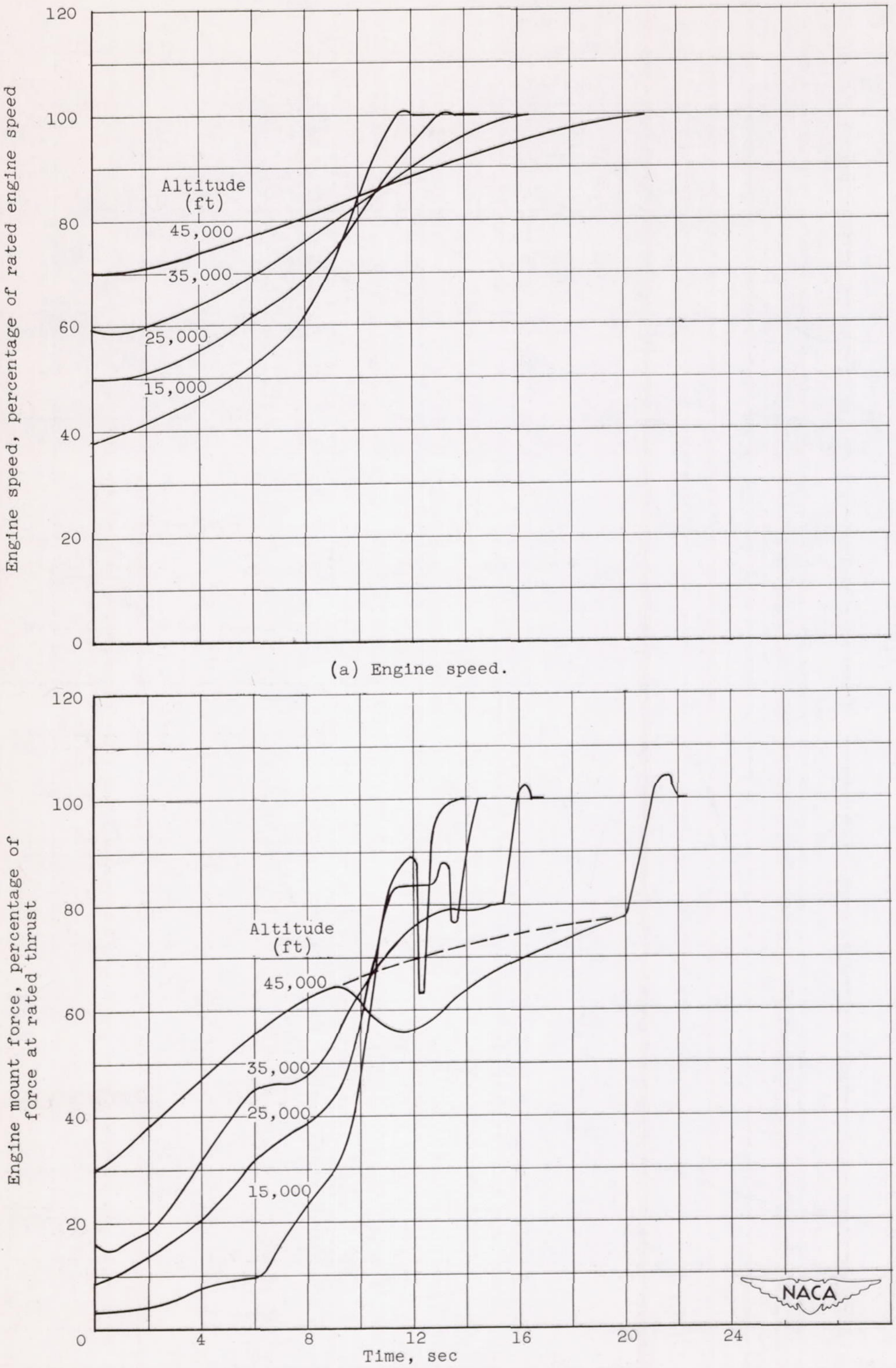


Figure 19. - Throttle-burst acceleration from idle speed to full dry thrust. Altitude, 35,000 feet; flight Mach number, 0.19.

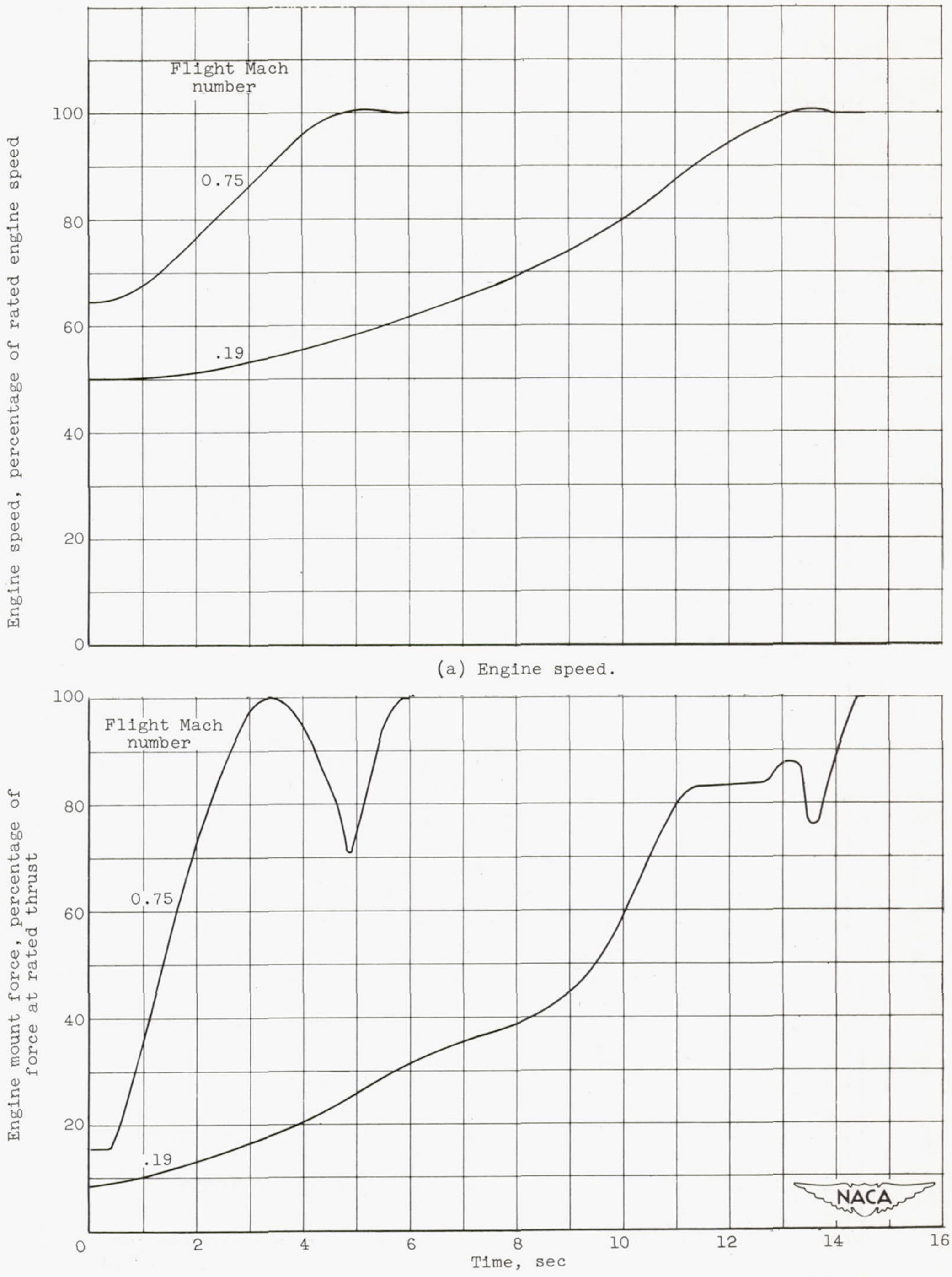
2180



(a) Engine speed.

(b) Engine mount force.

Figure 20. - Effect of altitude on variation of engine speed and engine mount force during thrust selector bursts from 10 to 90 degrees. Flight Mach number, 0.19.

