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RESEARCH MEMORANDUM

ALTITUDE OPERATIONAL CHARACTERISTICS OF PROTOTYPE J40-WE-8 TURBOJET ENGINE

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RESEARCH MEMORANDUM

ALTITUDE OPERATIONAL CHARACTERISTICS OF PROTOTYPE J40-WE-8 TURBOJET ENGINE

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2781

SUMMARY

The altitude operational characteristics of the prototype J40-WE-8 turbojet engine were determined in the NACA Lewis altitude wind tunnel. The operational characteristics that were obtained include (1) compressor surge characteristics, (2) acceleration and deceleration rates, (3) steady-state windmilling-engine characteristics, (4) altitude ignition characteristics, and (5) the effect of two grades of fuel on steady-state engine performance at one altitude and one flight Mach number.

The compressor surge line, when presented as a function of compressor pressure ratio and corrected engine speed, was not affected by changes in flight condition and was independent of engine-inlet installation and of the manner in which surge was approached, rapidly or slowly. Also, there was no effect of altitude or engine-inlet installation on the compressor surge recovery line when presented as a function of compressor pressure ratio and corrected engine speed.

The time required to accelerate from an initial engine speed to rated engine speed was found to increase as altitude was increased or as flight Mach number decreased. The time required to decelerate to a given engine speed from rated engine speed increased as altitude increased.

There was no effect of altitude on corrected steady-state windmilling-engine drag, speed, or air flow at a given flight Mach number.

An ignition procedure in which the fuel flow was maintained constant during the ignition period was found to give altitude ignition characteristics superior to those for an ignition procedure in which the fuel flow was varied in an approximate sinusoidal manner. Reduction in ignition fuel temperature from 50° to -28° F drastically reduced the range for which consistent ignition was obtained. The value of ignition fuel flow for which consistent ignition was obtained was found to be lower for the more volatile fuel (MIL-F-5624A grade JP-3) than for the less volatile fuel (MIL-F-5624A grade JP-4).

Similar effects on steady-state engine performance characteristics were obtained with the two grades of fuel at an altitude of 45,000 feet and flight Mach number of 0.20.

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INTRODUCTION

The altitude performance characteristics of the prototype J40-WE-8 turbojet engine were determined in the NACA Lewis altitude wind tunnel. The steady-state engine performance characteristics are presented in references 1 to 3. The altitude operational characteristics of the engine are presented herein. 2781

The operational characteristics that were obtained include (1) compressor surge characteristics, (2) acceleration and deceleration rates, (3) steady-state windmilling-engine characteristics, (4) altitude ignition characteristics, and (5) the effect of two grades of fuel on steady-state engine performance.

With two different engine-inlet installations and over a range of flight conditions, the compressor surge characteristics obtained during rapid changes in fuel flow in this investigation are compared with the surge limits obtained when surge is slowly approached by varying turbine-nozzle-diaphragm area (ref. 3). The effect of flight condition on acceleration and deceleration rates with the standard control operative is presented. Corrected steady-state windmilling-engine drag, speed, and air flow with variation of flight Mach number for a range of altitudes from 15,000 to 45,000 feet are given. The altitude ignition characteristics determined include the effects of (1) ignition fuel flow, (2) two ignition procedures, (3) fuel temperature, and (4) two grades of fuel on ignition characteristics. The results of a brief study of the effect of the two different grades of fuel (MIL-F-5624A grades JP-3 and JP-4) on steady-state engine performance at an altitude of 45,000 feet and a flight Mach number of 0.20 are also given.

APPARATUS

Engine

The prototype J40-WE-8 turbojet engine, with and without an afterburner, was used for this investigation. A view of the engine with the afterburner installed in the altitude wind tunnel is presented in figure 1.

A manufacturer's rating for the prototype J40-WE-8 turbojet engine, either with or without an afterburner, is not available at the present time. However, the rating of the prototype J40-WE-8 engine without afterburner would be similar to the rating of the J40-WE-6 engine, which has a static sea-level thrust of 7500 pounds at an engine speed of 7260 rpm and a turbine-inlet temperature of 1425° F (which corresponds to a turbine-outlet temperature of approximately 1120° F). At this operating condition, the air flow was approximately 142 pounds per second, and the compressor pressure ratio was about 5.1.

The principal components of the prototype J40-WE-8 engine with and without an afterburner are: an 11-stage axial-flow compressor, a single-annular combustor, a 2-stage turbine, a diffuser, and a variable-area exhaust nozzle. The maximum exhaust-nozzle-outlet area was 614 square inches for the engine with the afterburner and 534 square inches for the engine without the afterburner.

Engine fuel is sprayed downstream into the combustor through 16 spring-loaded variable-area nozzles equally spaced circumferentially at the upstream end of the combustor. Equal proportions of fuel are metered through each fuel nozzle by the combined action of the engine fuel distributor, the equalizing valves, and the spring-loaded variable-area nozzles.

The engine ignition system is of the low-voltage high-energy surface-gap ignition type. Two surface-gap spark plugs are located at the forward end of the combustor liner at two radial positions, approximately at 4 and at 8 o'clock. The spark generated by the ignition system produces 0.4 joule during a time duration of 40 microseconds. The spark frequency was 13 sparks per second.

Engine Control Systems

Standard control. - The term standard control refers to the manufacturer's integrated electronic control. The standard control was designed to provide approximately linear variation of engine thrust with throttle position for steady-state engine operation. This variation was accomplished by scheduling engine speed and turbine-outlet temperature as a function of throttle position. The standard control regulates engine fuel flow and exhaust-nozzle area to give the scheduled engine speed and turbine-outlet temperature. For acceleration from low speeds, in which the throttle is advanced in a stepwise manner, the standard control drives the exhaust-nozzle area to the open position for the initial portion of the transient and increases the fuel flow rapidly until the value of steady-state rated turbine-outlet temperature is reached. In the final portion of the acceleration, the standard control so adjusts the fuel flow and the exhaust-nozzle area that the required engine speed and turbine-outlet temperature as determined by the final throttle position are attained. During the acceleration period, the turbine-outlet temperature is limited to the value of steady-state rated turbine-outlet temperature by the standard control which regulates the fuel flow.

When the throttle is retracted suddenly during deceleration runs in which large speed changes occur, the exhaust-nozzle area is decreased during the initial portion of the deceleration. As the final engine speed, determined by the throttle position, is approached, the exhaust-nozzle area is increased to give the desired engine operating conditions. In the event of failure of components of the standard control, fail-safe features have been provided, and it is possible to operate the engine on an emergency hydraulic control system. A complete description of the operation of a control which is practically the same as the standard control is given in reference 4.

Modified control. - In order to obtain step changes in fuel flow for acceleration runs, the standard control was modified and is designated herein as the modified control. For engine operation with the modified control, the exhaust nozzle remained in the open position during the entire transient period; different size step changes in fuel flow were possible; and the fuel flow during the transient period was not affected by either turbine-outlet temperature or overspeed limits. As a safety feature, however, a switching arrangement was provided for suddenly reducing the fuel flow if dangerous engine operating conditions were encountered.

2781

INSTALLATION AND INSTRUMENTATION

Engine-inlet installation. - The engine was mounted on a wing section in the 20-foot-diameter test section of the altitude wind tunnel (fig. 1). Two different engine-inlet installations were used, ram pipe and bellmouth. A view of the ram-pipe installation is shown in figure 1 and a view of the bellmouth inlet, not installed on the engine, is presented in figure 2.

For the ram-pipe installation, dry refrigerated air was supplied to the engine through a duct (ram pipe) connected to the engine inlet. The air, at approximately sea-level pressure at the entrance of the make-up-air system, was throttled by manually controlled valves to a total pressure at the engine inlet corresponding to the desired flight condition; complete free-stream ram-pressure recovery was assumed. For the bellmouth installation, dry refrigerated air was supplied to the engine inlet from the tunnel air system. Tunnel fan speed, which determined the tunnel test-section air velocity, was varied to obtain the engine-inlet total pressure corresponding to the desired flight conditions; again, complete free-stream ram-pressure recovery was assumed. With the bellmouth installation, the range of flight Mach numbers possible was limited to low values; use of the ram-pipe installation, however, greatly increased the range of flight Mach numbers possible.

For transient engine operation with the ram-pipe installation, it was impossible to maintain a constant engine-inlet total pressure during the transient period. The variation in inlet pressure was due to the inherent slow-response time of the manually controlled valves in the ram-pipe system, which required continual adjustment during an engine transient. However, for the bellmouth installation, the engine-inlet total pressure remained constant for transient engine operation. For both engine-inlet installations, the static pressure in the tunnel test section remained approximately constant during an engine transient because of the large volume of the wind tunnel.

2781

Instrumentation. - Instrumentation for measuring steady-state values of pressures and temperatures was installed at various stations in the engine (fig. 3). A direct-inking electro-magnetic type recorder, with its associated amplifiers, was used for recording various engine parameters during transient engine operation (ref. 5).

The location and type of instrumentation used for the measurement of transient engine parameters is described in table I. Also included in this table is the type of steady-state instrumentation which was used for calibration of the transient instrumentation.

PROCEDURE

Acceleration data were obtained with the engine equipped with an afterburner. For engine operation with the standard control, acceleration was obtained by advancing the throttle in a stepwise manner, hereinafter described as a throttle burst, from several initial positions to that position at which rated engine conditions are scheduled. For engine operation with the modified control, acceleration was obtained by making several different size step-changes in fuel flow from the same initial engine speed. This procedure was repeated for different initial engine speeds.

The following table lists the flight conditions at which the engine acceleration data were obtained:

Engine control	Altitude, ft	Flight Mach number	Engine-inlet air temperature, °F
Ram-pipe installation			
Standard	10,000	0.20	30
	35,000	.96	18
Modified	10,000	.20	30
	35,000	.20	0
	35,000	.96	5
Bellmouth installation			
Standard	10,000	0.20	32
	35,000	.20	0
	45,000	.20	0
Modified	10,000	.20	32
	35,000	.20	-15

In a number of the acceleration runs listed in the preceding table, compressor surge was encountered. Therefore, it was possible to evaluate the effect of engine-inlet installation and flight condition on compressor surge characteristics.

In addition to the previously mentioned transient data, deceleration data were obtained at altitudes of 10,000, 35,000, and 45,000 feet at a flight Mach number of 0.20. The engine-inlet air temperature was 32° F at an altitude of 10,000 feet and 0° F at the other altitudes. The engine 2781
equipped with afterburner, bellmouth, and standard control was used for this phase of the investigation. For the deceleration runs, the throttle was retracted in a stepwise manner, which is designated herein as a throttle chop.

With the ram-pipe installation and without an afterburner, the steady-state windmilling-engine data were obtained at altitudes from 15,000 to 45,000 feet and over a range of flight Mach numbers. The engine-inlet air temperature was approximately 0° F.

Altitude ignition characteristics were determined with the ram-pipe installation and with an afterburner. Two methods of ignition, designated as ignition procedures A and B, were used. In both procedures the spark plugs were energized before the fuel was admitted to the combustor. For procedure A, the fuel flow was maintained at a constant value during the ignition period. This procedure was repeated at a number of constant fuel-flow values at a given windmilling-engine speed. For procedure B, the fuel flow was manually varied in an approximate sinusoidal manner from zero to the maximum possible at a particular engine speed. An ignition attempt was considered successful if ignition of the combustible mixture in the engine occurred within approximately 30 seconds after the fuel was admitted to the combustor. After a successful ignition, no attempt was made to accelerate the engine to determine whether an engine-speed dead-band region exists, because the varying engine-inlet total pressure encountered during accelerations from low engine speeds made it difficult to obtain data showing consistent trends.

With MIL-F-5624A grade JP-4 fuel at a temperature of 50° F, the effects of ignition fuel flow and ignition procedure on altitude ignition limits were determined at altitudes up to 50,000 feet over a range of windmilling-engine speeds. In addition, with ignition procedure A, ignition fuel-flow limits were obtained at an altitude of 45,000 feet and over a range of windmilling-engine speeds for MIL-F-5624A grades JP-3 and JP-4 fuels at temperatures of 50° and -28° F, respectively. Fuel temperature was measured upstream of the fuel distributor. For all the starting data, the engine-inlet air temperature was approximately 0° F.

2781

The effect of two grades of fuel, MIL-F-5624A JP-3 and JP-4, on steady-state engine performance was determined at an altitude of 45,000 feet, a flight Mach number of 0.20, and an engine-inlet air temperature of 0° F. Performance data were obtained at rated engine speed, 7260 rpm, for several nozzle areas and a constant exhaust-nozzle area of 614 square inches (full open) over a range of engine speeds. The fuel temperature for both grades of fuel was about 130° F as measured upstream of the fuel distributor.

All symbols used in this report and the methods of calculation are presented in appendixes A and B, respectively.

RESULTS AND DISCUSSION

Variation of Engine Parameters During Acceleration

Oscillograph traces showing typical variations of several different engine parameters with time during an acceleration are presented in figure 4. The engine parameters increase in magnitude in the upward direction on the oscillograph traces. With the bellmouth installation, for a throttle-burst run during which compressor surge was not encountered (fig. 4(a)), the engine-inlet total pressure remained constant, and the other engine parameters increased during the acceleration period. Data obtained in a throttle-burst run during which surge was encountered are presented in figure 4(b) for the bellmouth installation. The sudden reduction in compressor-outlet total pressure and the increase in turbine-outlet temperature indicate the point at which compressor surge is initiated. The period of engine operation with compressor surge is shown by the fluctuations of the pressure and temperature traces.

The variation in engine parameters during an acceleration with ram-pipe installation was similar to that obtained with the bellmouth installation except for the engine-inlet total pressure With the ram-pipe installation, the engine-inlet total pressure varied slightly during the acceleration if surge was not encountered (fig. 4(c)) and appreciably during an acceleration in which surge was encountered (fig. 4(d)).

Compressor Surge Characteristics

Compressor surge limits. - Compressor surge may be encountered in either steady-state or transient engine operation. In either case, air-flow instability or air-flow break-down occur within the compressor as a result of blade stall caused by operation with adverse engine-inlet air-flow conditions or at excessive pressure ratios. For instance, for transient engine operation at a given engine-inlet total pressure and initial engine speed, if the fuel flow is suddenly increased, the turbine-inlet temperature (as well as the turbine-outlet temperature) rises suddenly. This rise in turbine-inlet temperature is reflected at the compressor outlet as a sudden rise in total pressure. If this pressure exceeds a certain value, dependent upon the compressor characteristics, the compressor will surge. Oscillograph traces which show compressor surge encountered in this manner are presented in figures 4(b) and 4(d). When the compressor begins to surge, the air flow and the compressor-outlet pressure are suddenly reduced and begin to fluctuate. If the fuel flow is not reduced, the turbine-outlet temperature increases (because of decreased air flow and compressor efficiency) and also begins to fluctuate (figs. 4(b) and 4(d)). 2781

The compressor-surge data points presented in figure 5 were obtained from acceleration runs with both standard and modified engine controls and with the two different engine-inlet installations. The compressor pressure ratio at which surge occurred is presented in figure 5 as a function of corrected engine speed rather than corrected air flow, because air-flow measurements were not obtained during transients; also shown in this figure is the steady-state compressor surge line, which was obtained from reference 3. This steady-state compressor surge line was obtained with the use of a variable-area turbine-nozzle diaphragm which permitted the determination of the compressor surge line without the penalty of exceeding the rated turbine-inlet temperature. At a given engine speed, the turbine-nozzle-diaphragm area was slowly reduced until compressor surge occurred. The steady-state compressor surge line was determined from data obtained at an altitude of 45,000 feet and flight Mach number of 0.21, and at an altitude of 30,000 feet and flight Mach number of 0.64.

Although the surge data, obtained under transient conditions, were determined over only a limited range of corrected engine speeds, a good correlation between the surge data and the steady-state surge line is indicated in figure 5. Thus, the compressor surge line for this engine is not affected by changes in flight condition and is independent of engine-inlet installation and of the manner in which surge is approached (rapidly or slowly).

Compressor recovery limits. - Data for the operating conditions at which the compressor recovered from surge were obtained from those acceleration runs in which compressor surge was encountered. It was possible to recover the compressor from surge either by reducing the fuel flow or by allowing the engine speed to increase. In either case, the point of compressor recovery was evidenced by the sudden increase in compressor-outlet total pressure and by the elimination of engine pressure and temperature fluctuations (figs. 4(b) and 4(d)). These compressor recovery data points, as presented in figure 6, can be represented by a single line indicating that there is no effect of altitude

2781

or type of engine-inlet installation on the recovery line. As the surge line was not affected either by altitude or by flight Mach number (fig. 5) and there was no effect of altitude on the recovery line (fig. 6), it is reasonable to assume that there is also no effect of flight Mach number on the recovery line.

Relation Between Compressor Surge Characteristics, and Transient and Steady-State Engine Operating Lines

A steady-state operating line, compressor surge characteristics, and typical variations of compressor pressure ratio with corrected engine speed for two throttle-burst runs are shown in figure 7. This figure presents the relation existing among the various engine operating lines. Because the recovery line is below the steady-state operating line, it is evident that the compressor after entering surge can recovery only if the compressor pressure ratio is reduced to less than its steady-state operating value at a particular engine speed. This reduction requires either a reduction of fuel flow or an increase in engine speed (during the surge period) to a speed at which the transient surge compressor pressure ratio equals the recovery compressor pressure ratio. An example of the latter method of recovery is shown in figure 7 by the path of a typical throttle-burst run during which surge was encountered. In this run, the fuel flow remained approximately constant during the surge period, and the exhaust nozzle remained in the open position until rated engine speed, 7260 rpm, was approached. Following the sudden reduction in compressor pressure ratio when surge occurred, both engine speed and compressor pressure ratio increased during the surge period. Even though the engine speed increases during surge, the engine acceleration rates were substantially lower for the acceleration runs in which surge was encountered than for the runs in which surge was not encountered. During the entire surge period, the compressor pressure ratio was at a value lower than its steady-state value at the same corrected engine speed. The increase in compressor pressure ratio with increasing corrected engine speed during the surge period was slight until the compressor recovered, at which point the compressor pressure ratio increased abruptly to a value dependent upon fuel flow and engine speed. Also shown in figure 7 is the path of a typical throttle-burst run in which no surge was encountered.

Since the distance between the steady-state operating line and the surge line is indicative of the excess power available for acceleration, it is evident that, for the lower values of corrected engine speed, the available power for acceleration is limited. Because of the steady-state performance characteristics of the engine for a constant exhaust-nozzle area, the steady-state compressor pressure ratio increases as altitude is increased at a given corrected engine speed. Therefore, since the surge line is not affected by changes in altitude, the available power for acceleration at a given corrected engine speed tends to decrease with increasing altitude. Inasmuch as similar variations in steady-state engine performance occur for a decrease in flight Mach number as for an increase in altitude, the available power for acceleration at a given corrected engine speed and exhaust-nozzle area would also tend to decrease as flight Mach number is decreased.

The relation between steady-state corrected fuel flow for a fixed 2781 exhaust-nozzle area and the steady-state corrected fuel flow at surge for different flight conditions is shown in figure 8. The corrected fuel flow required for steady-state operation at rated engine conditions is also shown. The data for the steady-state surge lines were obtained by extrapolation of steady-state engine performance data (refs. 1 and 3) to the compressor surge line. The steady-state surge lines as well as the steady-state operating lines and the data for rated engine conditions reflect the trend of varying component efficiencies that result from engine operation at the different flight conditions.

Unpublished NACA data indicate that, for acceleration runs in which the fuel flow is rapidly increased, the combustion efficiency during the initial transient period is lower than at steady-state conditions. Because of the lower combustion efficiency initially encountered during a transient, it may be possible to obtain a higher corrected fuel flow at a given corrected engine speed without encountering surge during transient than during steady-state operation. The steady-state surge lines presented in figure 8 would then give a conservative picture of limiting corrected fuel flow for avoiding surge if applied to transient operation. Figure 8 shows that at an altitude of 15,000 feet and a flight Mach number of 0.62, a fuel-flow step large enough to result in rated conditions without encountering surge cannot be introduced at an initial corrected engine speed below approximately 6900 rpm. At this flight condition, an initial corrected engine speed of 5400 rpm, and a corrected fuel flow of 1900 pounds per hour, the largest step which can be made without encountering surge is to a fuel flow value of 3700 pounds per hour, which is about one-fourth that required to reach rated conditions for which the fuel flow is 8450 pounds per hour.

Acceleration of Engine with Standard Control

Engine acceleration runs obtained by throttle burst are presented in figures 9 to 11. For this portion of the investigation several adjustments were made to the standard control to eliminate a temporary malfunctioning, as a result, the control schedule used was different from the original schedule as specified by the manufacturer. For some of the acceleration runs in which compressor surge was encountered, it may have been possible to avoid surge if the correct schedule had been used. However, for most of the acceleration characteristics presented herein, the adverse effect of surge on acceleration times was considered minor. The overshoot and the oscillation of engine speed at rated engine conditions (figs. 9 to 11) are due to the temporary malfunctioning of the standard control.

2781

The rate of engine acceleration is dependent upon: (1) the inertia and the mechanical friction of the rotating parts, (2) the ram energy of the engine-inlet air, (3) the internal aerodynamic friction of the engine, and (4) the excess power available from the turbine. As either altitude is increased or flight Mach number is decreased, the ram energy of the air, the internal aerodynamic friction of the engine, and the excess energy developed by the turbine decrease, but the inertia and the mechanical friction of the rotating parts of the engine remain constant. In addition, at any flight condition, the maximum power available for acceleration is limited by the compressor surge characteristics. As mentioned previously, the power available (in terms of increased compressor pressure ratio) at any given corrected engine speed for acceleration without surge decreases with increasing altitude or decreasing flight Mach number. Therefore, as shown in figures 9 and 10, the time required to accelerate from any constant initial engine speed to rated engine speed was increased either as altitude increased (fig. 9) or as flight Mach number decreased (fig. 10). For example, the time required to accelerate from 5800 rpm to rated engine speed, 7260 rpm, increased from approximately 3 seconds at an altitude of 10,000 feet to about 10 seconds at an altitude of 45,000 feet (fig. 9(b)).

Because the time interval during which surge was encountered was small, it was assumed that the effect of compressor surge on acceleration time shown in figures 9 and 10 is small. However, to illustrate the adverse effect that surge may have on acceleration time, acceleration runs obtained for two throttle-burst runs at an altitude of 35,000 feet and flight Mach number of 0.20 are presented in figure 11. The acceleration time required for an engine speed change from 4940 rpm to rated engine speed was 9 seconds for run 1. For run 2, the time required to accelerate from an engine speed of 4340 rpm to rated engine speed was about 29 seconds. This large increase in acceleration time for run 2 as compared with run 1 was primarily due to the longer period of compressor surge encountered.

Where throttle-burst runs were obtained with both ram-pipe and bellmouth installations at the same flight conditions, comparison of acceleration rates indicates no difference with either engine-inlet installation provided that surge was not encountered during the acceleration. However, when surge was encountered during the acceleration, the acceleration rates were slightly higher for the ram-pipe installation than for the bellmouth installation because of the higher engine-inlet total pressure occurring during surge with the ram-pipe installation (fig. 4(d)).

Deceleration of Engine with Standard Control

The effect of altitude on the variation of engine speed with time for throttle chop is presented in figure 12. When the engine speed was reduced from rated speed to either idle speed or to a given final speed by throttle chop, the deceleration rates decreased as altitude was increased. This decrease in deceleration rates with increasing altitude can be attributed to the previously discussed factors which affected acceleration rates. The time required for the engine to decelerate from rated engine speed, 7260 rpm, to about 6000 rpm increased from approximately 2 seconds at an altitude of 10,000 feet to 6 seconds at an altitude of 45,000 feet at a flight Mach number of 0.20 (fig. 12(b)). 2781

Steady-State Windmilling-Engine Characteristics

Data for the corrected steady-state windmilling-engine drag, speed, and air flow are presented in figures 13 to 15, respectively, as functions of flight Mach number for a range of altitudes from 15,000 to 45,000 feet. Changes in altitude did not affect the corrected steady-state windmilling-engine parameters at a given flight Mach number. As flight Mach number was increased from 0.20 to 0.90, corrected steady-state windmilling-engine drag varied from 30 pounds to 1070 pounds. The corrected steady-state windmilling-engine drag of 1070 pounds corresponds to about 14 percent of the rated sea-level static engine thrust. For the same range of flight Mach numbers, the corrected steady-state windmilling-engine speed increased from 680 to 3180 rpm; while the corrected steady-state windmilling-engine air flow increased from 12.5 to 49.0 pounds per second.

With the afterburner-equipped engine, which had a maximum exhaust-nozzle area of 614 square inches, some steady-state windmilling-engine data were obtained but these data are not presented herein. However, a comparison of the data, showing the variation of corrected steady-state windmilling-engine speed with flight Mach number for a range of altitudes, indicated no difference between the two exhaust-nozzle areas, 534 and 614 square inches.

Altitude Ignition Characteristics

Many factors influence the altitude ignition characteristics of a turbojet engine. Some of the major factors are, combustor configurations, fuel nozzles, ignition systems, fuel properties, and combustor-inlet air-flow conditions. The combustor configuration, the fuel nozzles, and the ignition system used in this portion of the investigation were supplied by the manufacturer and were not changed during the ignition program. Ignition data presented herein show the effects of fuel flow at different altitudes, ignition procedure, fuel temperature, and two grades of fuel on ignition characteristics.

Most of the ignition data were obtained at altitudes of 40,000, 45,000, and 50,000 feet; therefore, the combustor-inlet air-flow conditions at these altitudes are presented in figure 16 for a range of steady-state windmilling-engine speeds. The combustor velocity, determined from engine air flow, combustor-inlet total pressure, and temperature, was based on the maximum cross-sectional area of the combustor, 6.40 square feet. (See appendix B.) As the engine-inlet air temperature was constant at 0° F, combustor velocity and combustor-inlet total temperature were not affected at a given windmilling-engine speed by increasing the altitude from 40,000 to 50,000 feet (fig. 16).

Effect of fuel flow. - The altitude ignition data presented in figure 17 were obtained with ignition procedure A (where the fuel flow was maintained at a constant value during the ignition period) for MIL-F-5624A grade JP-4 fuel and for three altitudes over a range of windmilling-engine speeds. (An alternate abscissa scale of flight Mach number, which was shown to be directly related to windmilling-engine speed, is presented for convenience in figure 17 and in all subsequent figures relating to ignition characteristics.) The maximum ignition fuel flow was limited by the engine fuel system; the fuel-flow limit increased as windmilling-engine speed increased at a given altitude or as altitude decreased at a given windmilling-engine speed. The fuel-flow limits encountered are designated in figure 17 and in subsequent figures by dashed lines. The ignition data shown in figure 17 define regions in which consistent ignition occurred and regions in which ignition is doubtful. The upper limit of the region of consistent ignition was not obtained because of the engine fuel-flow limit.

At a given altitude as the windmilling-engine speed is increased, the minimum fuel flow required for consistent ignition increased, because of the increase in windmilling-engine air flow. If the windmilling-engine speed is increased above a certain value at a given altitude, ignition is uncertain. This trend may be attributed to an increase in combustor velocity resulting from an increase in windmilling-engine speed, for the other factors influencing ignition, such as combustor-inlet total pressure and temperature (fig. 16) and fuel-flow spray characteristics, vary in a favorable direction.

The ignition data indicate that, at a constant windmilling-engine speed, as altitude is increased from 40,000 to 50,000 feet, the minimum fuel flow necessary for consistent ignition increased. For example, at a constant windmilling-engine speed of 1700 rpm, as altitude is increased from 40,000 to 50,000 feet, the minimum fuel flow required for consistent ignition increased from 640 to 770 pounds per hour (figs. 17(a) and 17(c)). The maximum windmilling-engine speed at which consistent ignition was obtained was reduced from 2500 to 1750 rpm as altitude was increased from 40,000 to 50,000 feet. These trends may be accounted for by the decrease in combustor-inlet total pressure. At a constant windmilling-engine speed as altitude is increased, the combustor-inlet total temperature and combustor velocity and the fuel-flow spray characteristics do not change, but the combustor-inlet total pressure is reduced as shown in figure 16.

Ignition procedure. - The altitude ignition data obtained with ignition procedure B for MIL-F-5624A grade JP-4 fuel are presented in figure 18. For altitudes from 40,000 to 50,000 feet, the maximum windmilling-engine speeds at which consistent ignition occurred for ignition procedure A (fig. 17) are also presented for comparison in figure 18. 2781 For ignition procedure B, the ignition fuel flow was varied in an approximate sinusoidal manner from zero to the maximum flow possible at a particular windmilling-engine speed. Again, the maximum fuel flow was limited by the engine fuel system. For ignition procedure B, the maximum windmilling-engine speed at which consistent ignition occurred progressively decreased from 3300 to 1800 rpm as altitude was increased from 30,000 to 48,000 feet; at an altitude of 48,000 feet, it decreased abruptly from 1800 to 720 rpm as altitude was increased to 50,000 feet.

Ignition procedure A was superior to ignition procedure B at all altitudes at which a comparison was possible (fig. 18). For example, at an altitude of 50,000 feet, the maximum consistent-ignition windmilling-engine speed was 1750 rpm for ignition procedure A and it decreased to 720 rpm for ignition procedure B.

Fuel temperature. - Ignition data obtained at an altitude of 45,000 feet over a range of windmilling-engine speeds with ignition procedure A with MIL-F-5624A grade JP-4 fuel at a temperature of -28° F are presented in figure 19. A comparison of the ignition limits obtained with this fuel at temperatures of 50° F (fig. 17(b)) and -28° F (fig. 19) is shown in figure 20. At a fuel temperature of 50° F, consistent ignition was obtained at a maximum windmilling-engine speed of 2500 rpm and an ignition fuel flow of 1530 pounds per hour. However, when the fuel temperature was decreased to -28° F, the maximum windmilling-engine speed at which consistent ignition was obtained occurred at a speed of 1300 rpm and an ignition fuel flow of 700 pounds per hour. This trend of decreasing ignition limits with decreasing fuel temperature is probably the result of a lower rate of fuel evaporation at the lower fuel temperature. The large reduction in the engine fuel-flow limit for the fuel at a temperature of -28° F as compared to that at a fuel temperature of 50° F was a result of a change in engine throttle-linkage system.

Different fuels. - The ignition data shown in figure 21 were obtained at an altitude of 45,000 feet and over a range of windmilling-engine speeds with MIL-F-5624A grade JP-3 fuel at a temperature of 50° F. As shown in table II, the major difference between MIL-F-5624A grades JP-3 and JP-4 fuels was the Reid vapor pressure or volatility. The Reid vapor pressure was 5.4 pounds per square inch for MIL-F-5624A grade JP-3 fuel and 2.8 pounds per square inch for MIL-F-5624A grade JP-4 fuel. The ignition limits obtained at an altitude of 45,000 feet with MIL-F-5624A grades JP-3 (fig. 21) and JP-4 (fig. 17(b)) are compared in figure 22 to illustrate the effect of volatility on ignition limits. An increase in volatility would increase the rate of fuel evaporation and hence should improve the ignition characteristics. As shown in figure 22, at a constant windmilling-engine speed, consistent ignition was obtained at a lower ignition fuel flow for MIL-F-5624A grade JP-3 fuel than for MIL-F-5624A grade JP-4 fuel; this indicates that the less volatile fuel (MIL-F-5624A grade JP-4) was more difficult to ignite. For instance, at a windmilling-engine speed of 1800 rpm, the ignition fuel flow was 640 pounds per hour for MIL-F-5624A grade JP-3 fuel and 860 pounds per hour for MIL-F-5624A grade JP-4 fuel. However, as the windmilling-engine speed was increased, the beneficial effect of the more volatile fuel on ignition characteristics was decreased with the result that about the same maximum consistent-ignition windmilling-engine speeds were obtained for both grades of fuel.

Engine Performance Evaluation with Two Different Fuels

The effect of two grades of fuel, MIL-F-5624A grades JP-3 and JP-4, on steady-state engine performance was determined at an altitude of 45,000 feet, flight Mach number of 0.20, and engine-inlet air temperature of 0° F. Engine-performance data obtained with these two grades of fuel are presented in figures 23 to 26 in which turbine-outlet temperature distribution, net thrust, combustion efficiency, and specific fuel consumption are shown for a range of engine operating conditions. As shown in figures 23 to 26, similar effects on engine performance were obtained with the two grades of fuel for the flight condition and range of engine operating conditions investigated.

SUMMARY OF RESULTS

The following results were obtained in an altitude operational investigation of the prototype J40-WE-8 turbojet engine in the NACA Lewis altitude wind tunnel:

1. The compressor surge line, when presented as a function of compressor pressure ratio and corrected engine speed, was not affected by changes in flight condition and was independent of engine-inlet installation and of the manner in which surge was approached, rapidly or slowly.

2. There was no effect of either altitude or engine-inlet installation on compressor surge recovery when presented as a function of compressor pressure ratio and corrected engine speed. At any given corrected engine speed, the recovery compressor pressure ratio was lower than the steady-state compressor pressure ratio.

3. At a flight Mach number of 0.20, the time required to accelerate from 5800 rpm to rated engine speed, 7260 rpm, increased from about 3 seconds at an altitude of 10,000 feet to about 10 seconds at an altitude of 45,000 feet. For the same flight Mach number, the engine decelerated from rated engine speed, 7260 rpm, to about 6000 rpm in approximately 2 seconds at an altitude of 10,000 feet, and 6 seconds at an altitude of 45,000 feet. 2781

4. There was no effect of altitude on corrected steady-state windmilling-engine drag, speed, or air flow for a given flight Mach number. At a flight Mach number of 0.90, the corrected steady-state windmilling-engine drag, speed, and air flow were 1070 pounds, 3180 rpm, and 49.0 pounds per second, respectively.

5. An ignition procedure (procedure A) in which the fuel flow was maintained constant during the ignition period, was found to have superior altitude ignition characteristics to another ignition procedure (procedure B) in which the fuel flow was varied in an approximately sinusoidal manner. At an altitude of 50,000 feet, the maximum windmilling-engine speed at which consistent ignition was obtained decreased from 1750 rpm for ignition procedure A to 720 rpm for ignition procedure B.