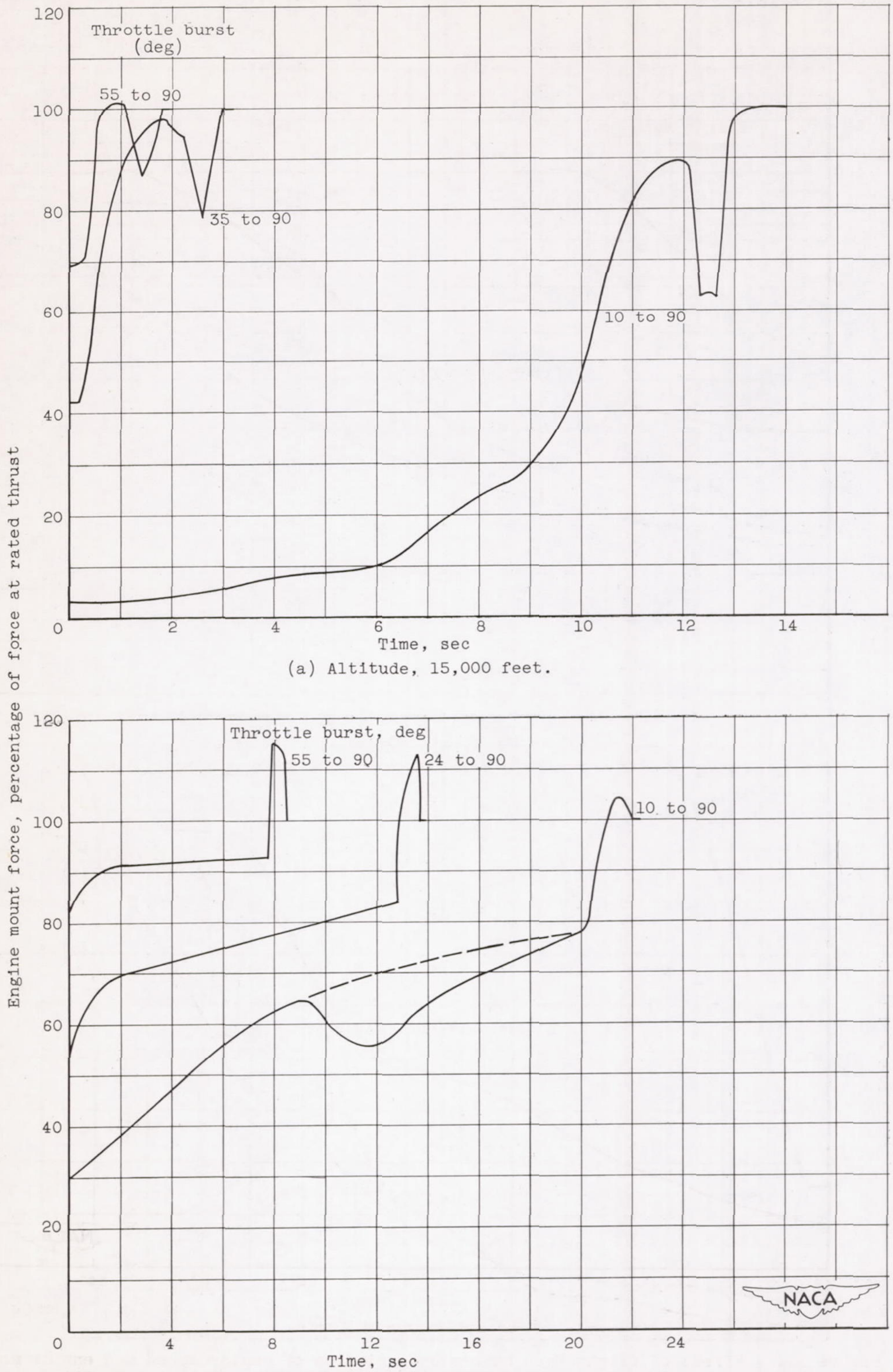


(b) Engine mount force.

Figure 21. - Effect of flight Mach number on variation of engine speed and engine mount force during thrust selector bursts from 10 to 90 degrees. Altitude, 25,000 feet.

2180

2180

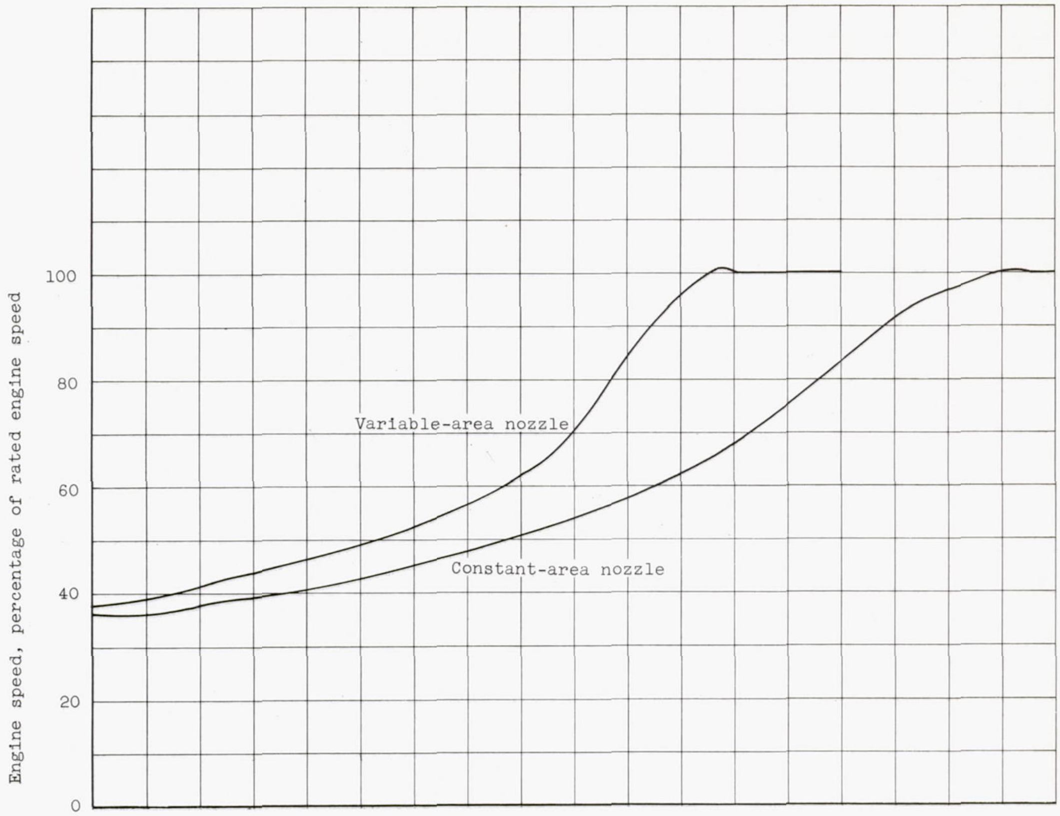


(a) Altitude, 15,000 feet.

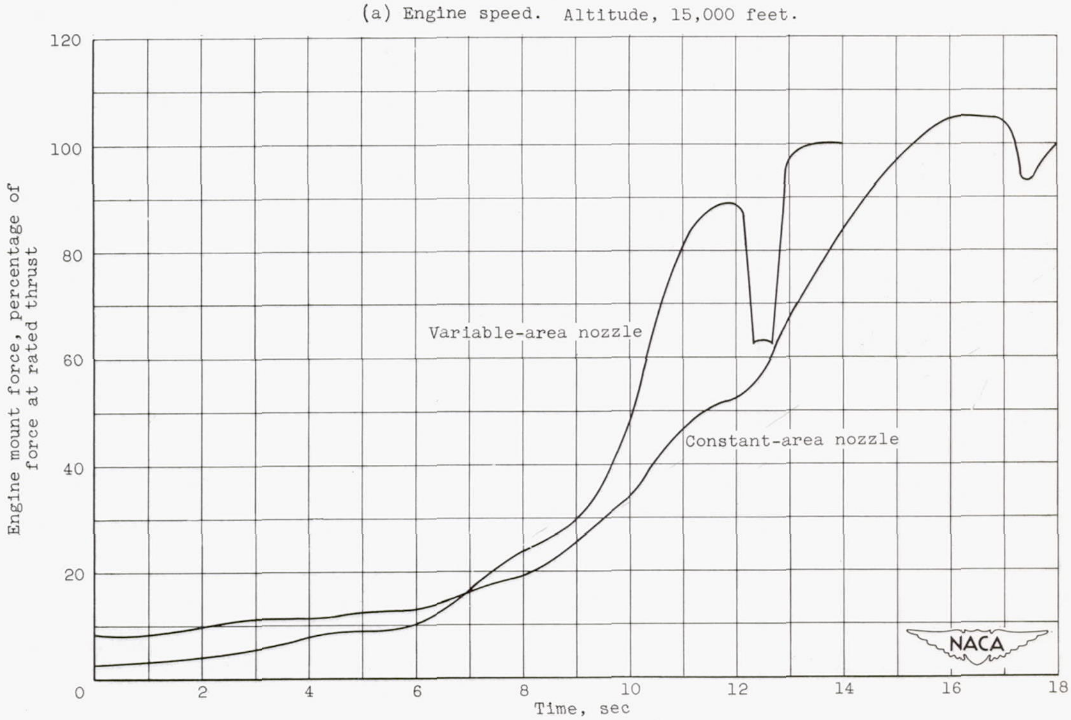
(b) Altitude, 45,000 feet.

Figure 22. - Comparison of acceleration for throttle bursts from three initial thrust-selector positions to 90 degrees. Flight Mach number, 0.19.

2180



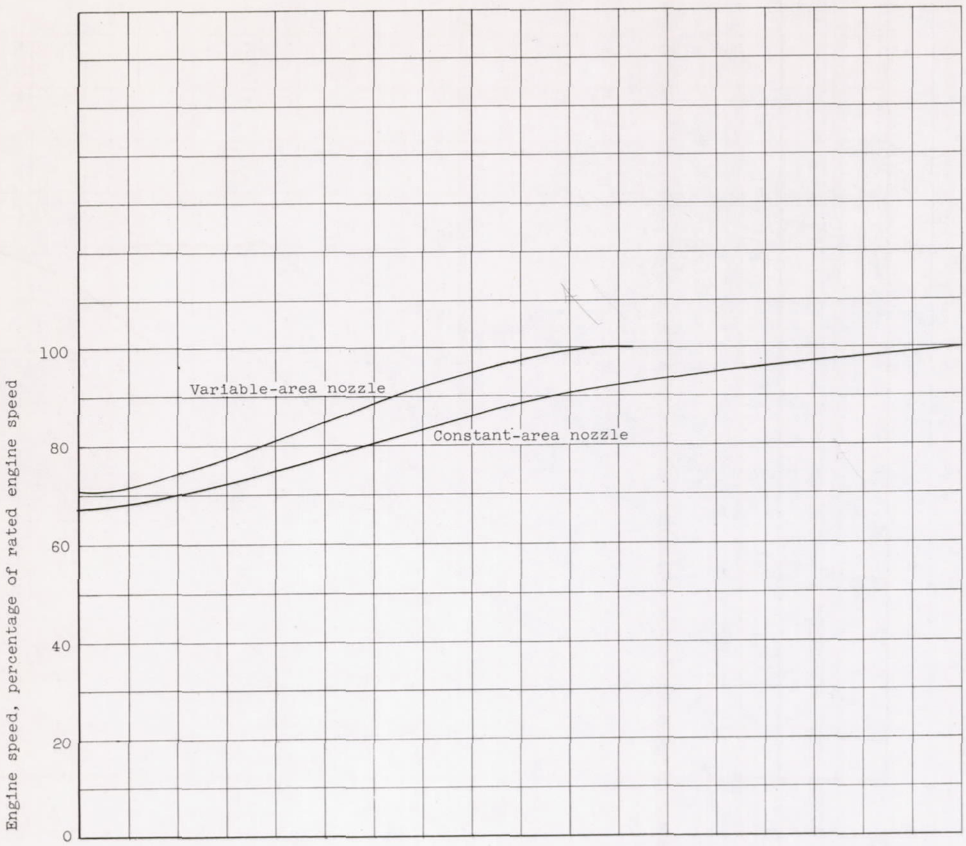
(a) Engine speed. Altitude, 15,000 feet.



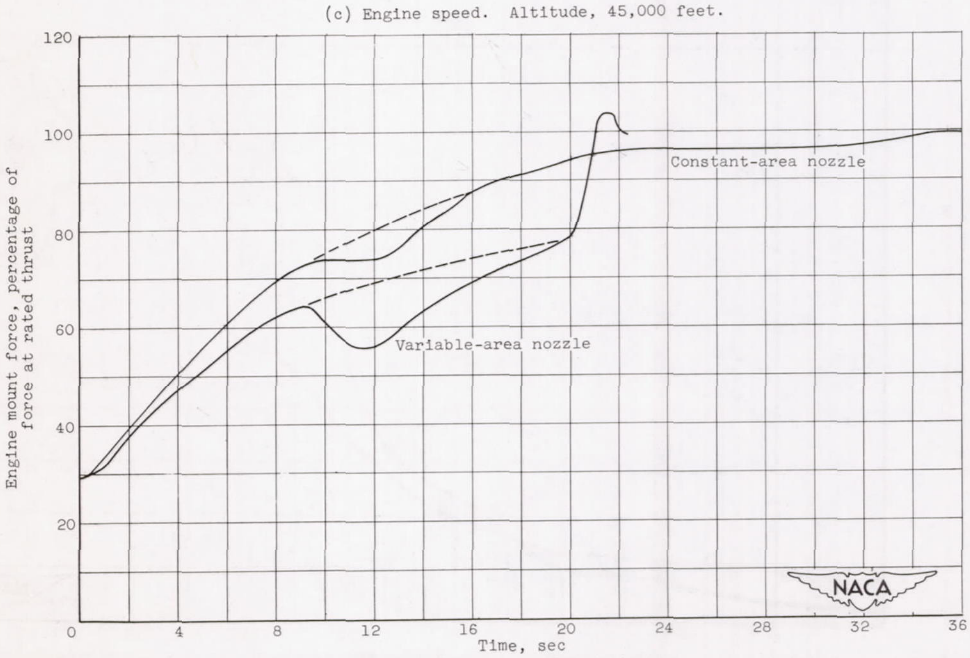
(b) Engine mount force. Altitude, 15,000 feet.

Figure 23. - Comparison of acceleration times with variable-area nozzle and constant-area nozzle for thrust selector bursts from 10 to 90 degrees. Flight Mach number, 0.19.

2180



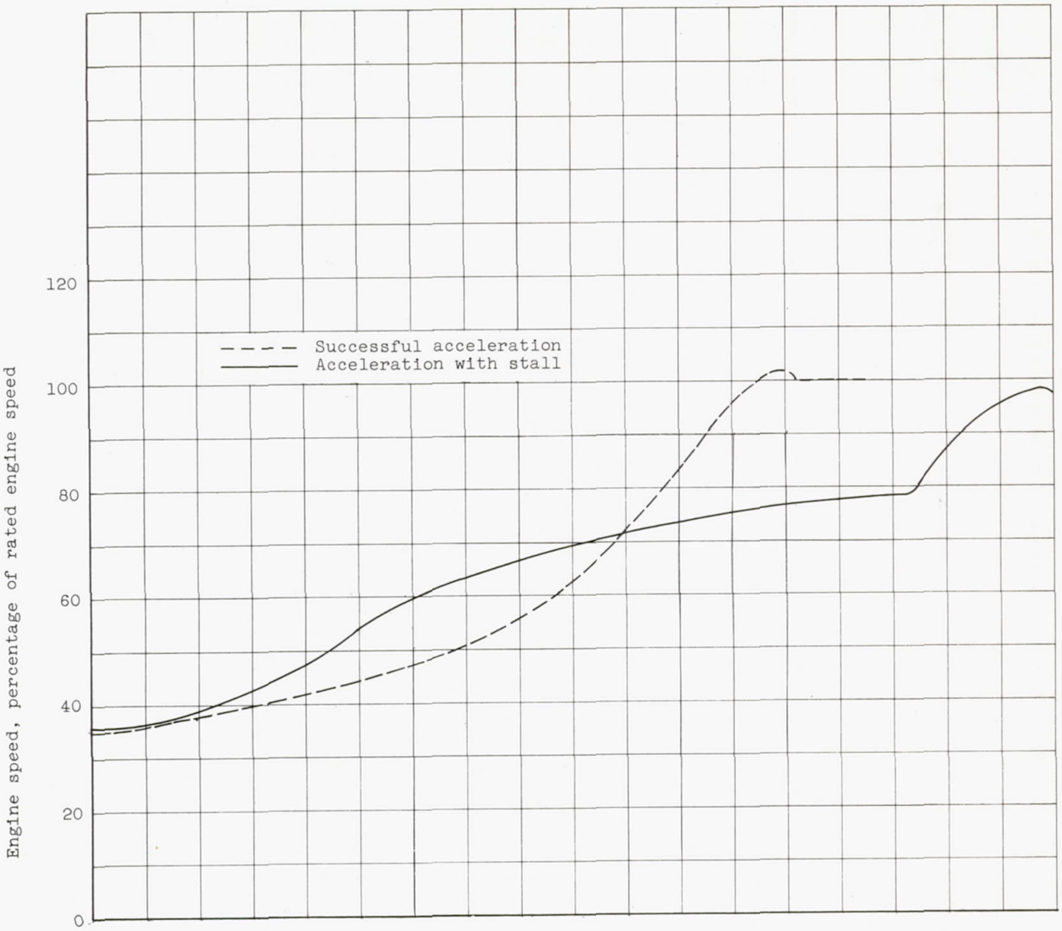
(c) Engine speed. Altitude, 45,000 feet.



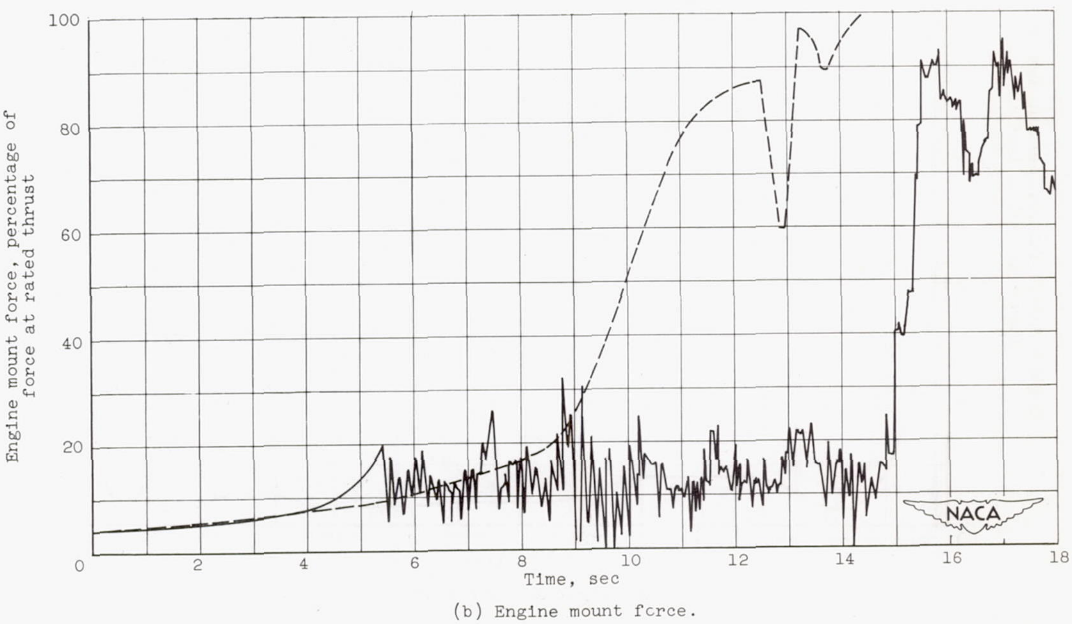
(d) Engine mount force. Altitude, 45,000 feet.

Figure 23. - Concluded. Comparison of acceleration times with variable-area nozzle and constant-area nozzle for thrust selector bursts from 10 to 90 degrees. Flight Mach number, 0.19.

2180



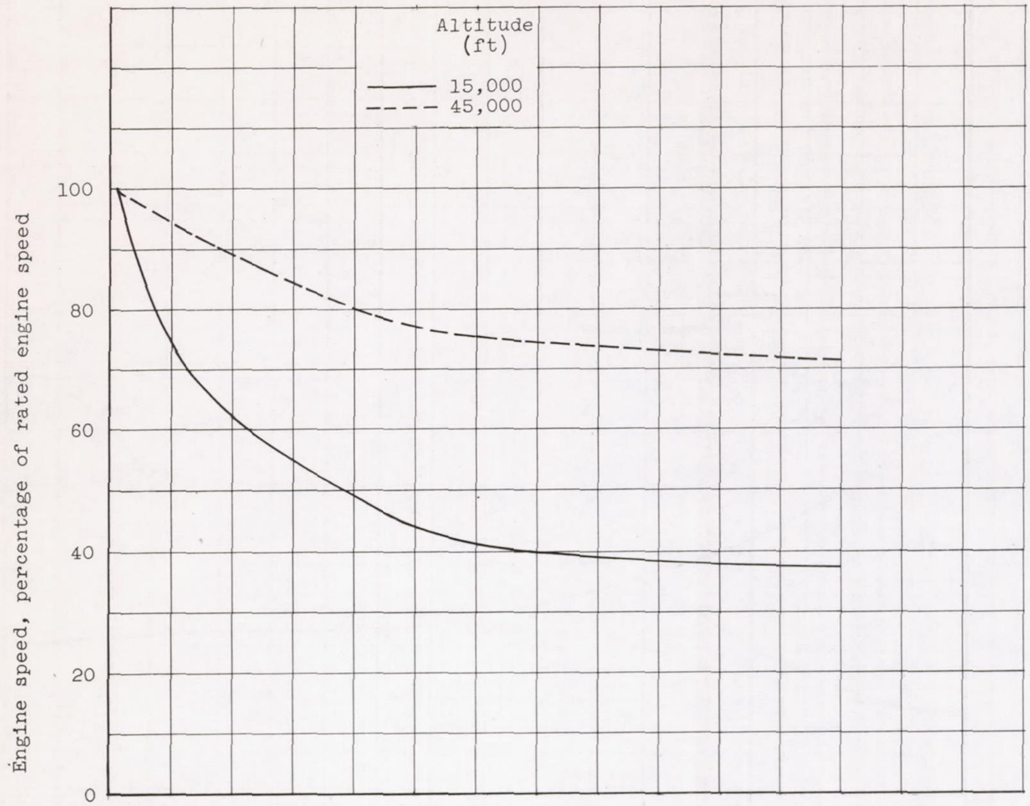
(a) Engine speed.