6. At an altitude of 45,000 feet and using ignition procedure A, the maximum windmilling-engine speed, at which consistent ignition was obtained, decreased from 2500 to 1300 rpm as the fuel (MIL-F-5624A grade JP-4) temperature was decreased from 50° to -28° F. The ignition fuel flows at these engine speeds were 1530 and 700 pounds per hour, respectively.

7. At an altitude of 45,000 feet, using ignition procedure A, and at a given windmilling-engine speed, the fuel flow, at which consistent ignition was obtained, was lower for the higher volatile fuel (MIL-F-5624A grade JP-3) than for the lower volatile fuel (MIL-F-5624A grade JP-4).

8. Similar effects on steady-stage engine performance characteristics were obtained with the two grades of fuel (MIL-F-5624A grades JP-3 and JP-4) at an altitude of 45,000 feet and flight Mach number of 0.20.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 4, 1952

APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	cross-sectional area, sq ft
B	thrust scale reading, lb
c_p	specific heat at constant pressure, Btu/(lb)($^{\circ}$R)
c_v	specific heat at constant volume, Btu/(lb)($^{\circ}$R)
D	external drag of engine installation, lb
D_r	exhaust-nozzle tail-rake drag, lb
D_w	windmilling-engine drag, lb
F	thrust, lb
g	acceleration due to gravity, 32.2 ft/sec^2
H	enthalpy, Btu/lb
K	coefficient
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft abs.
p	static pressure, lb/sq ft abs.
R	gas constant, 53.4 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec
W_a	air flow, lb/sec

W_f fuel flow, lb/hr

W_g gas flow, lb/sec

γ ratio of specific heats, c_p/c_v

δ pressure correction factor, P/2116 (total pressure divided by NACA standard sea-level pressure)

η efficiency

θ temperature correction factor, $\gamma T/(1.4)(519)$ (product of γ and total temperature divided by product of γ at standard sea-level temperature and standard NACA sea-level temperature)

Subscripts:

0 free-stream condition

1 cowl-inlet

2 engine-inlet

4 compressor-outlet, combustor-inlet

6 turbine-outlet

9 exhaust-nozzle outlet

a air

b combustor

f fuel

j jet

n net

r rake

s scale

x instrumentation station upstream from station 1

2781

APPENDIX B

METHODS OF CALCULATION

Air flow. - Air flow was calculated at station 1 (fig. 3) by use of the following equation:

$$W_{a,1} = p_1 A_1 \sqrt{\frac{2\gamma_1 g}{(\gamma_1 - 1)\, Rt_1}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]}$$

Gas flow downstream of the combustor is

$$W_g = W_{a,1} + \frac{W_f}{3600}$$

Combustor velocity. - Before calculating the combustor velocity, based on a combustor maximum cross-sectional area of 6.40 square feet, the combustor Mach number must be determined by use of the following equation:

$$\frac{M_b}{\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)^{\frac{\gamma_4 + 1}{2(\gamma_4 - 1)}}} = \frac{W_{a,1}\sqrt{T_4}}{0.776\, A_b P_4 \sqrt{\gamma_4}}$$

then

$$V_b = M_b \sqrt{\gamma_4 g R t_4}$$

where

$$t_4 = \frac{T_4}{\left(1 + \frac{\gamma_4 - 1}{2} M_b^2\right)}$$

Combustion efficiency. - On the assumption that compressor and turbine work are equal, combustion efficiency is defined as the ratio of the actual enthalpy rise of the gas while passing through the engine to the theoretical increase in enthalpy that would result from complete combustion of the fuel:

$$\eta_b = \frac{\text{actual enthalpy rise of gas across engine}}{\text{heat input}}$$

$$= \frac{3600\left[W_{a,1}H_a\right]_{T_1}^{T_6} + \left[W_f H_f\right]_{T_f}^{T_6}}{18{,}700\ W_f}$$

2781

where 18,700 Btu per pound of fuel is the lower heating value of the fuel.

Net thrust. - The net thrust developed by the engine was determined from the expression

$$F_n = K\,F_{j,r} - \frac{W_{a,1}V_0}{g}$$

which presents the net thrust as the actual jet thrust less the inlet momentum for which,

$$V_0 = \sqrt{\frac{2\gamma_1}{\gamma_1 - 1}\ gRT_1\left[1 - \left(\frac{p_0}{P_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}}\right]}$$

The experimentally determined coefficient K is applied to the rake jet thrust to account for pressure losses in the exhaust nozzle.

The rake jet thrust which is the ideal thrust available is calculated from the conditions existing at the exhaust-nozzle outlet and is expressed by

$$F_{j,r} = \frac{W_g}{g}\sqrt{\frac{2\gamma_6}{\gamma_6 - 1}\ gRT_6\left[1 - \left(\frac{p_0}{P_q}\right)^{\frac{\gamma_6 - 1}{\gamma_6}}\right]}$$

For all engine operating conditions at an altitude of 45,000 feet and a flight Mach number of 0.20, subcritical nozzle pressure ratios occurred ($p_0/P_q > 0.528$).

Steady-state windmilling-engine drag. - The windmilling-engine drag was calculated from the following expression:

$$D_w = \frac{W_{a,1}V_0}{g} - F_{j,s}$$

The windmilling-engine scale jet thrust was determined from balance-scale measurements by the use of the following equation:

$$F_{j,s} = D + D_r + B + \frac{W_{a,1}V_x}{g} + A_x\ (p_x - p_0)$$

2781

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2. Sobolewski, Adam E., Miller, Robert R., and McAulay, John E.: Altitude Performance Investigation of Two Single-Annular Type Combustors and the Prototype J40-WE-8 Turbojet Engine Combustor With Various Combustor-Inlet Air Pressure Profiles. NACA RM E52J07. 2781

3. Conrad, E. William, Finger, Harold B., and Essig, Robert H.: Effect of Rotor- and Stator-Blade Modifications on Surge Performance of an 11-Stage Axial-Flow Compressor. II - Redesigned Compressor for XJ40-WE-6. NACA RM E52I10.

4. Vasu, George, and Hinde, William L.: Effect of Engine and Control Limits on Steady-State and Transient Performance of Turbojet Engine with Variable-Area Exhaust Nozzle. NACA RM E52E23, 1952.

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

TABLE I - INSTRUMENTATION

[redacted]

Measured quantity	Station	Steady-state instrumentation	Transient instrumentation	
			Sensor	Range over which frequency response curve is flat, cycles/sec
Engine-inlet pressure	2	Bourdon-type gage	Aneroid-type pressure sensor with strain gage element	0-10 at sea-level static pressure
Engine speed	-	Chronometric tachometer	Modified electronic tachometer	0-10
Compressor-outlet pressure	4	Bourdon-type gage	Aneroid-type pressure sensor with strain gage element	0-10 at sea-level static pressure
Fuel flow	-	Rotameter	Aneroid-type pressure sensor with strain gage element for measuring pressure drop across variable orifice in fuel line	----
Turbine-outlet temperature	6	Nine thermocouples in parallel connected to self-balancing potentiometer	Five unshielded, loop thermocouples connected in series	0-1 at sea-level static pressure
Exhaust-nozzle area	-	Microameter connected to exhaust-nozzle area potentiometer	Exhaust-nozzle area potentiometer	0-100

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TABLE II - FUEL ANALYSES

Fuel properties	MIL-F-5624A	
	Grade JP-3	Grade JP-4
A.S.T.M. distillation, °F		
Initial boiling point	117	139
Percent evaporated		
5	155	201
10	187	224
20	234	250
30	266	269
40	291	286
50	312	303
60	333	322
70	358	344
80	394	375
90	449	421
95	487	453
Final boiling point	523	486
Residue, percent	1.3	1.2
Loss, percent	1.3	0.3
Reid vapor pressure, lb/sq in.	5.4	2.8
Gravity		
°API	53.4	54.6
Specific	0.765	0.760
Hydrogen-carbon ratio	.170	.171
Heat of combustion, Btu/lb	18,700	18,730
Aniline point, °F	134.6	------

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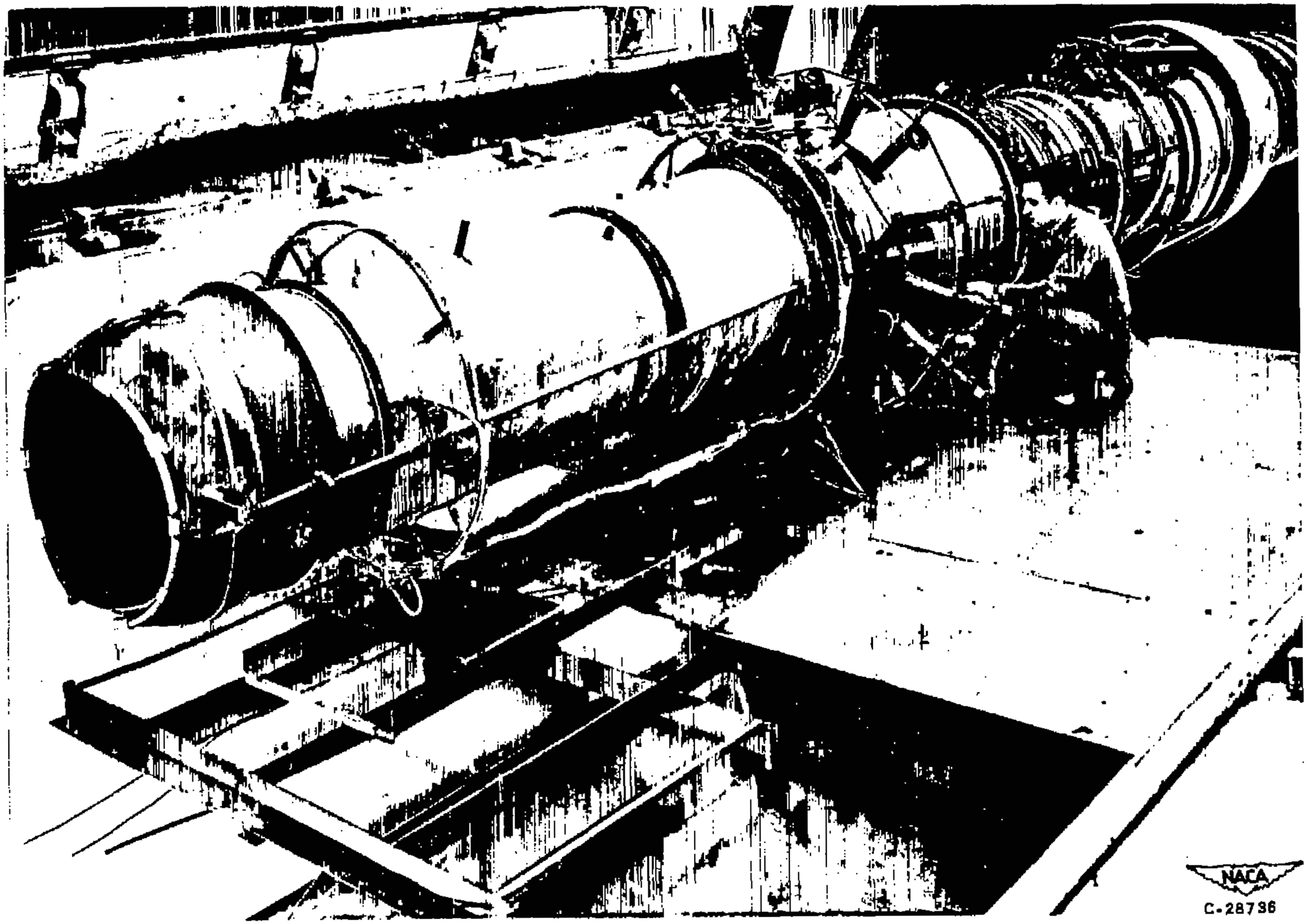

Figure 1. - Ram-pipe engine-inlet installation of engine with afterburner in altitude wind tunnel test section.

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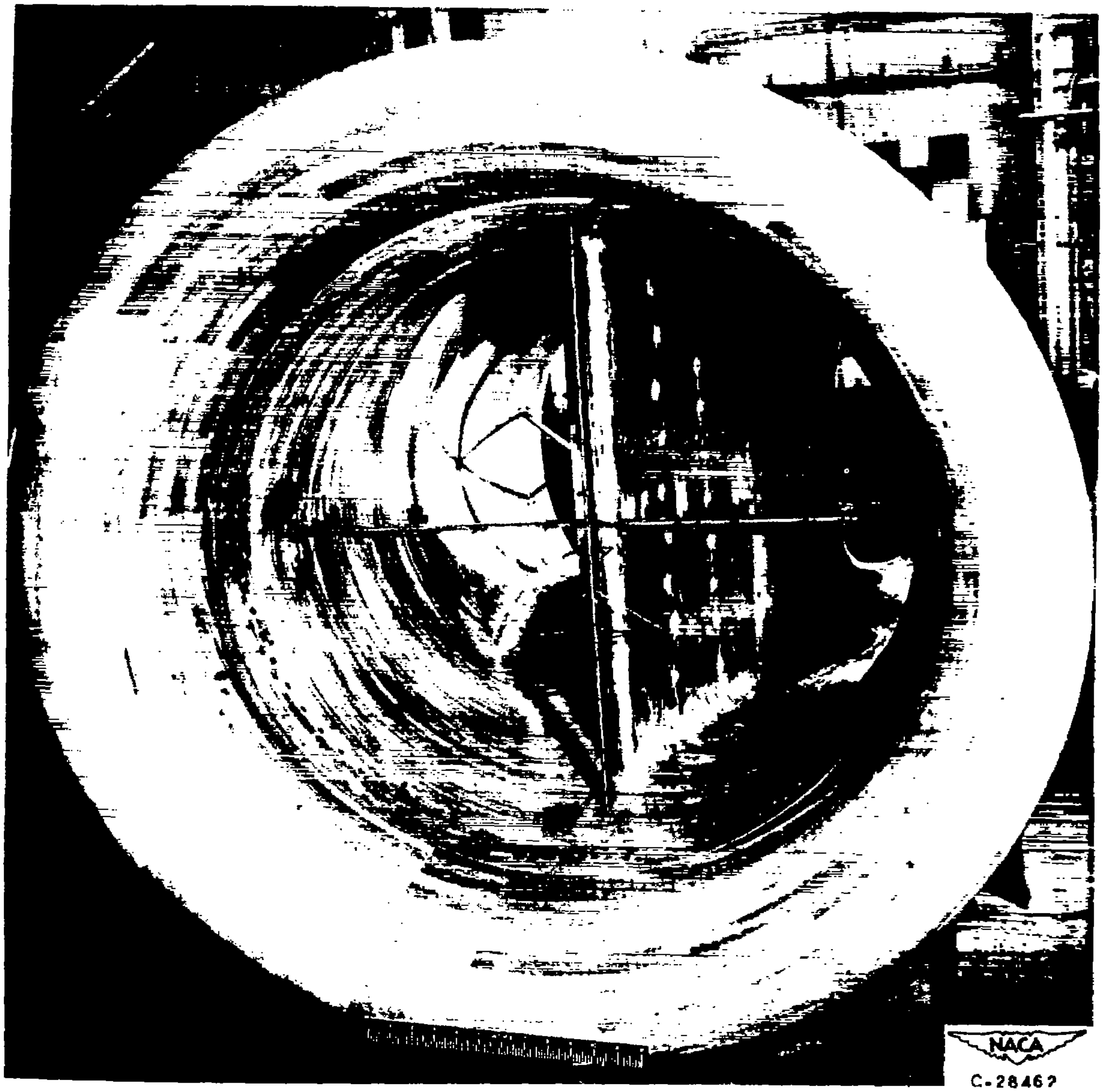
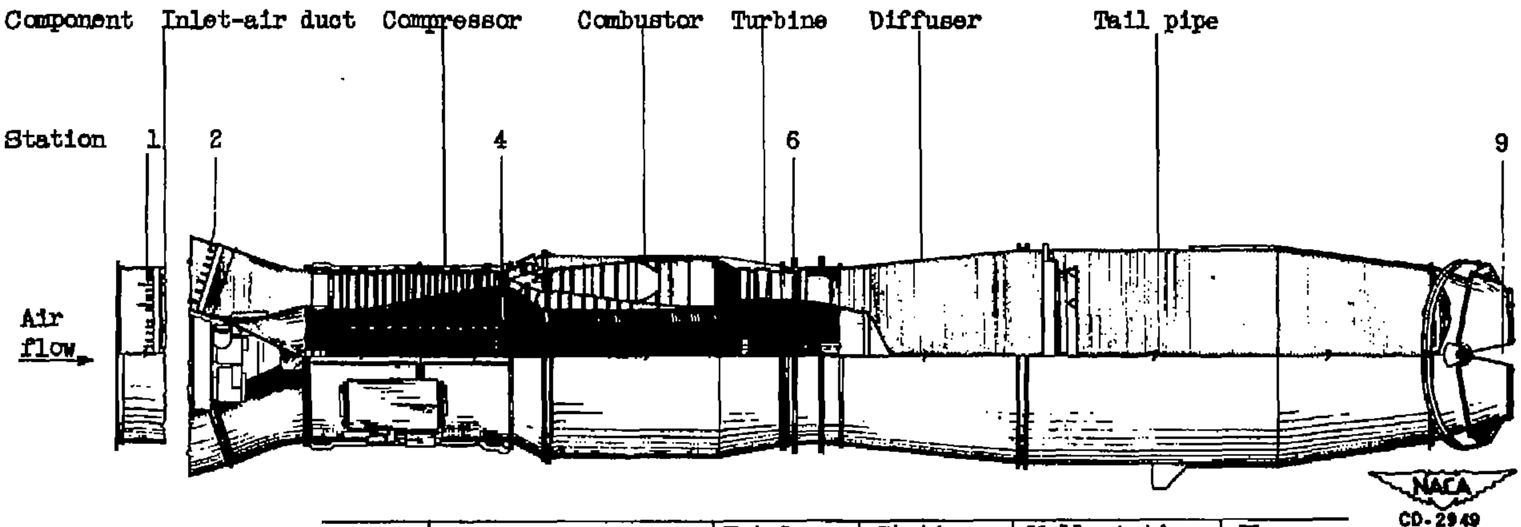


Figure 2. - Bellmouth engine inlet, viewed looking downstream.