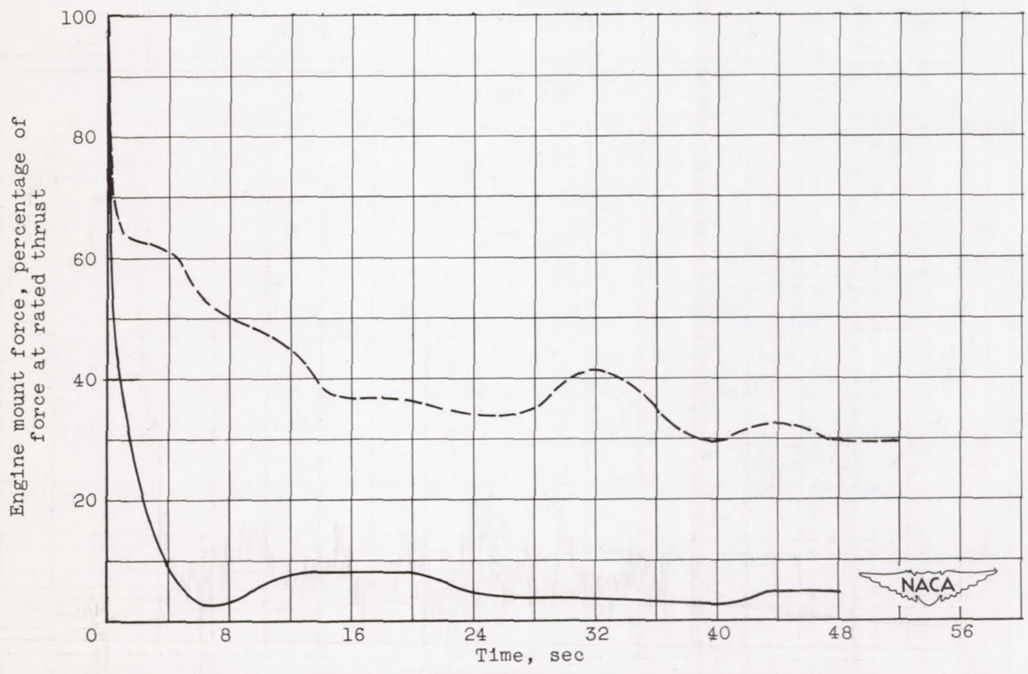
(b) Engine mount force.

Figure 24. - Comparison of thrust selector bursts from 10 to 90 degrees with and without stall. Altitude, 10,000 feet; flight Mach number, 0.19.

2180



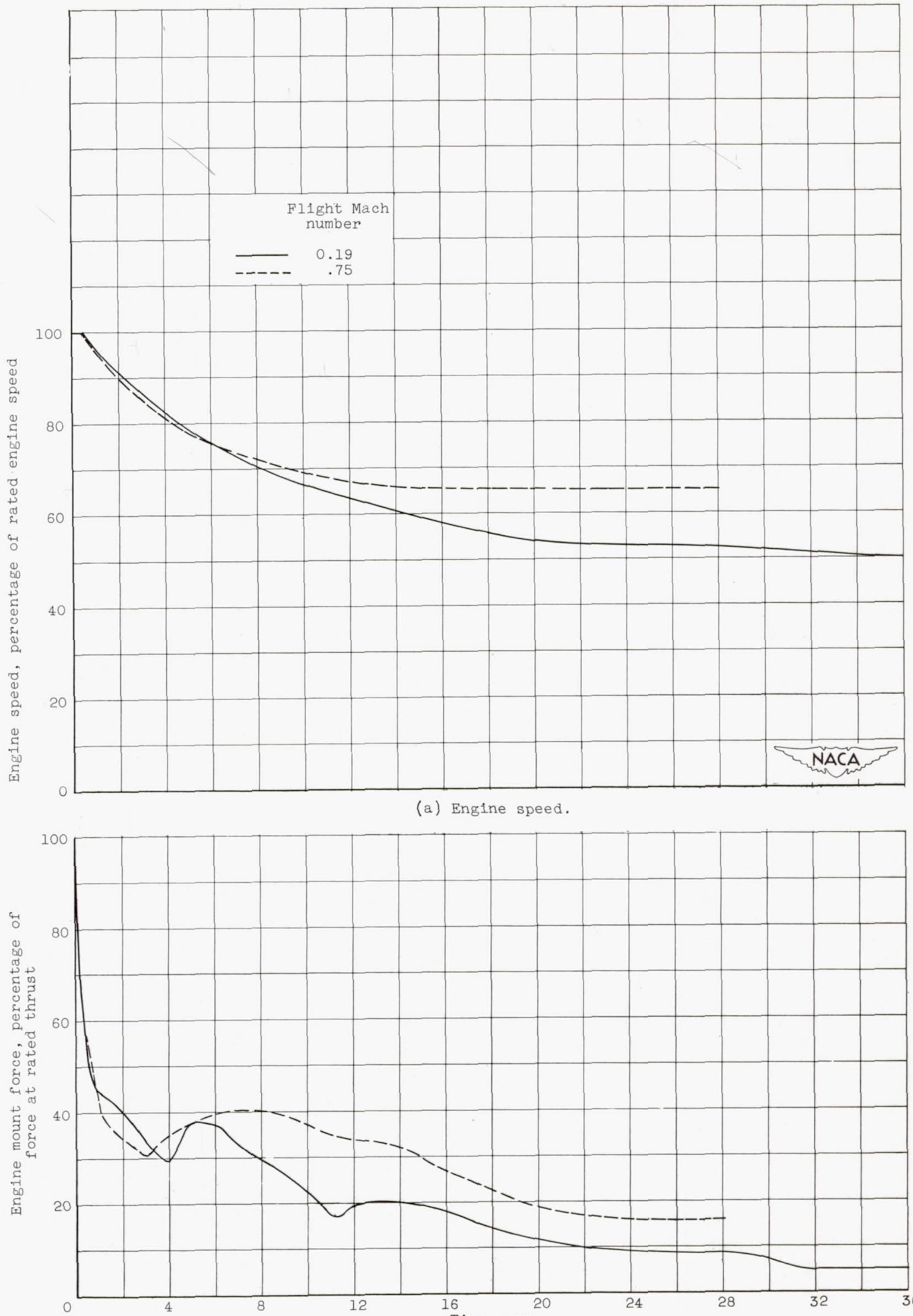
(a) Engine speed.



(b) Engine mount force.

Figure 25. - Effect of altitude on variation of engine speed and engine mount force during thrust selector chops from 90 to 10 degrees. Flight Mach number, 0.19.

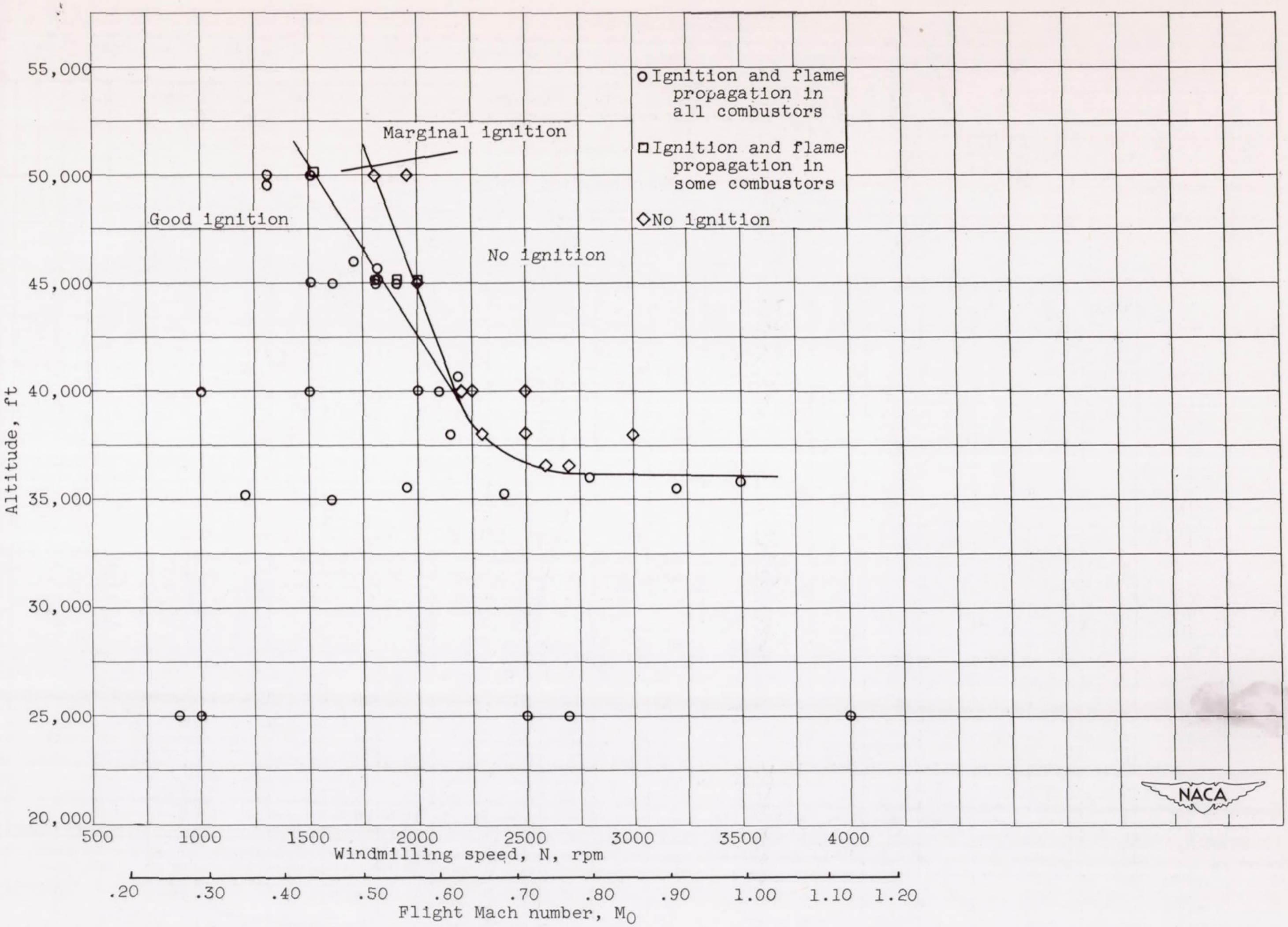
2180



(a) Engine speed.

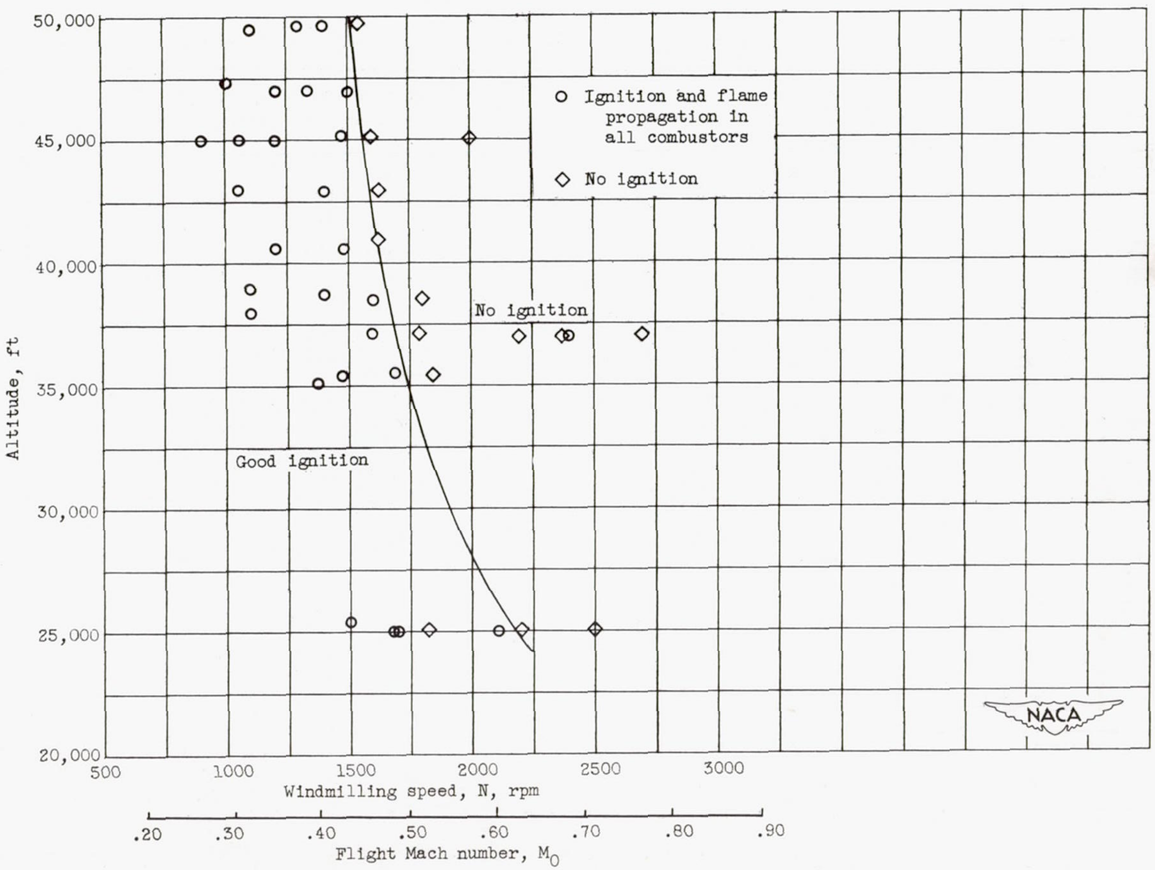
(b) Engine mount force.

Figure 26. - Effect of flight Mach number on variation of engine speed and engine mount force during thrust selector chops from 90 to 10 degrees. Altitude, 25,000 feet.



(a) Reid vapor pressure, 7 pounds.

Figure 27. - Altitude starting characteristics using MIL-F-5624 (AN-F-58) fuel.



(b) Reid vapor pressure, 1 pound.

Figure 27. - Concluded. Altitude starting characteristics using MIL-F-5624 (AN-F-58) fuel.

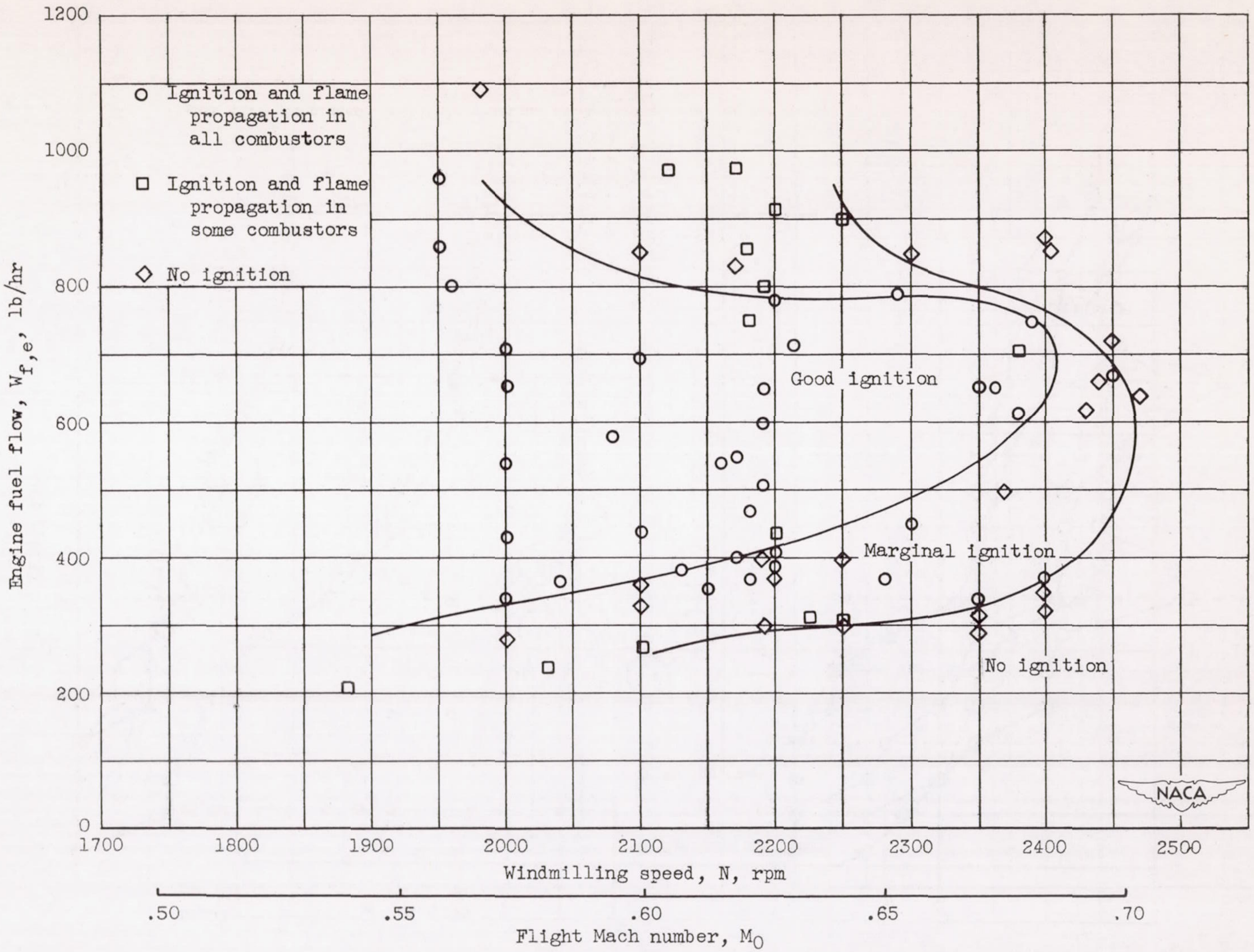
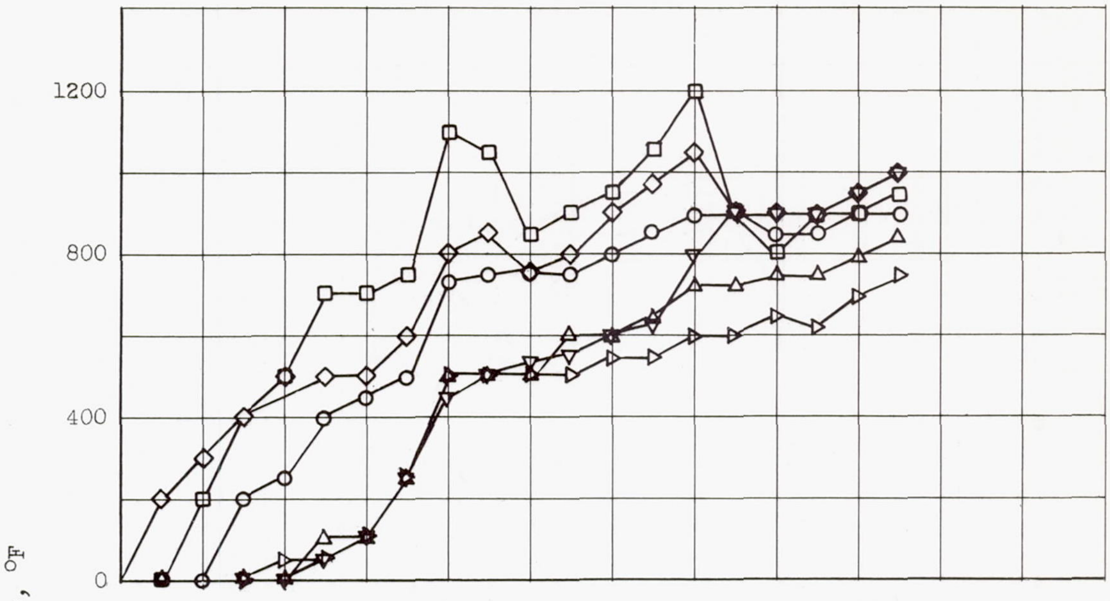
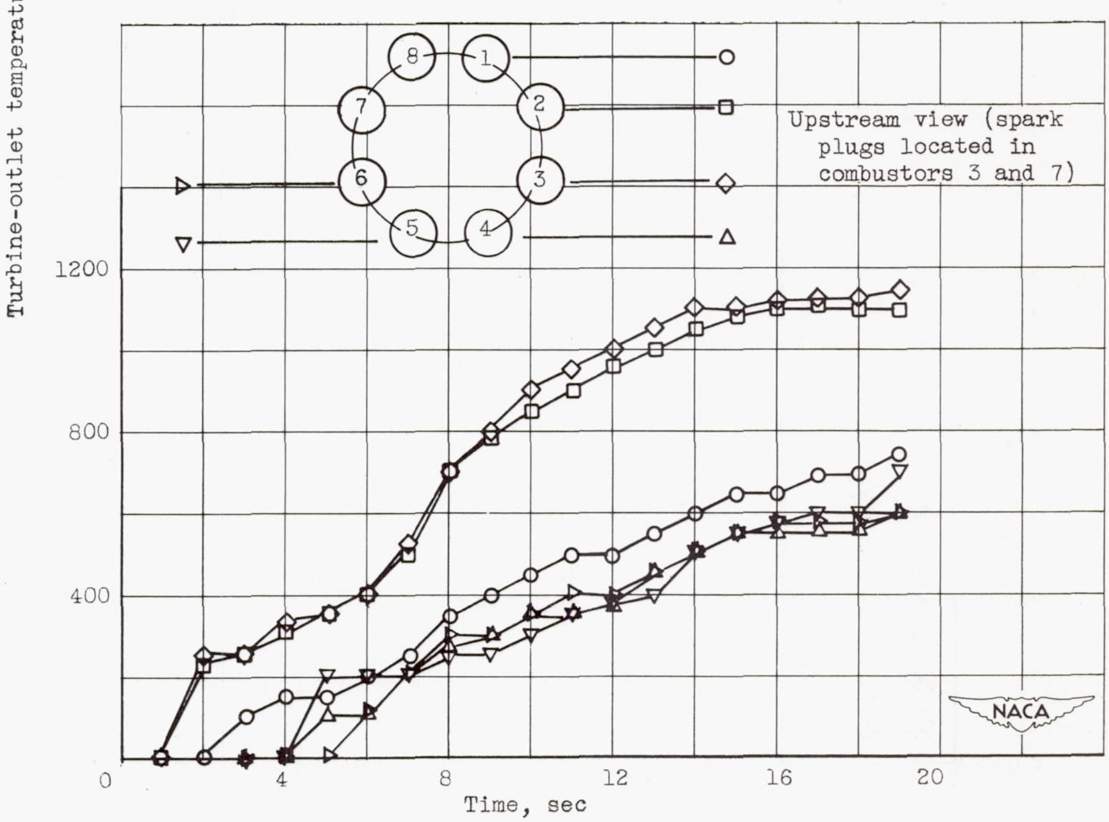