Station	Location	Total-pressure tubes	Static-pressure tubes	Wall static-pressure orifices	Thermo-couples
1	Inlet-air duct	29	12	6	10
2	Engine inlet	18	0	4	0
4	Compressor outlet	18	0	3	6
6	Turbine outlet	20	0	8	24
9	Exhaust nozzle outlet	17	6	0	0

Figure 3. - Cross section of engine with afterburner showing stations at which steady-state instrumentation was installed.

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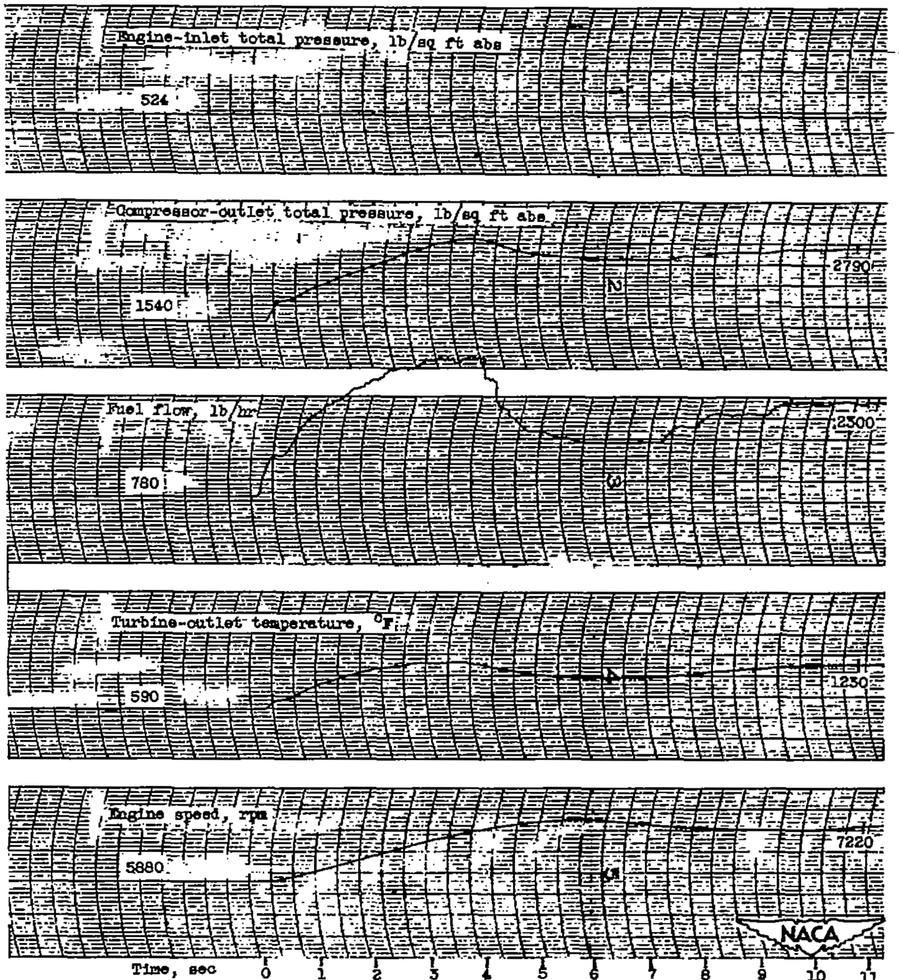

(a) Bellmouth engine-inlet installation; no surge encountered during throttle burst to rated engine conditions.

Figure 4. - Oscillograph traces showing variations of different engine parameters during acceleration. Altitude, 35,000 feet; flight Mach number, 0.20.

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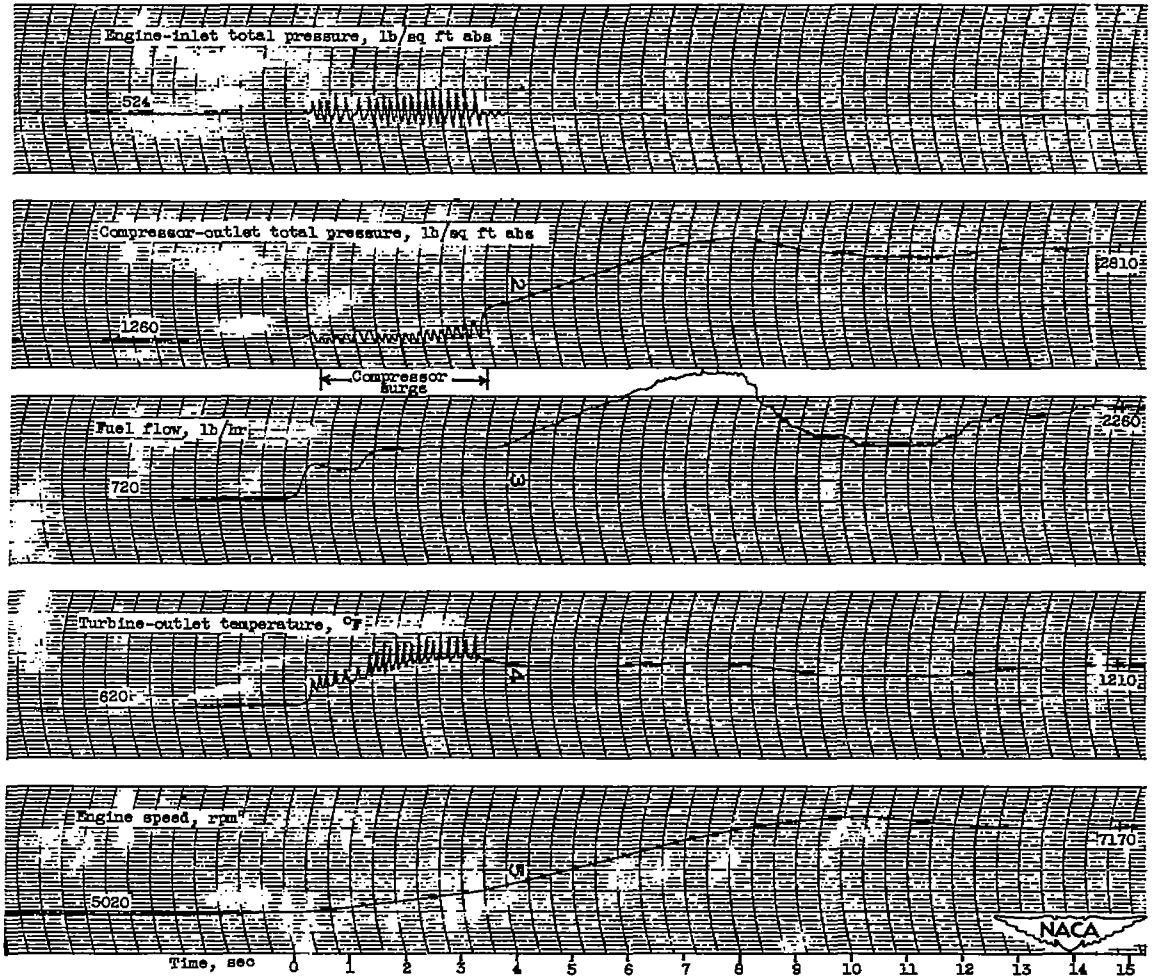
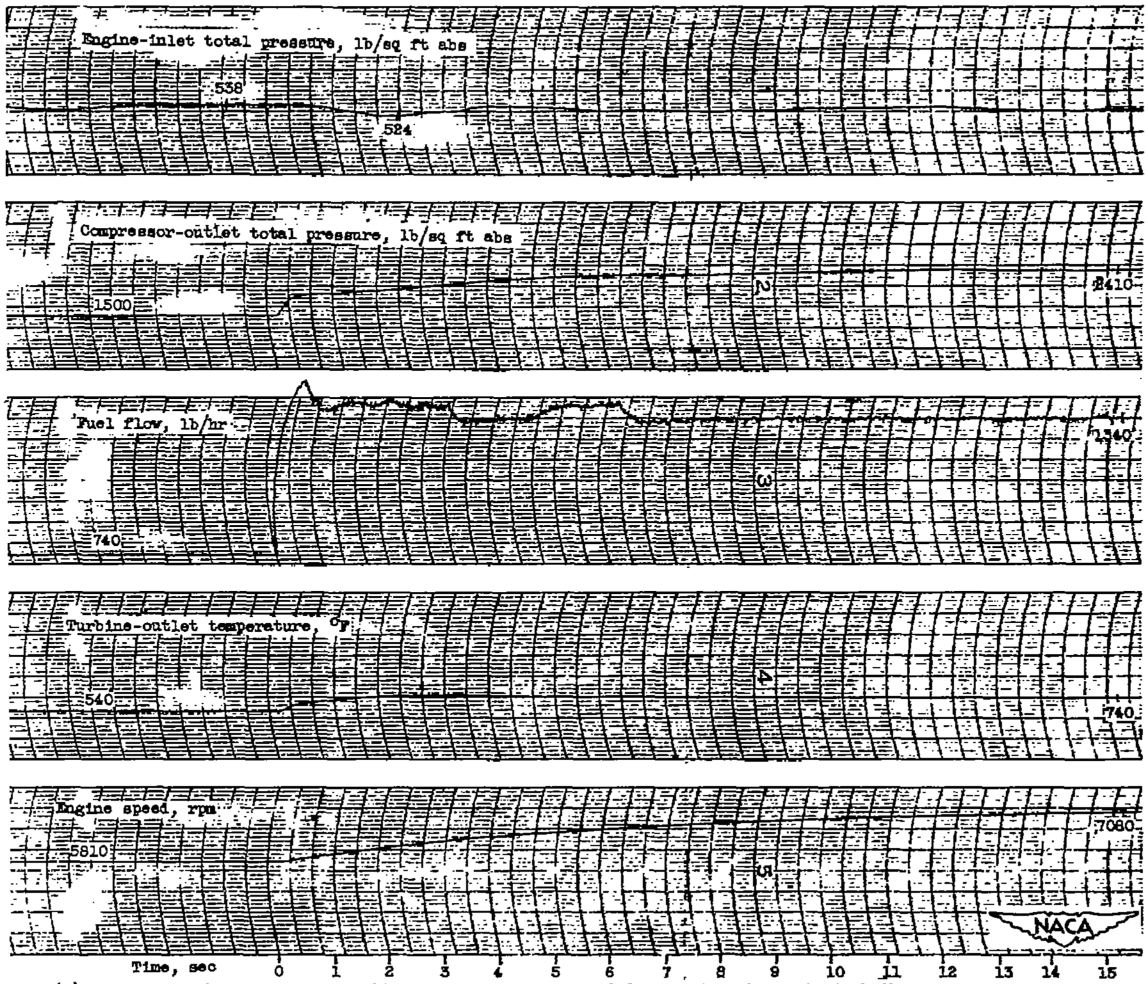


(b) Bellmouth engine-inlet installation; surge encountered during throttle burst to rated engine conditions.

Figure 4. - Continued. Oscillograph traces showing variations of different engine parameters during acceleration. Altitude, 35,000 feet; flight Mach number, 0.20.



(c) Ram-pipe engine-inlet installation; no surge encountered during step-change in fuel-flow acceleration run.

Figure 4. - Continued. Oscillograph traces showing variations of different engine parameters during acceleration. Altitude, 35,000 feet; flight Mach number, 0.20.

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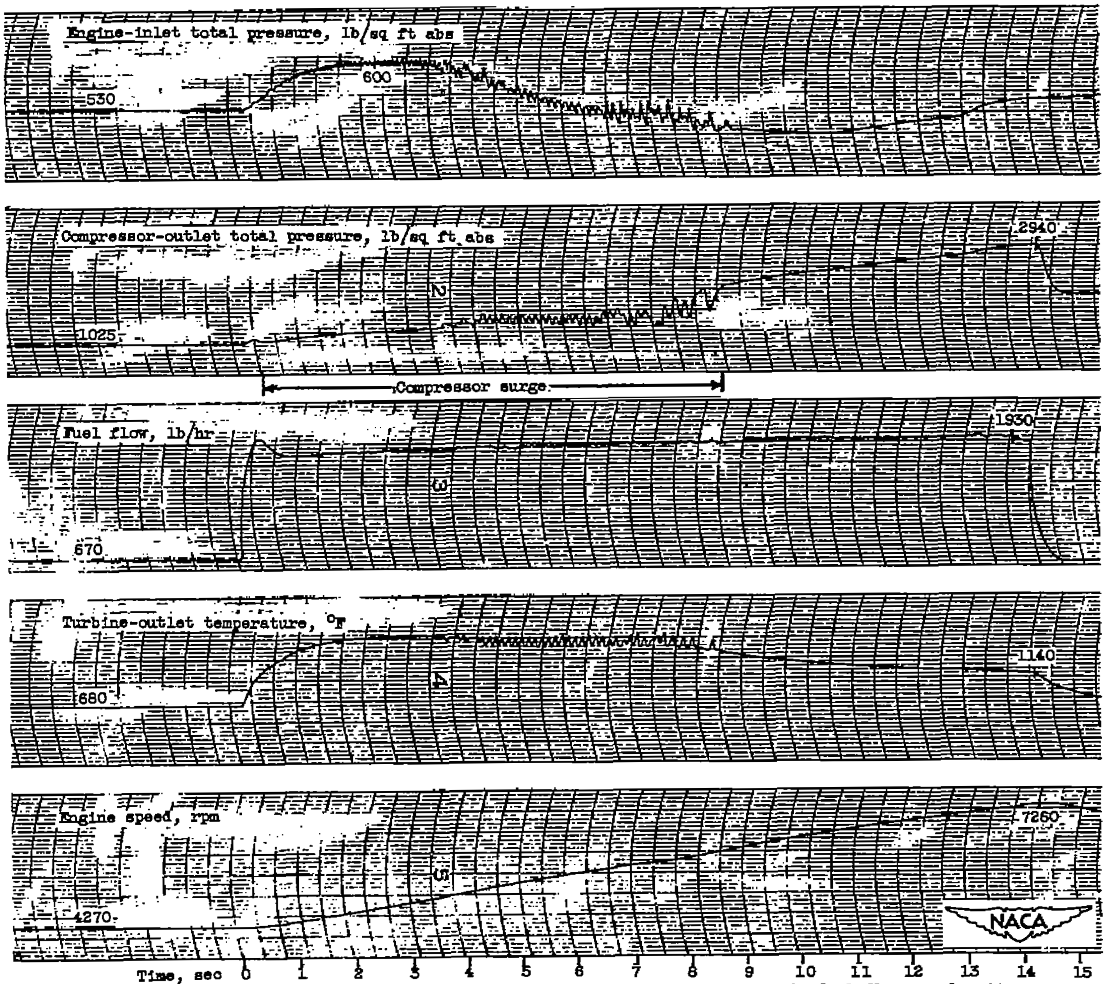

(d) Ram-pipe engine-inlet installation; surge encountered during step-change in fuel-flow acceleration run.

Figure 4. - Concluded. Oscillograph traces showing variations of different engine parameters during acceleration. Altitude, 35,000 feet; flight Mach number, 0.20.

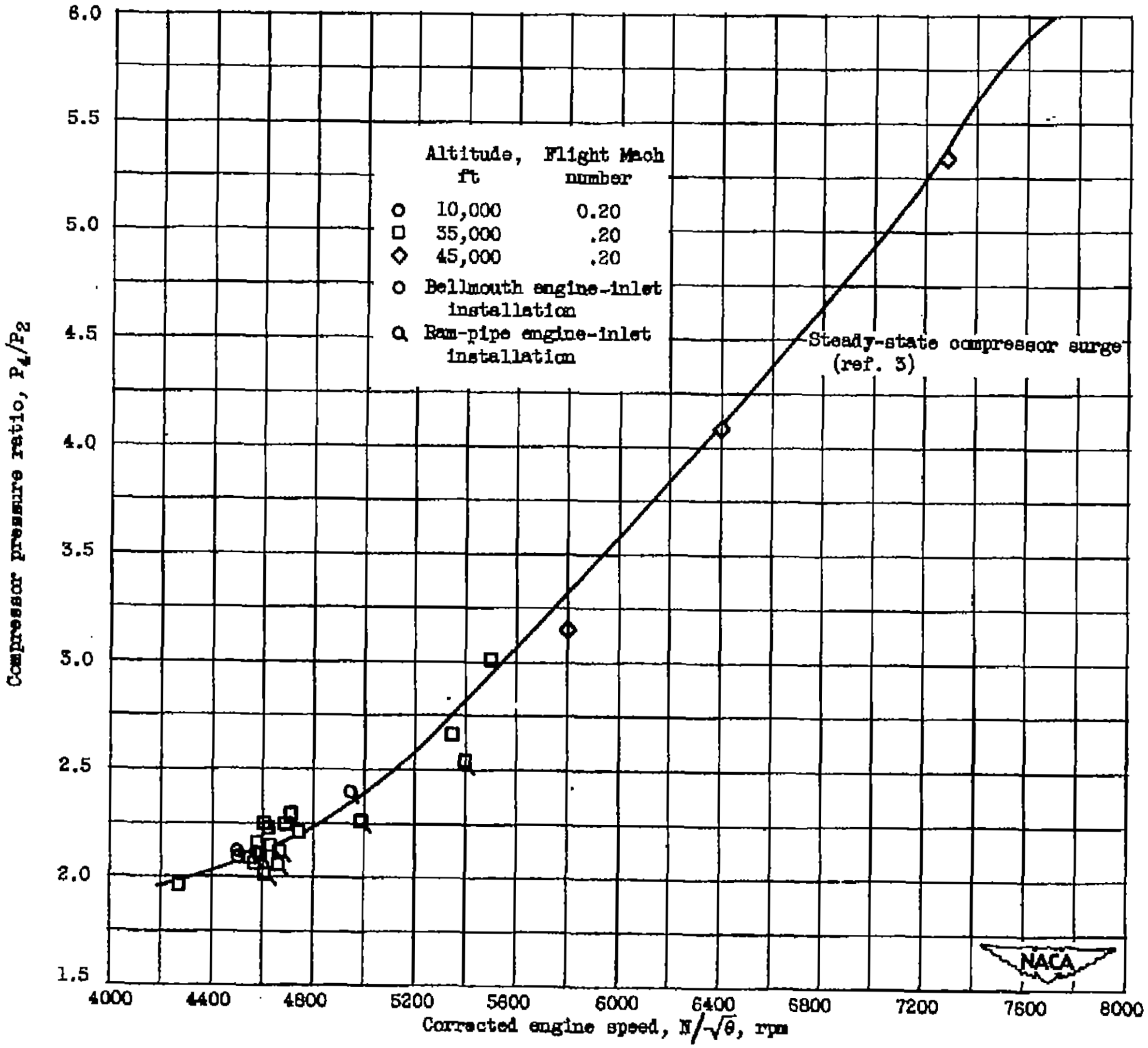

Figure 5. - Compressor surge limits. (Surge line represents data obtained at altitude of 30,000 ft at flight Mach number of 0.64 and at altitude of 45,000 ft at flight Mach number of 0.21.)

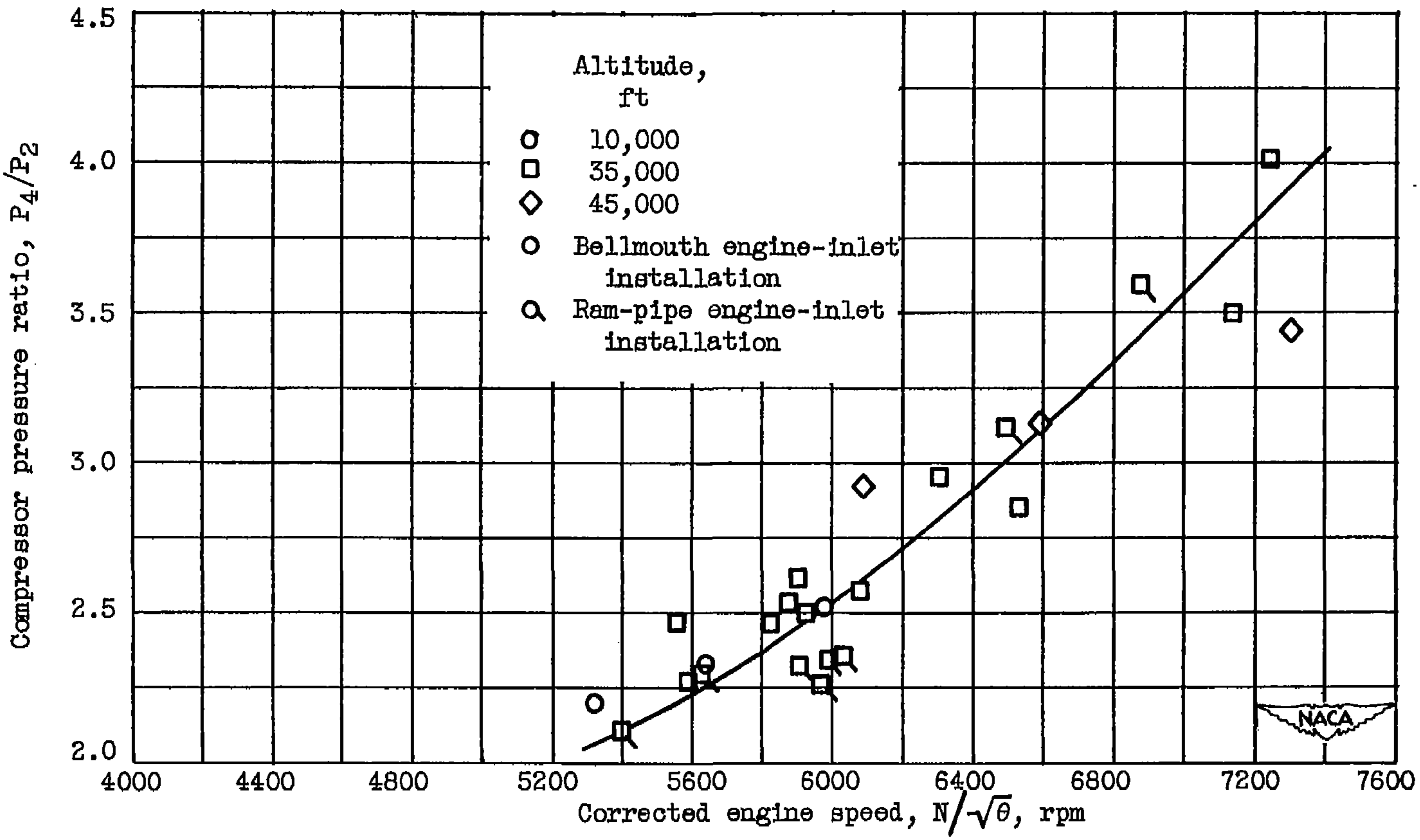

Figure 6. - Compressor recovery limits. Flight Mach number, 0.20.

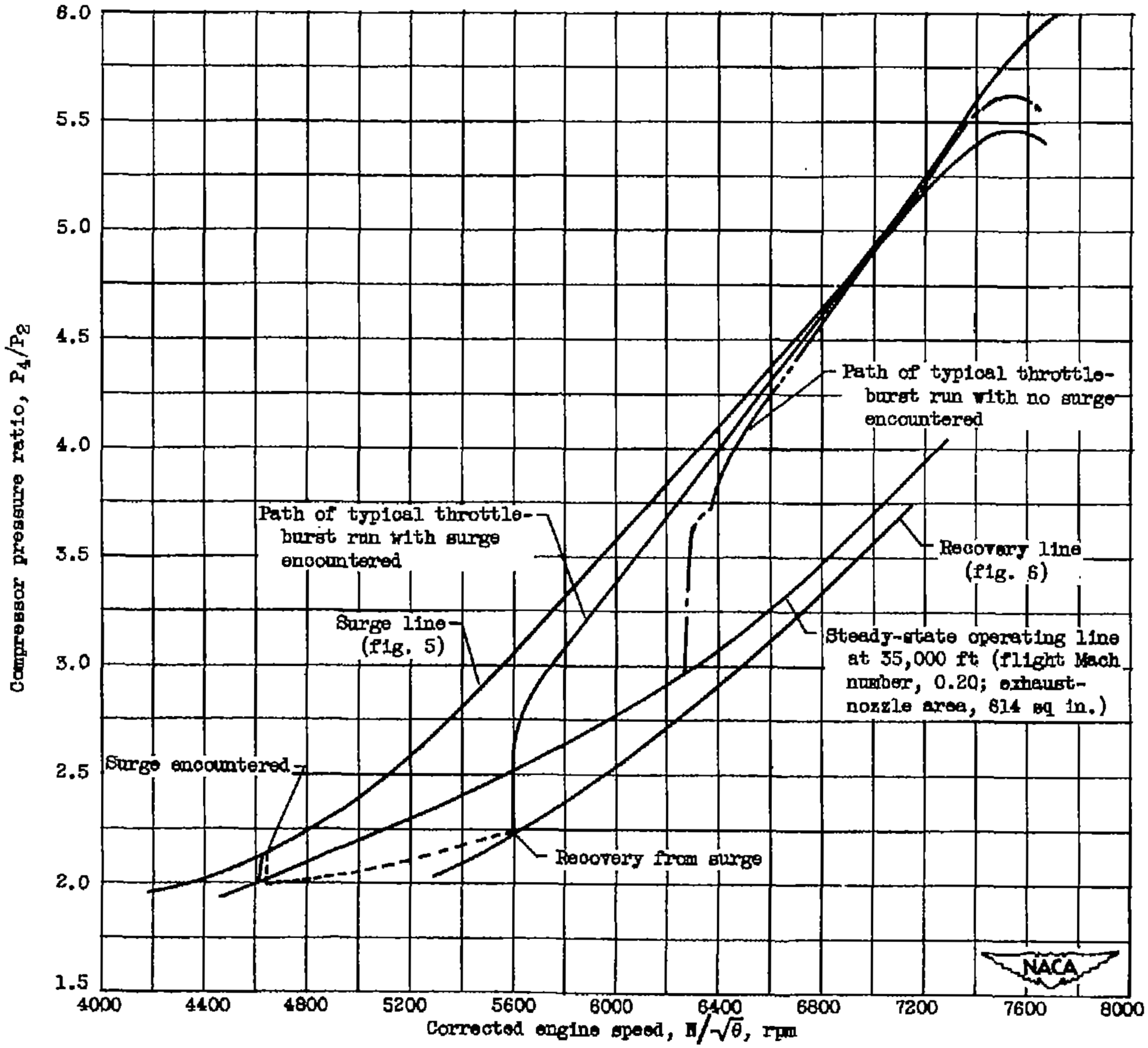

Figure 7. - Relation of compressor steady-state operating line to surge and recovery limits.

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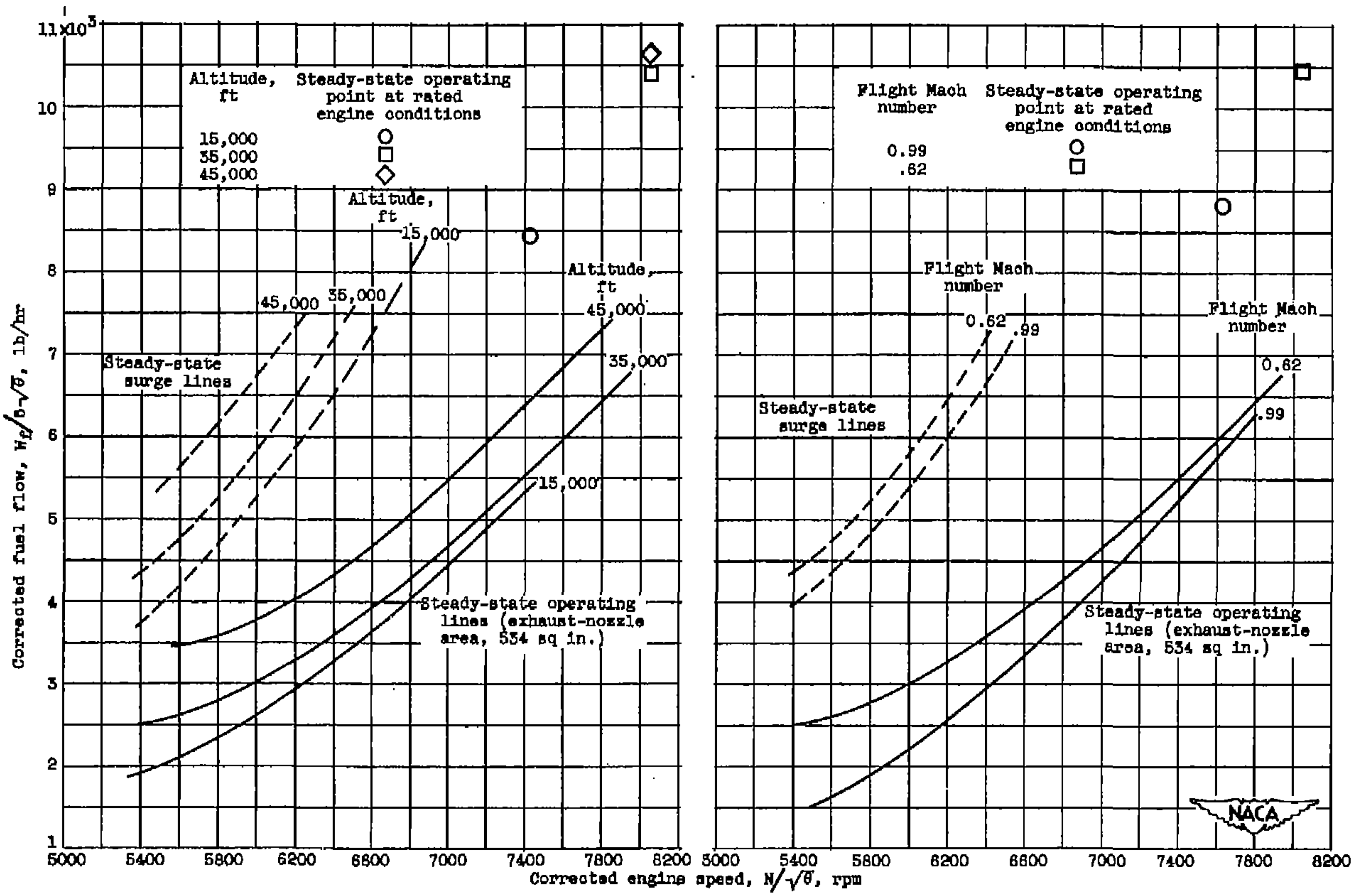

(a) Flight Mach number, 0.62.

(b) Altitude, 35,000 feet.

Figure 8. - Relation of engine steady-state operating line to surge line for different flight conditions.

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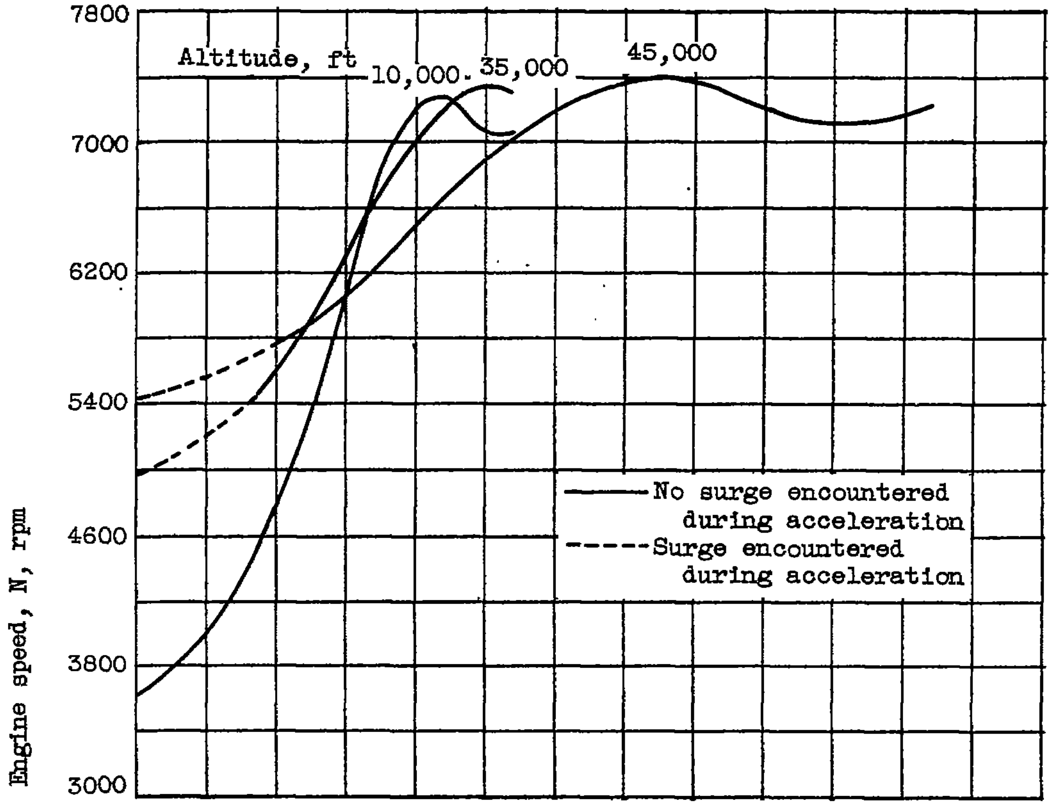
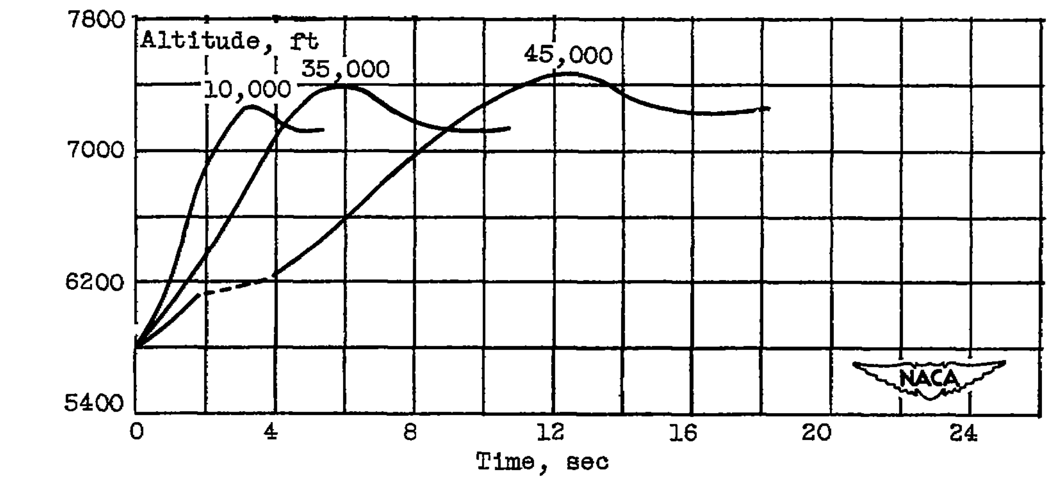


(a) Acceleration from idle engine speed.