Figure 28. - Effect of engine fuel flow on altitude starting characteristics using MIL-F-5624 (AN-F-58) fuel with Reid vapor pressure of 7 pounds. Altitude, 40,000 feet.

2180



(a) Windmilling speed, 1200 rpm.



(b) Windmilling speed, 1600 rpm.

Figure 29. - Variation of turbine-outlet temperatures behind individual combustors with time for two typical starts at altitude of 35,000 feet.

2180

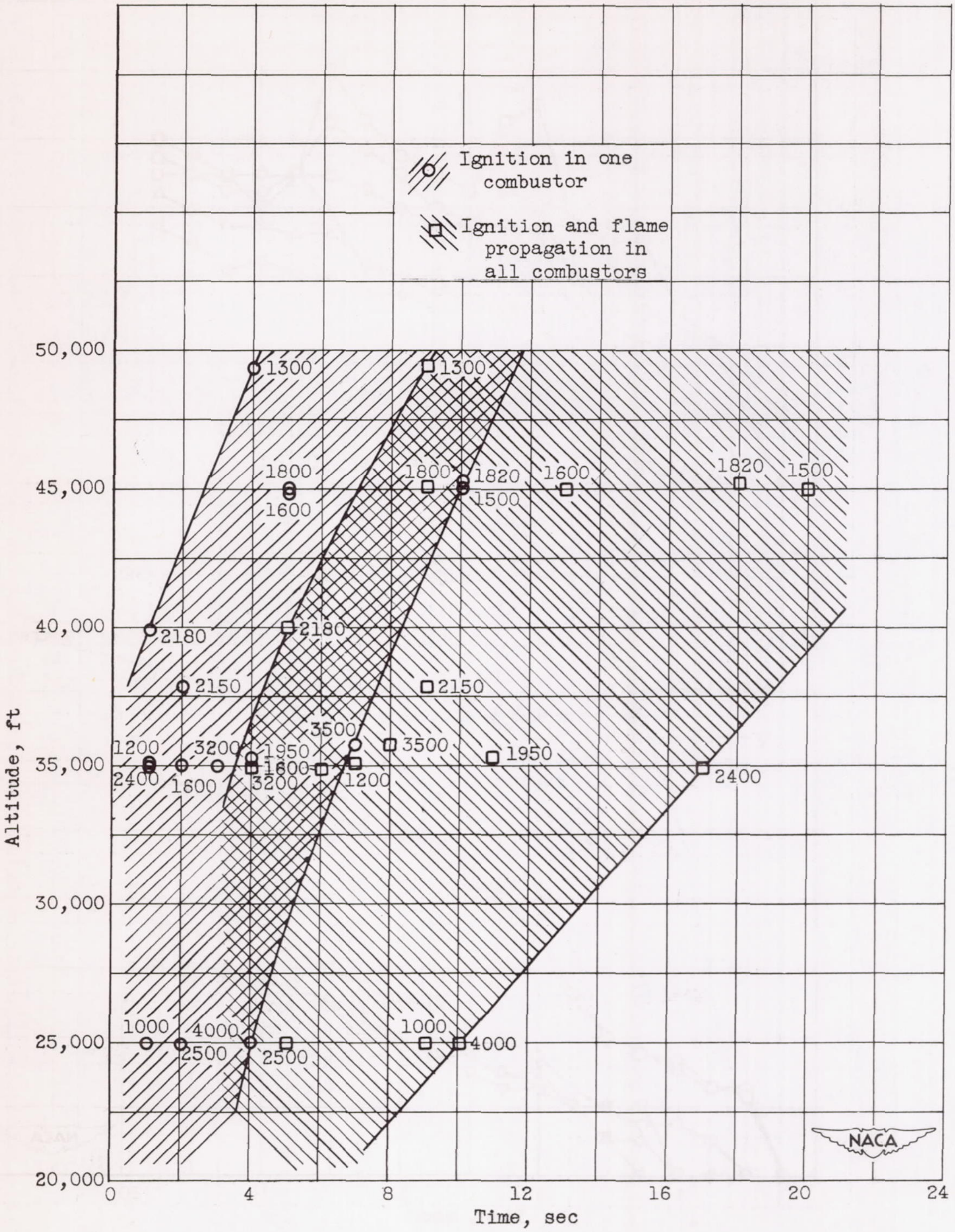


Figure 30. - Variation of time required for ignition and flame propagation with altitude for altitude starts at various windmilling speeds. Values by data points indicate initial engine speed at which ignition system turned on.

2180

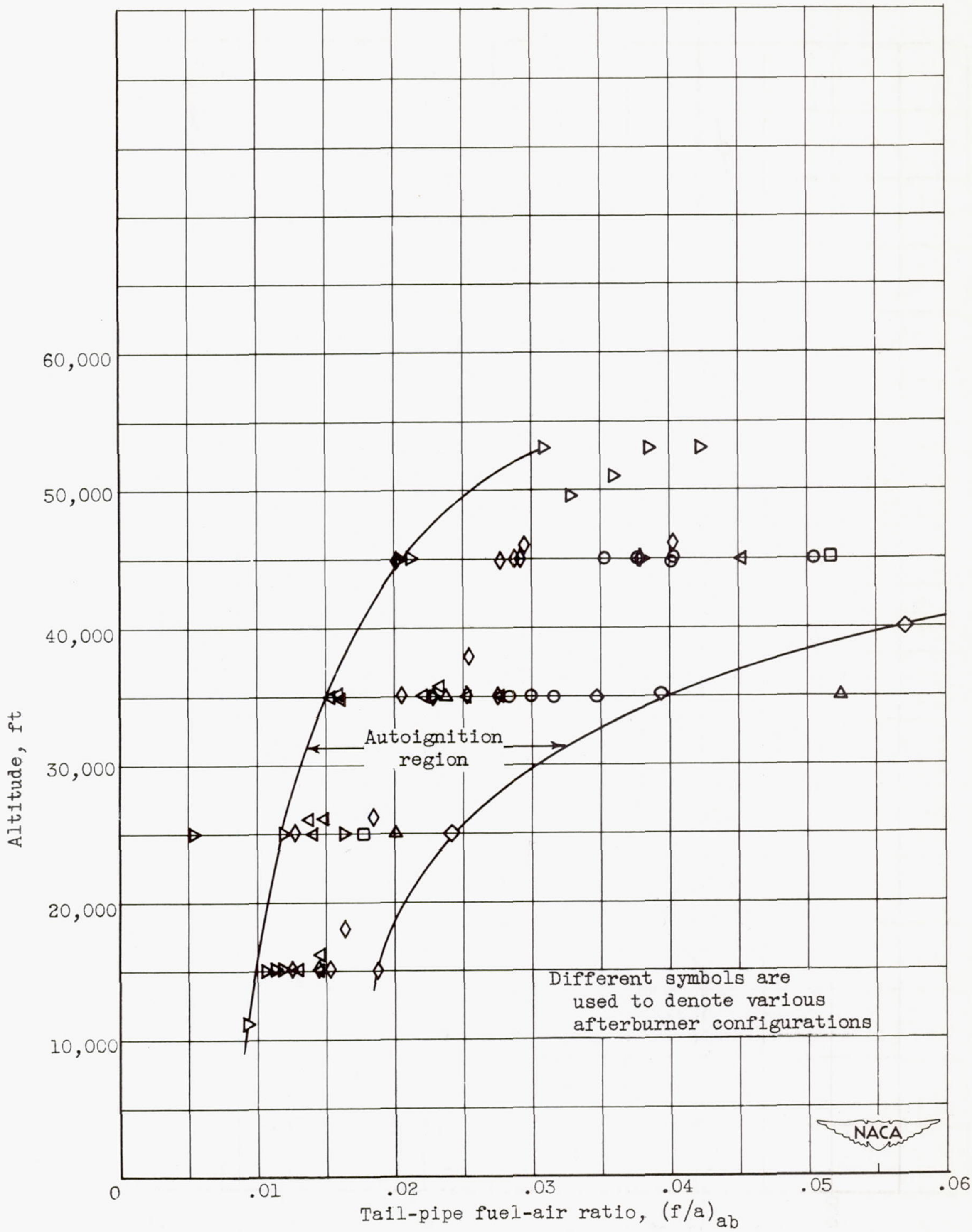


Figure 31. - Tail-pipe fuel-air ratios required for autoignition with several afterburner configurations. Fuel, MIL-F-5624 (AN-F-58); burner-inlet temperature, 1250° to 1300° F; flight Mach number, 0.19.

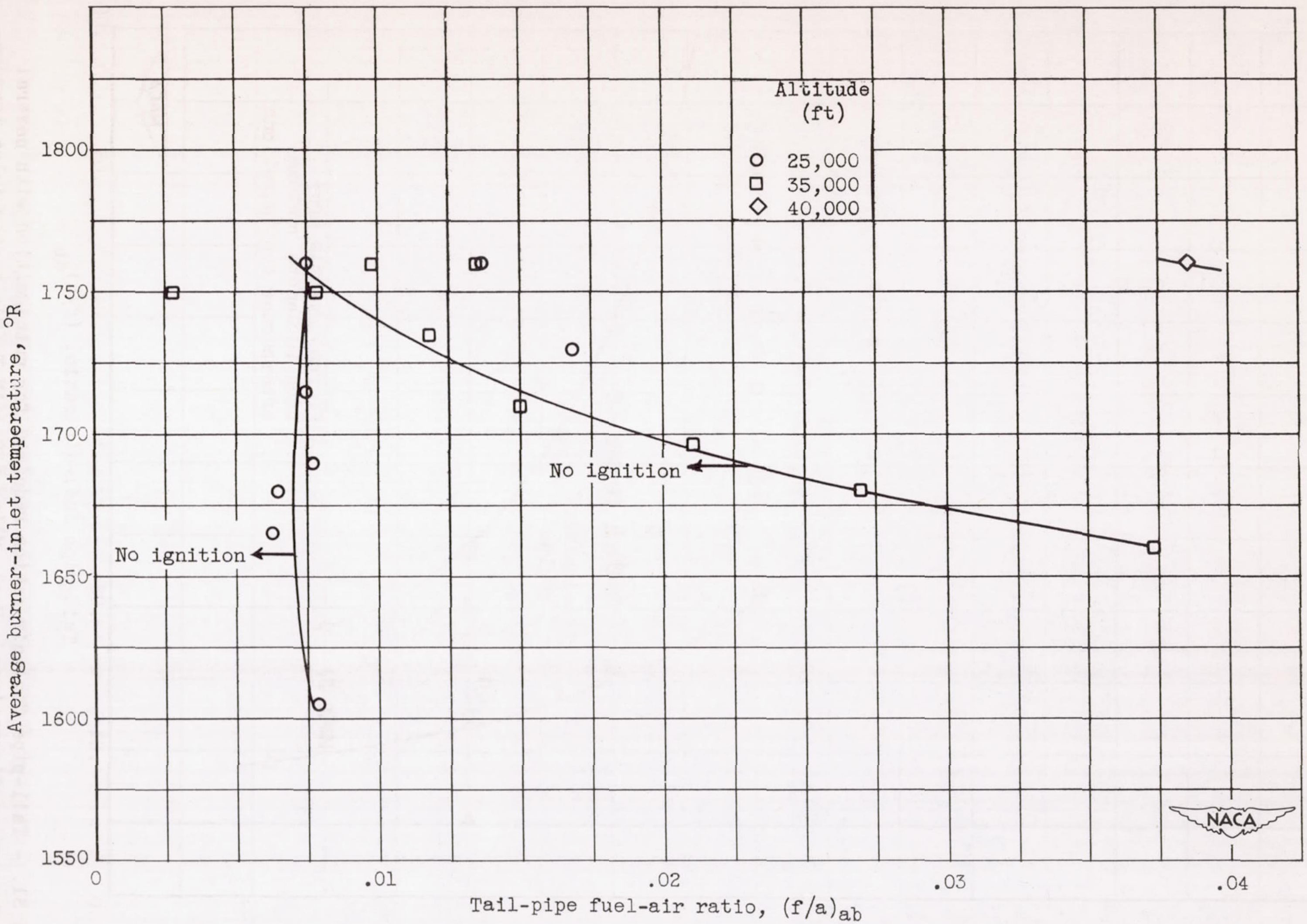


Figure 32. - Autoignition characteristics of tail-pipe burner using MIL-F-5624 (AN-F-58) fuel with Reid vapor pressure of 7 pounds. Flight Mach number, 0.19.

2180

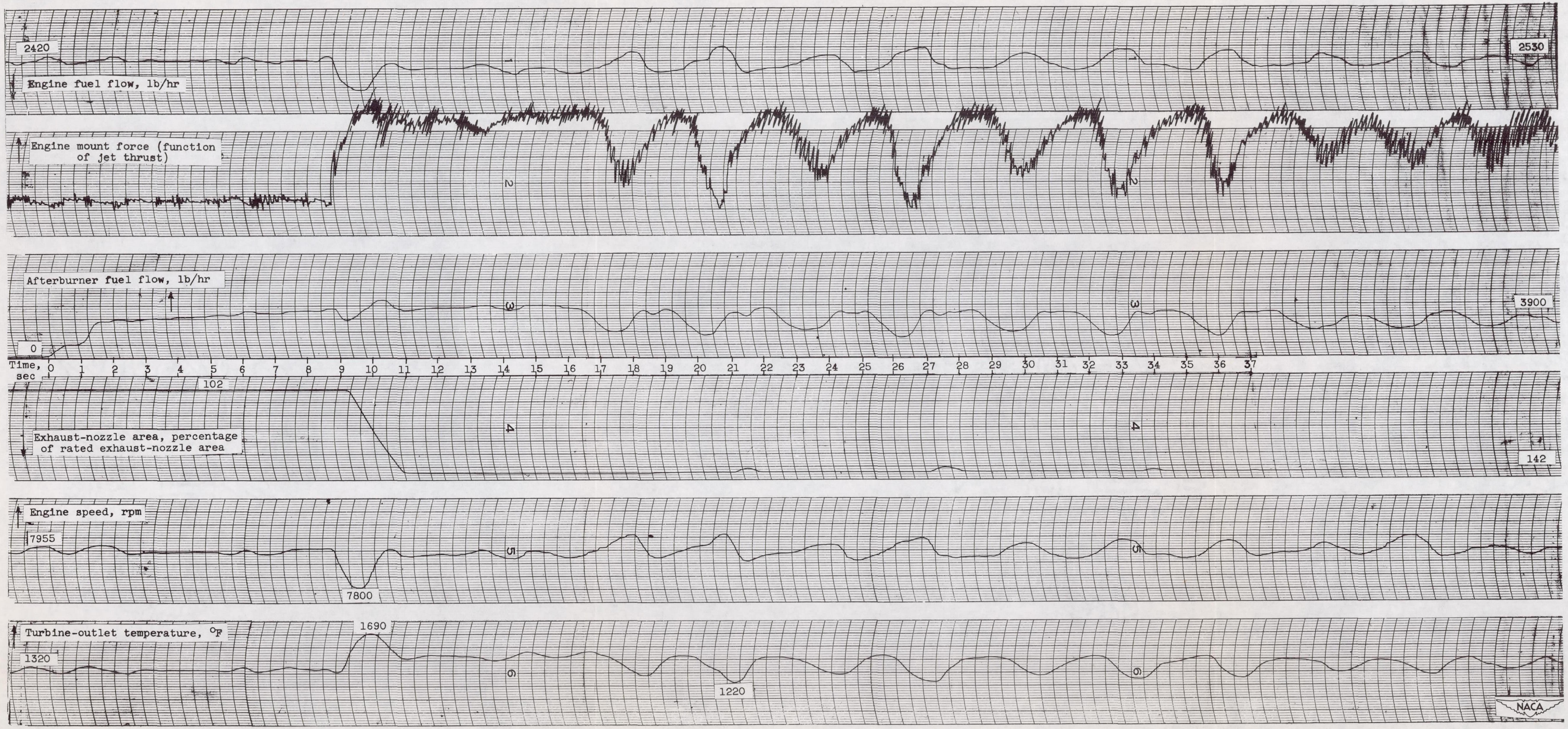


Figure 33. - Behavior of several engine variables during automatically controlled acceleration from full dry thrust to full afterburning. Altitude, 25,000 feet; flight Mach number, 0.19.