(b) Acceleration from given initial engine speed.

Figure 9. - Effect of altitude on variation of engine speed with time during throttle bursts to rated engine conditions. Flight Mach number, 0.20.

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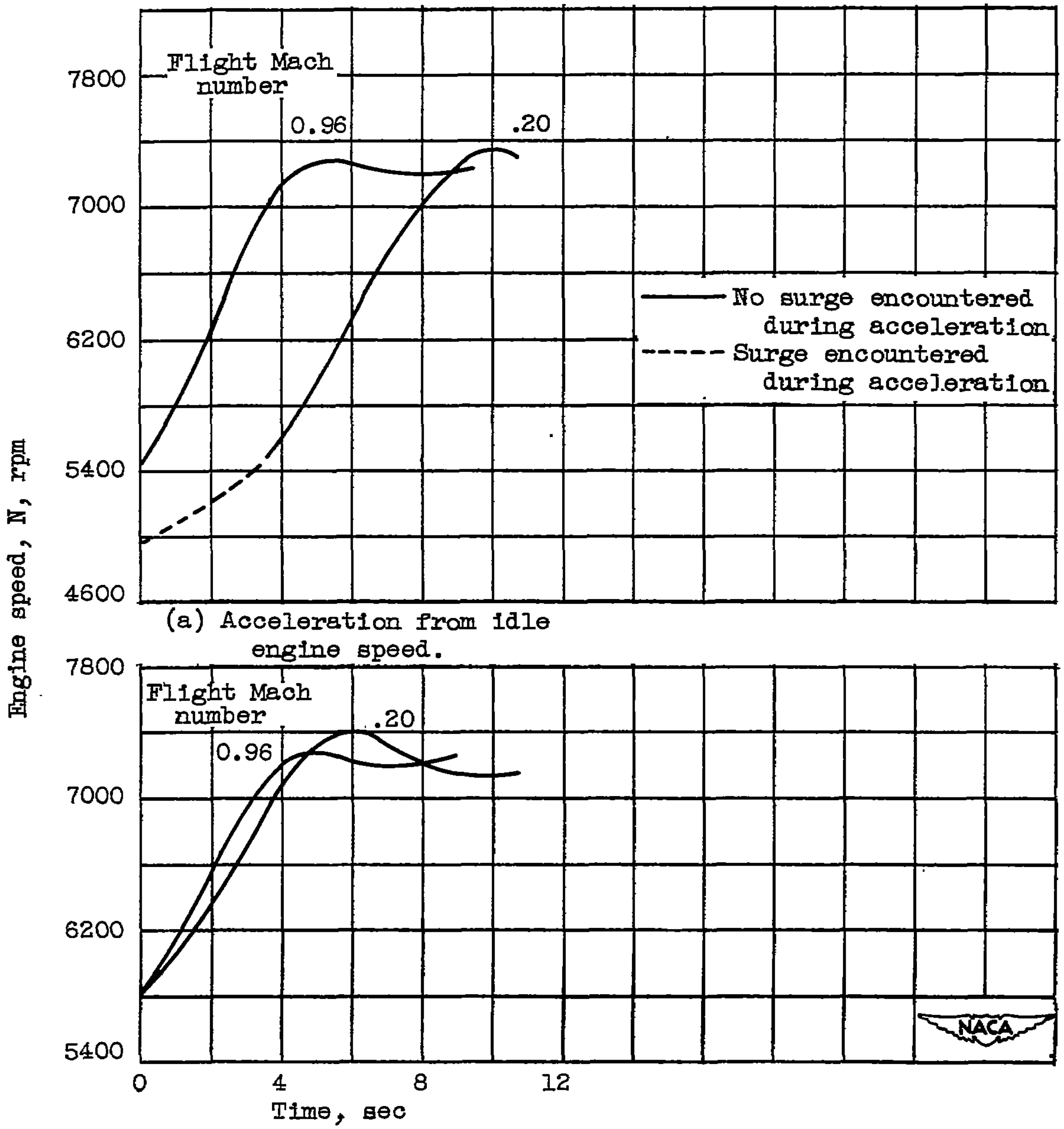

(a) Acceleration from idle engine speed.

(b) Acceleration from given initial engine speed.

Figure 10. - Effect of flight Mach number on variation of engine speed with time during throttle bursts to rated engine conditions. Altitude, 35,000 feet.

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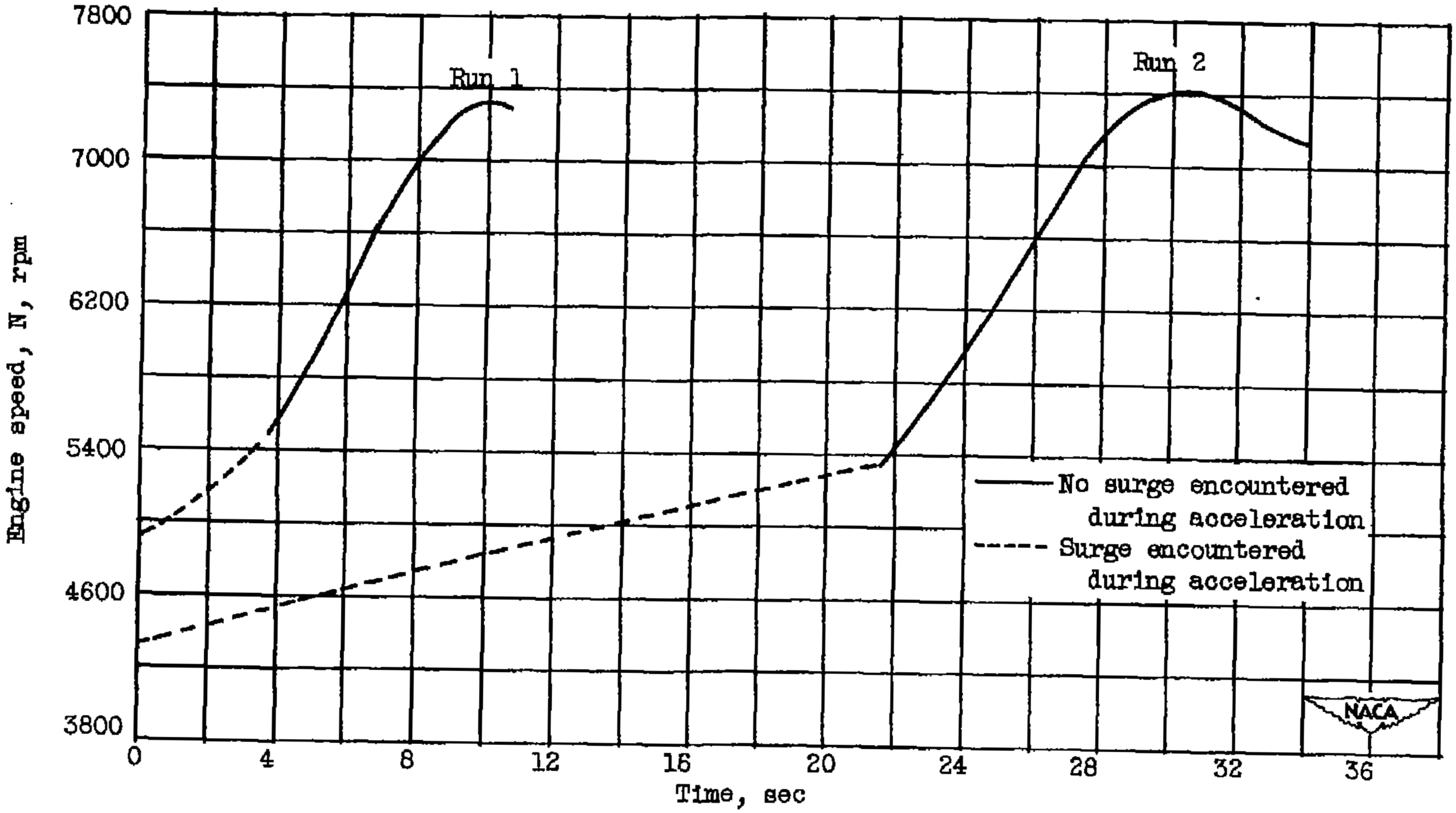

Figure 11. - Effect of compressor surge on variation of engine speed with time during throttle bursts to rated engine conditions. Altitude, 35,000 feet; flight Mach number, 0.20.

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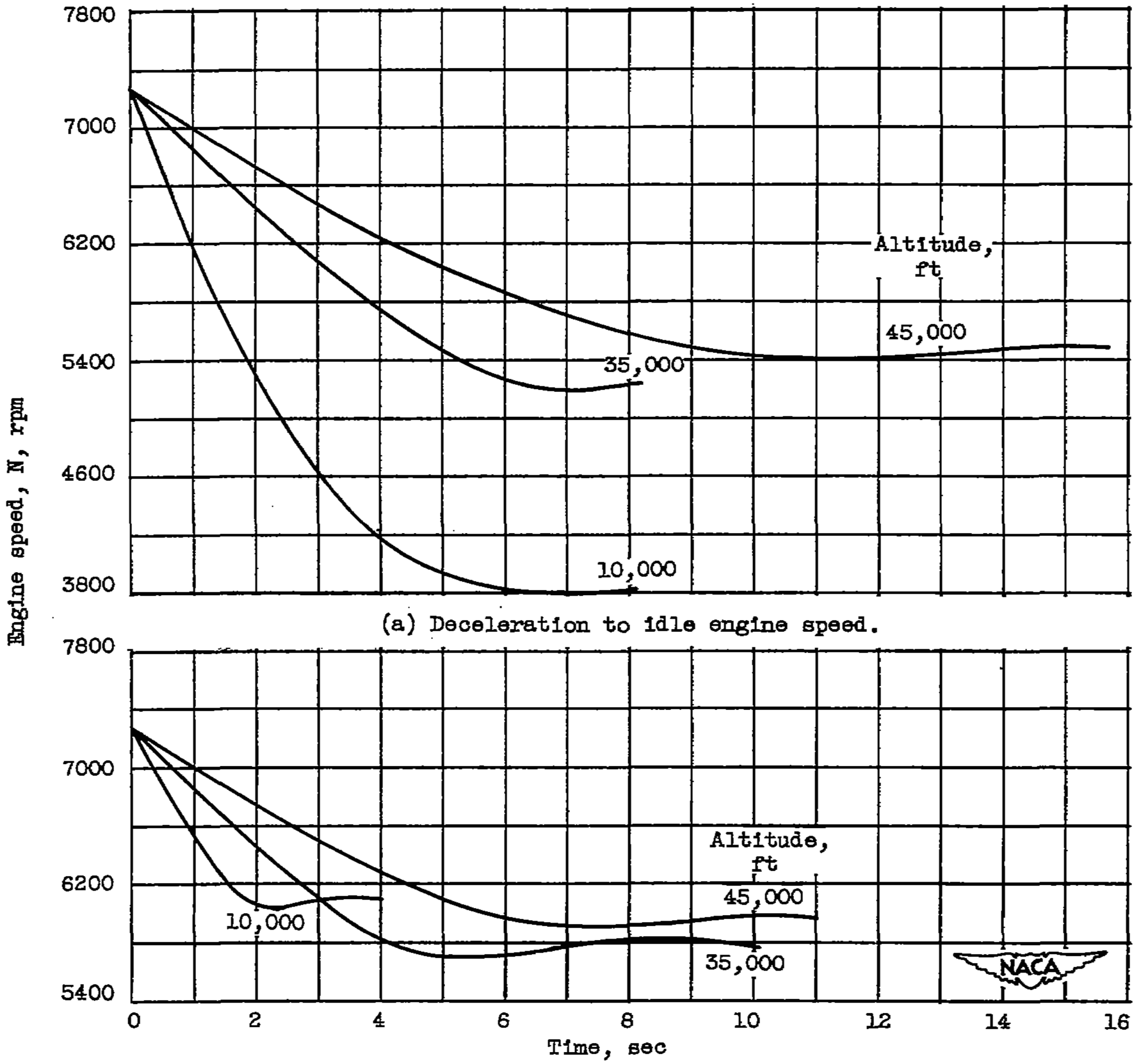

(b) Deceleration to given final engine speed.

Figure 12. - Effect of altitude on variation of engine speed with time during throttle chops from rated engine conditions. Flight Mach number, 0.20.

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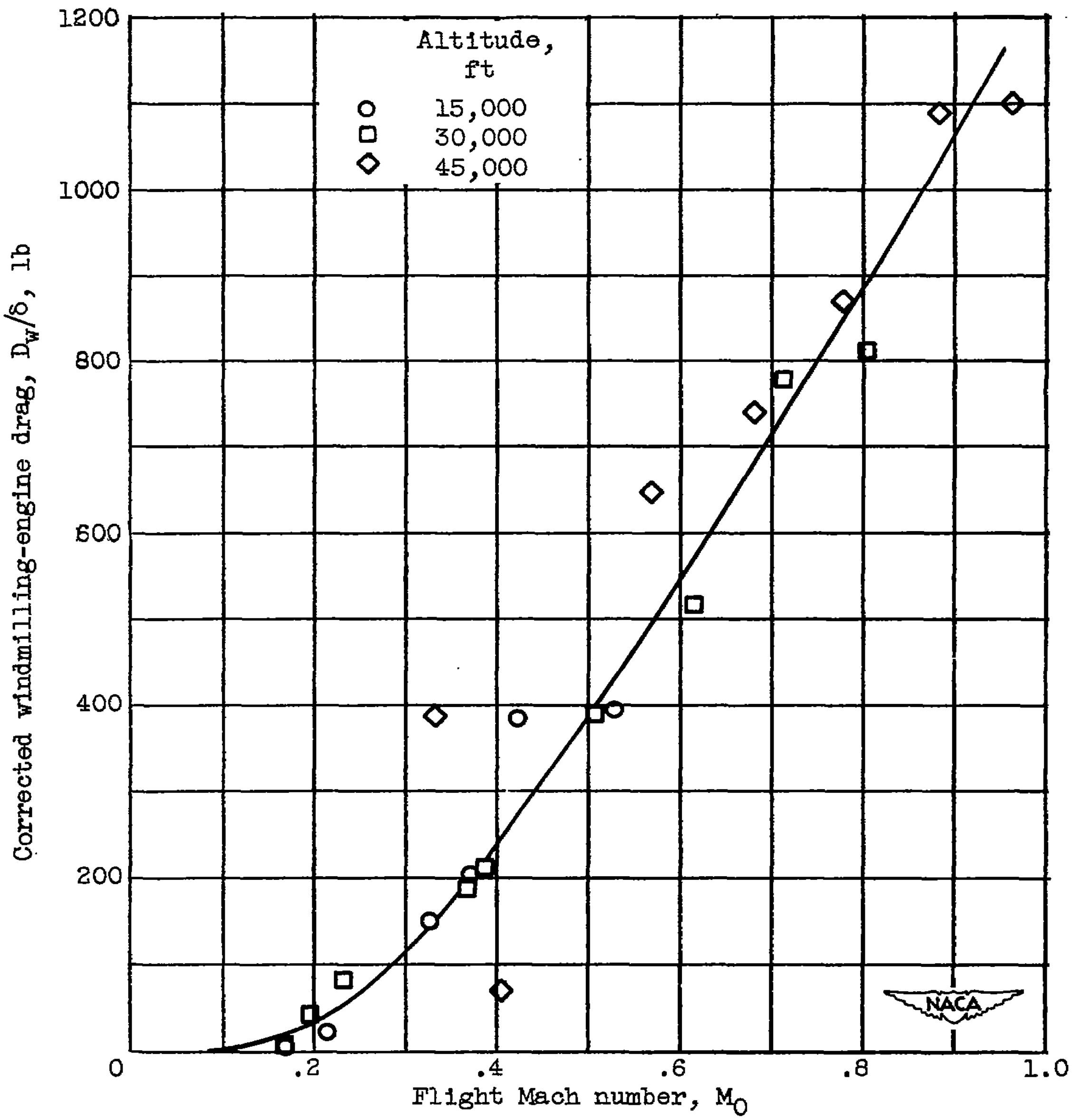

Figure 13. - Variation of steady-state windmilling-engine drag with flight Mach number. Exhaust nozzle full-open (534 sq in.).