2180

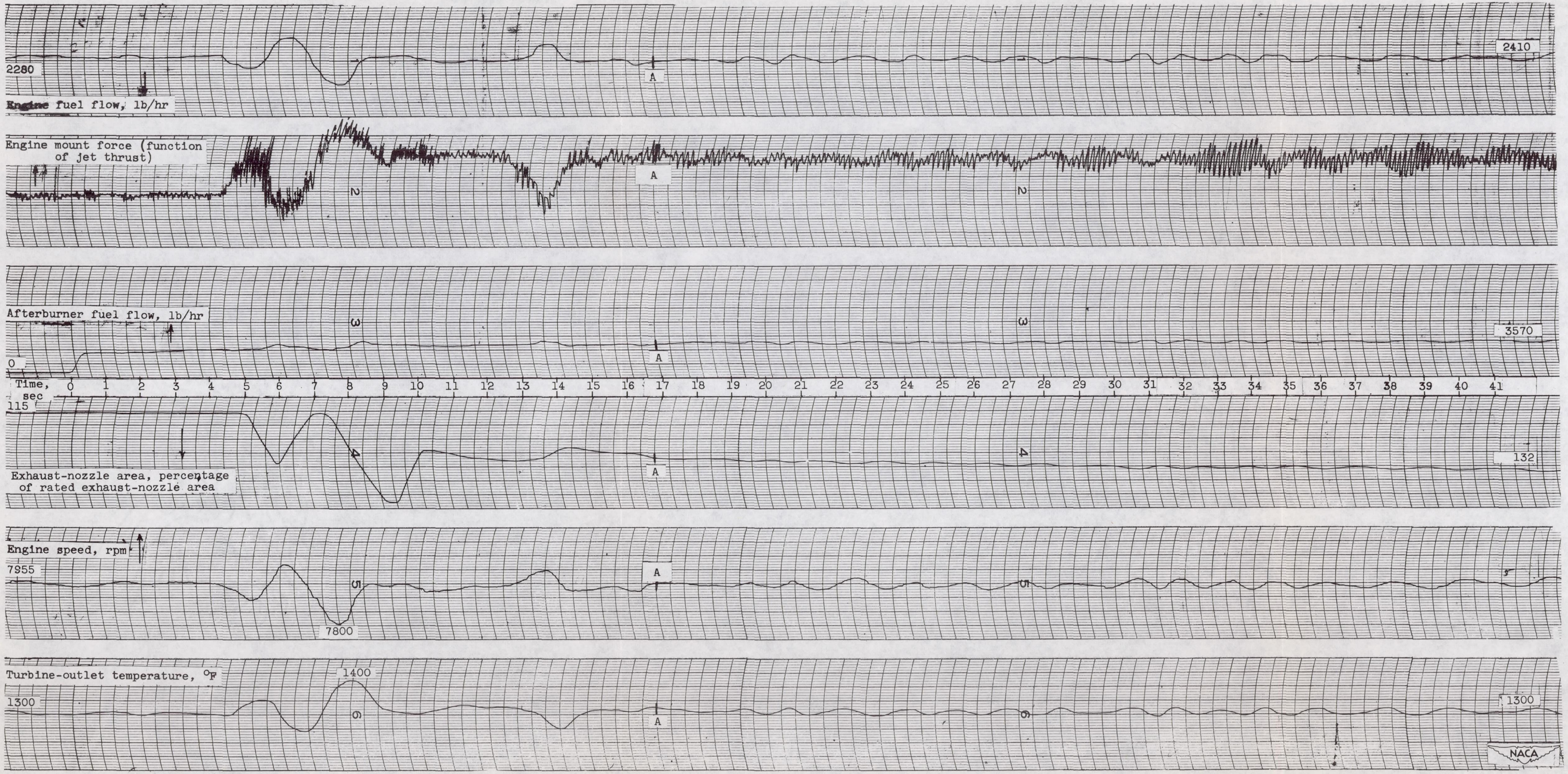


Figure 34. - Behavior of several engine variables during automatically controlled acceleration from full dry thrust to partial afterburning. Altitude, 25,000 feet; flight Mach number, 0.19.

2180

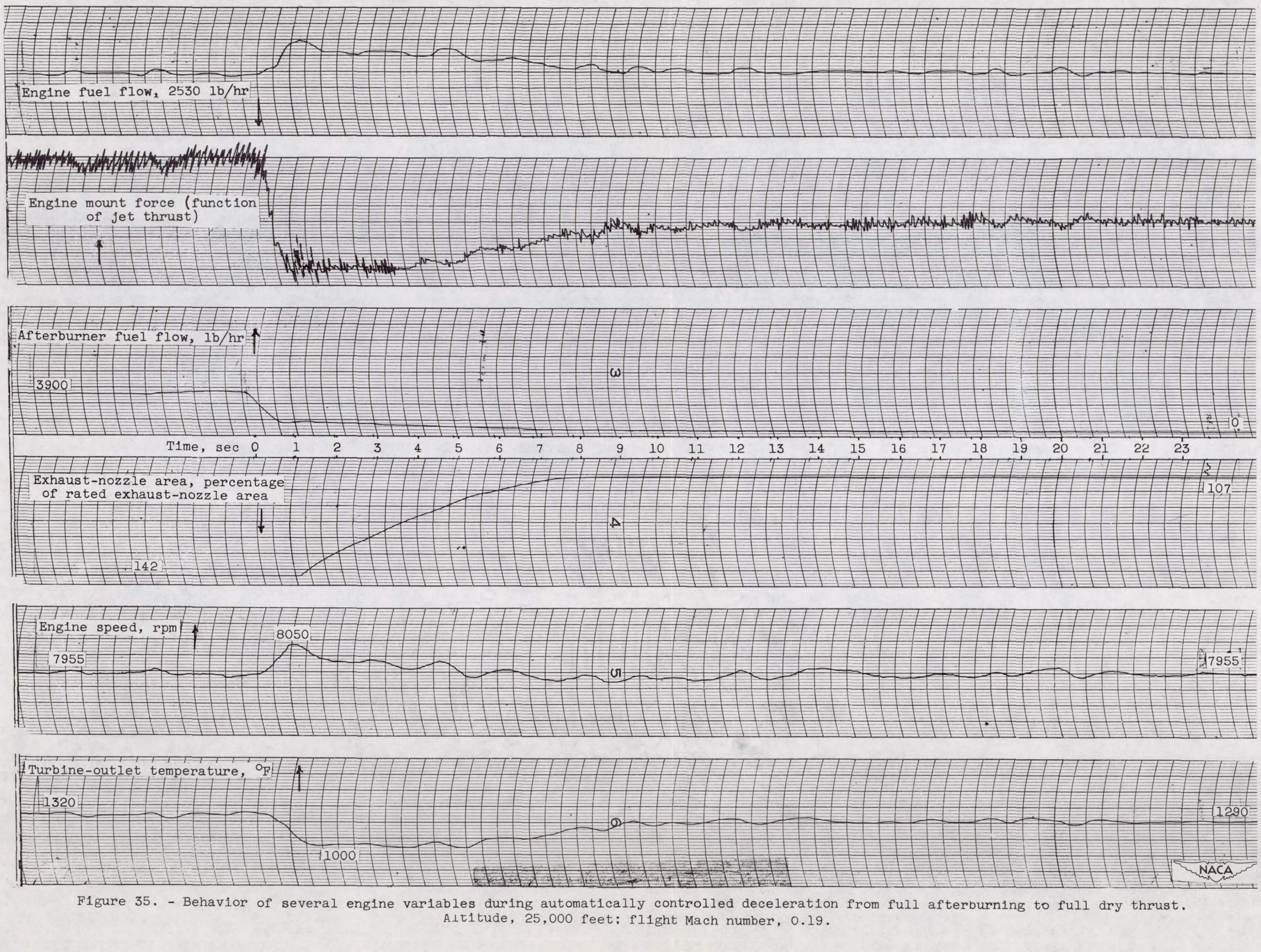


Figure 35. - Behavior of several engine variables during automatically controlled deceleration from full afterburning to full dry thrust. Altitude, 25,000 feet; flight Mach number, 0.19.

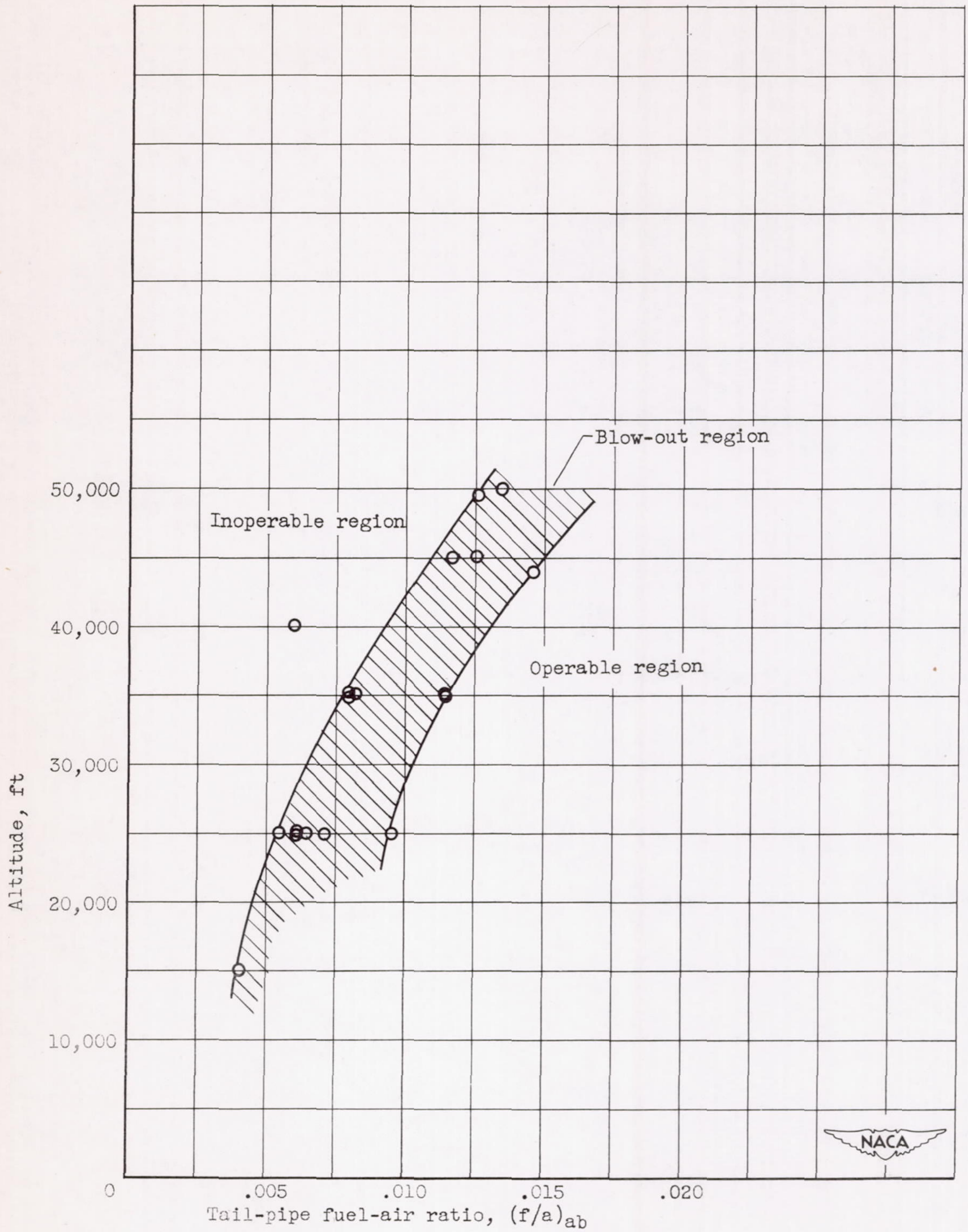


Figure 36. - Lean blow-out limits obtained with several configurations using MIL-F-5624 (AN-F-58) fuel at burner-inlet temperatures from 1250° to 1300° F. Flight Mach number, 0.19.