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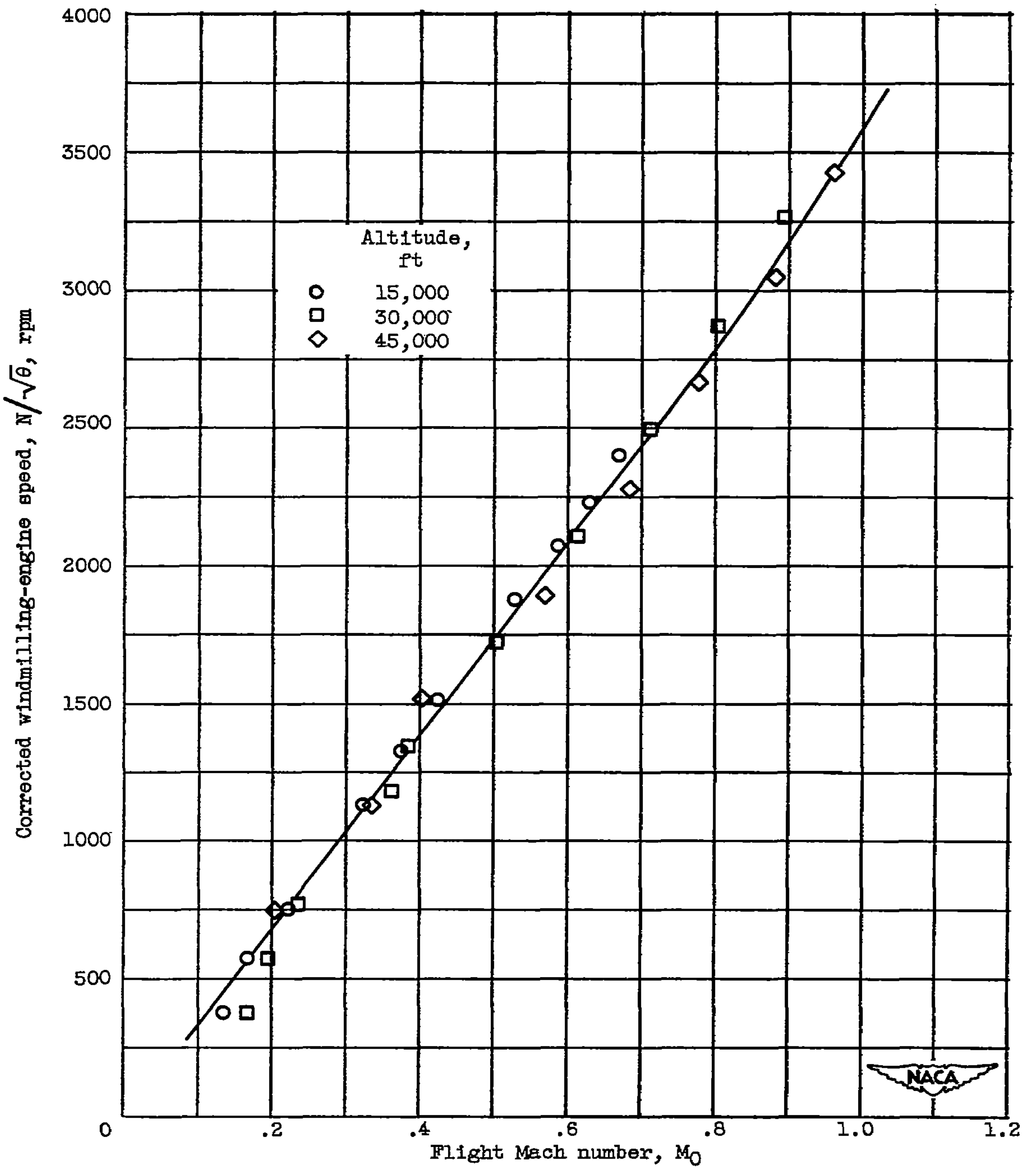

Figure 14. - Variation of steady-state windmilling-engine speed with flight Mach number. Exhaust nozzle full-open (534 sq in.).

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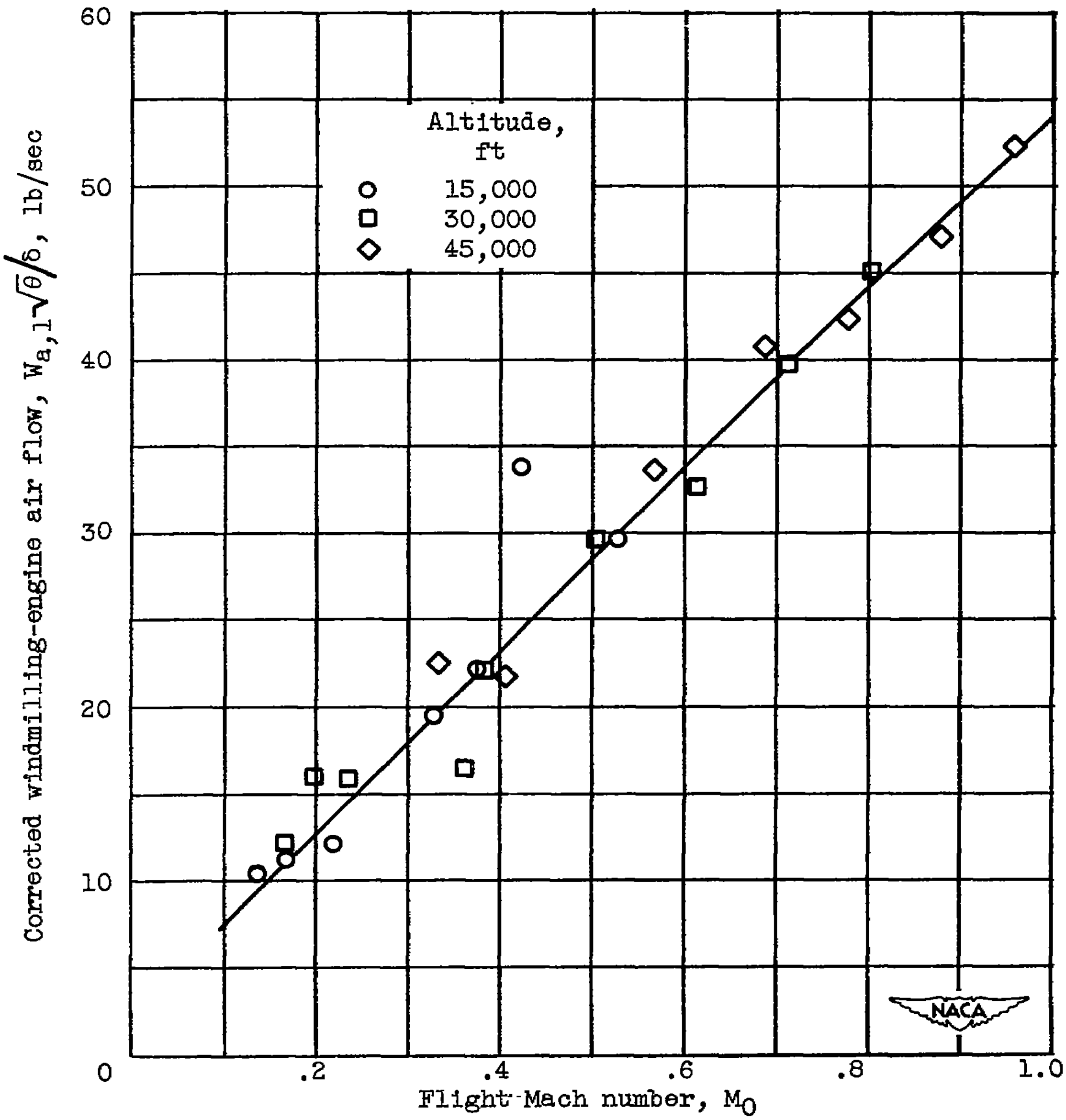

Figure 15. - Variation of steady-state air flow with flight Mach number. Exhaust nozzle full-open (534 sq in.).

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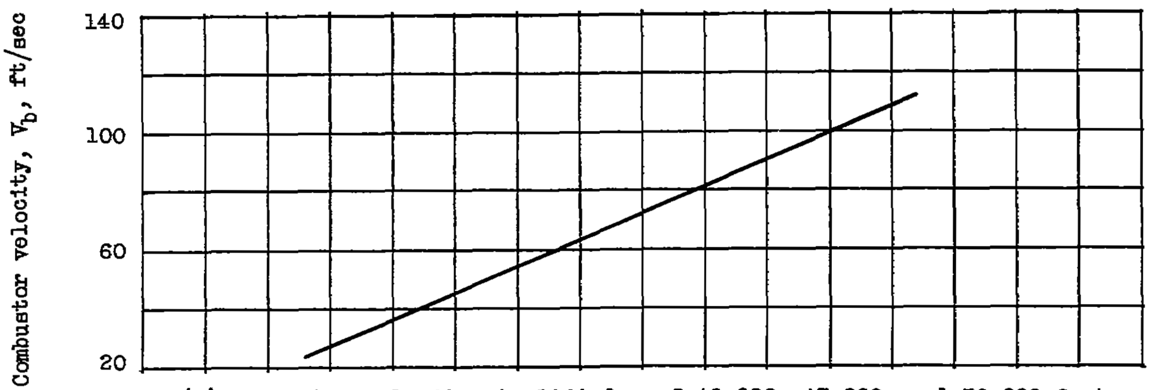
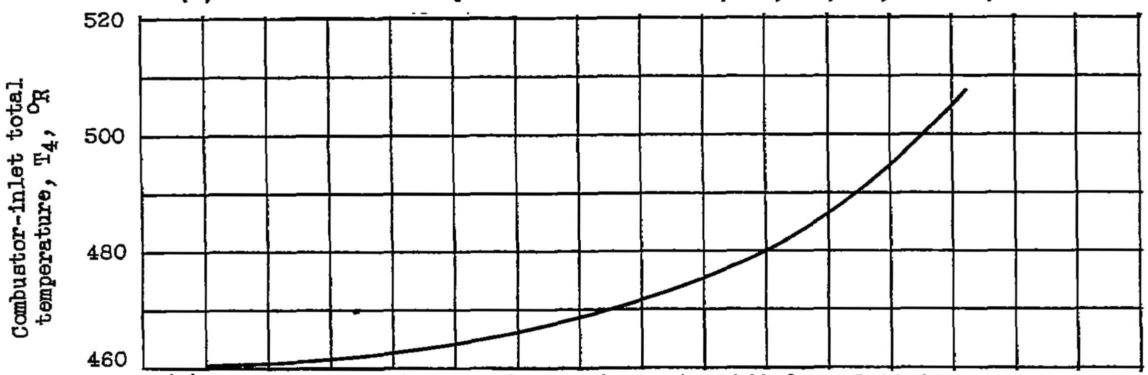
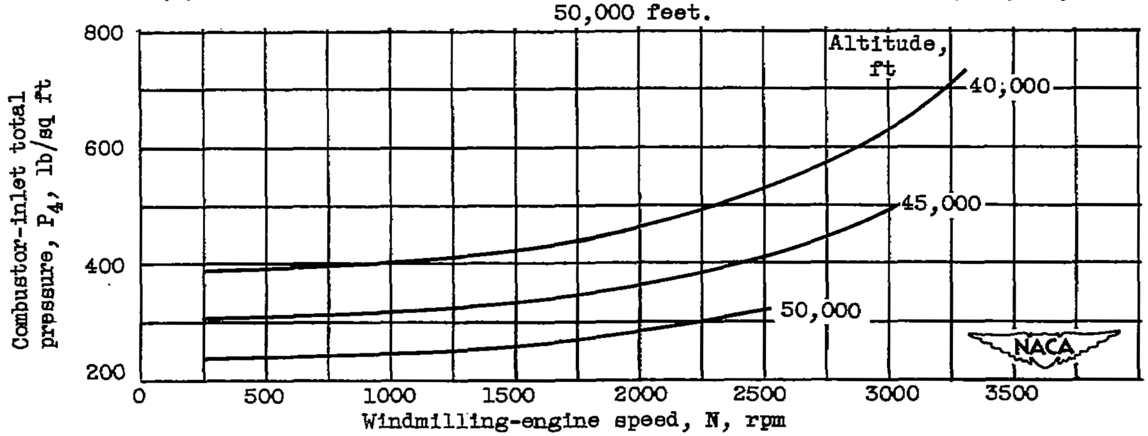


(a) Combustor velocity at altitudes of 40,000, 45,000, and 50,000 feet.



(b) Combustor-inlet total temperature at altitudes of 40,000, 45,000, and 50,000 feet.



(c) Combustor-inlet total pressure,

Figure 16. - Variation of steady-state windmilling-engine combustor-inlet conditions with steady-state windmilling-engine speed. Engine-inlet air temperature, 0° F.

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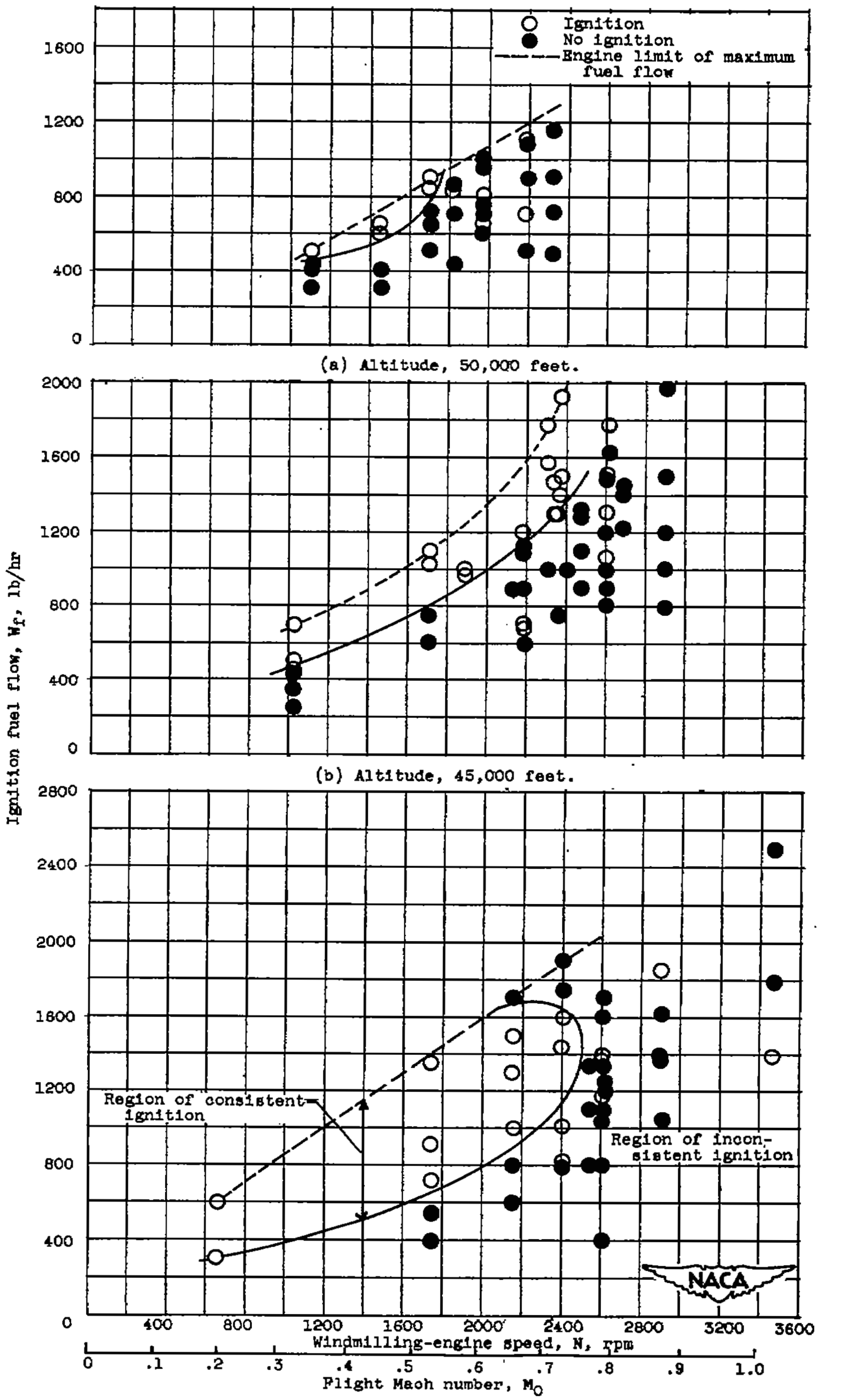

(c) Altitude, 40,000 feet.

Figure 17. - Effect of fuel flow on altitude ignition characteristics with MIL-F-5624A grade JP-4 fuel. Fuel temperature, approximately 50° F; engine-inlet air temperature, approximately 0° F. Ignition procedure A.

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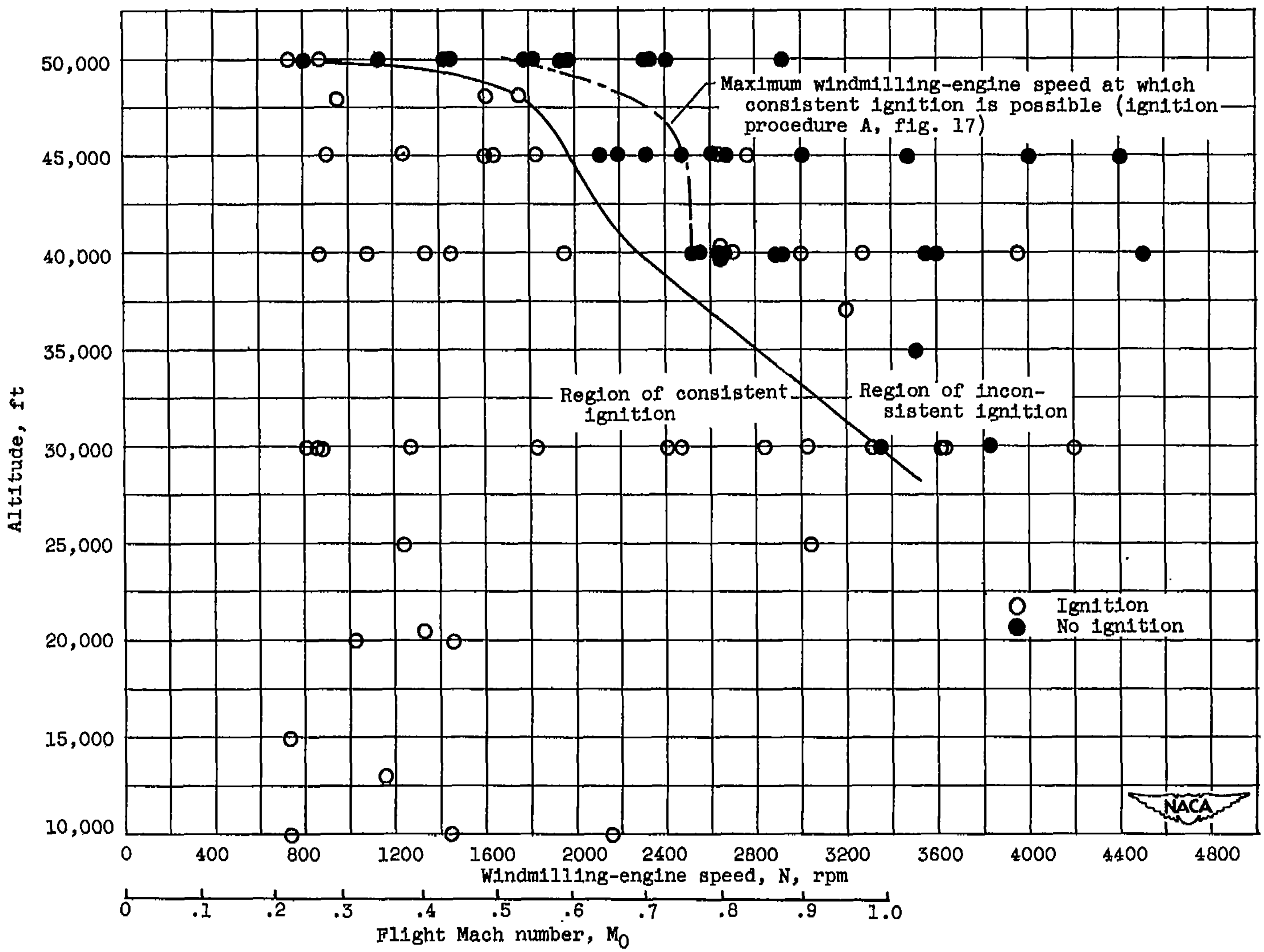

Figure 18. - Altitude ignition characteristics with MIL-F-5624 grade JP-4 fuel. Fuel temperature, approximately 50° F; engine-inlet air temperature, approximately 0° F. Ignition procedure B.

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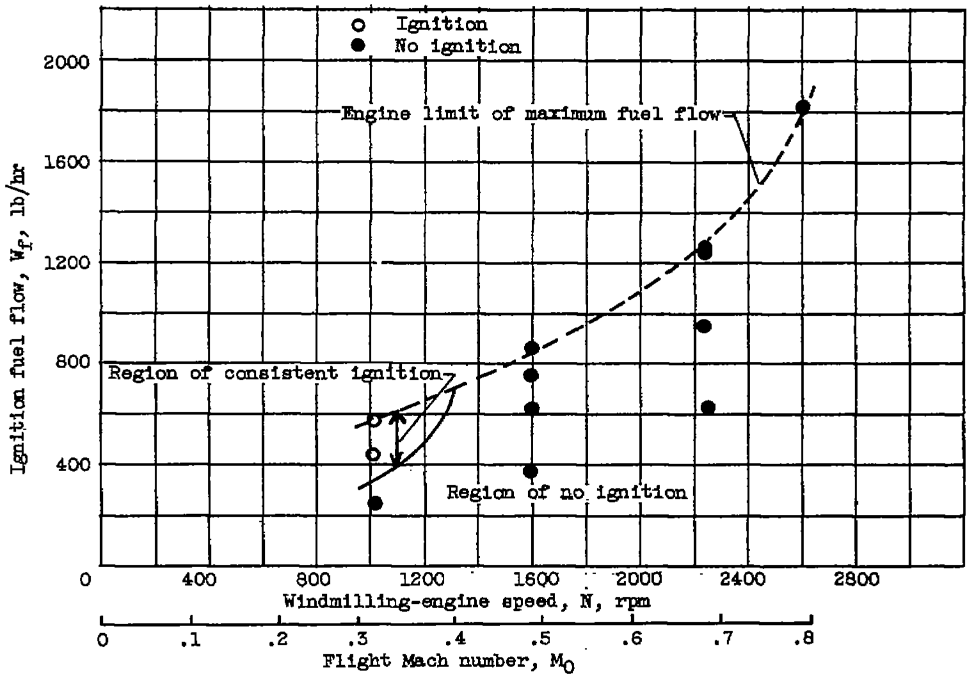


Figure 19. - Effect of fuel flow on altitude ignition characteristics with MIL-F-5624A grade JP-4 fuel. Fuel temperature, approximately -28° F; engine-inlet air temperature, approximately 0° F; altitude, 45,000 feet. Ignition procedure A.

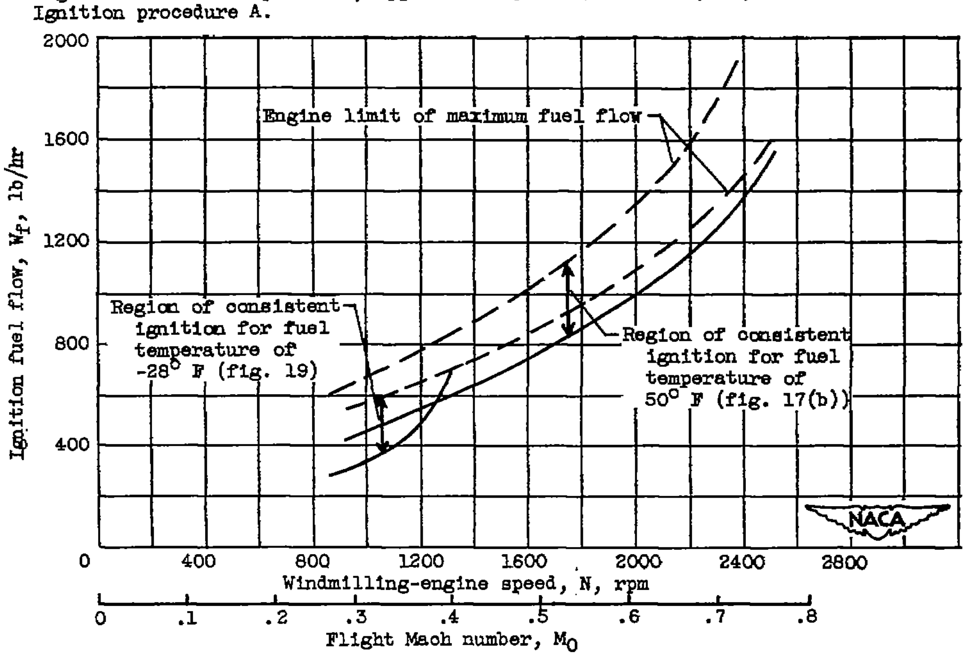


Figure 20. - Effect of fuel temperature on altitude ignition characteristics with MIL-F-5624A grade JP-4 fuel. Engine-inlet air temperature, approximately 0° F; altitude, 45,000 feet. Ignition procedure A.

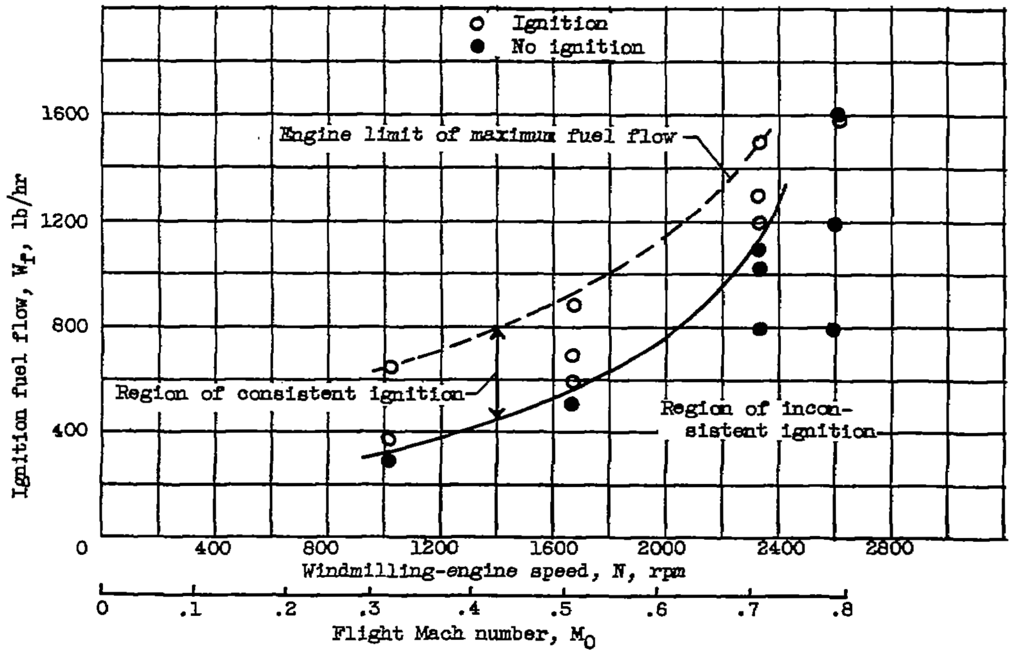


Figure 21. - Effect of fuel flow on altitude ignition characteristics with MIL-F-5624A grade JP-3 fuel. Fuel temperature, approximately 50° F; engine-inlet air temperature, 0° F; altitude, 45,000 feet. Ignition procedure A.

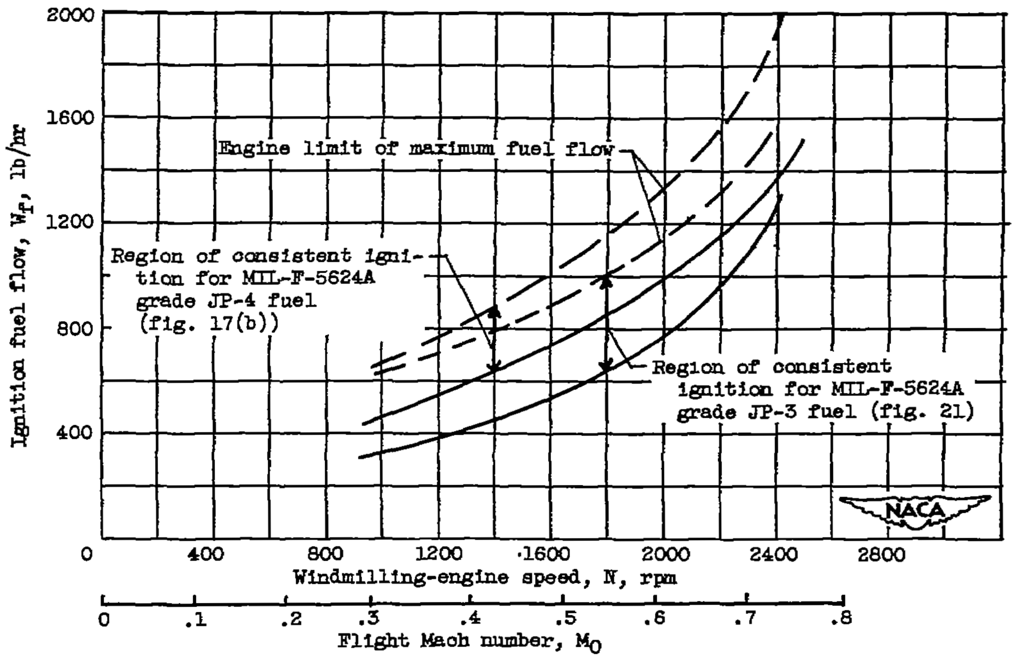
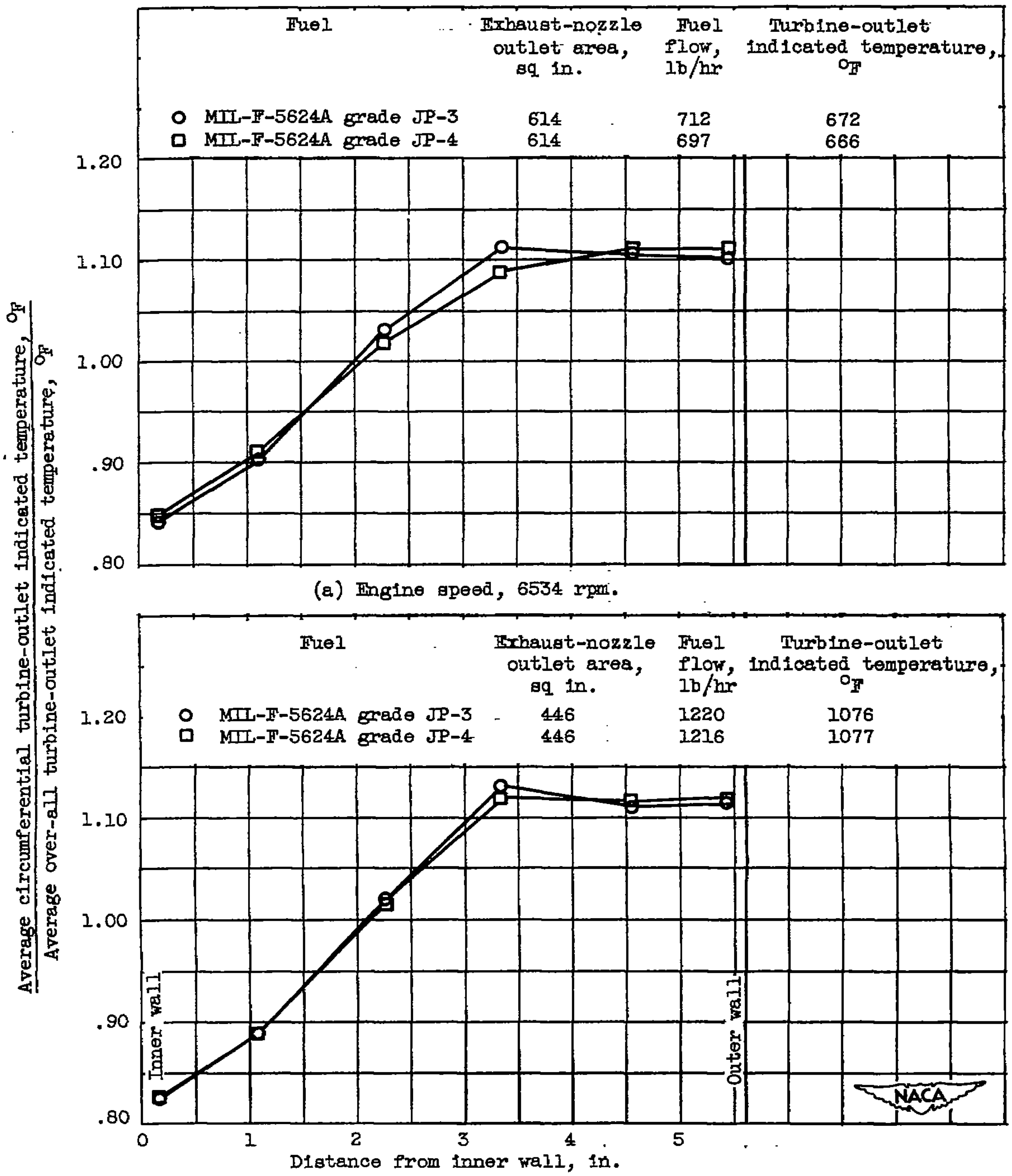


Figure 22. - Effect of two grades of fuel on altitude ignition characteristics. Fuel temperature, approximately 50° F; engine-inlet air temperature, approximately 0° F; altitude, 45,000 feet. Ignition procedure A.



(a) Engine speed, 6534 rpm.

(b) Engine speed, 7260 rpm.

Figure 23. - Effect of two grades of fuel on turbine-outlet temperature distributions. Altitude, 45,000 feet; flight Mach number. 0.20: engine-inlet air temperature, 0° F.

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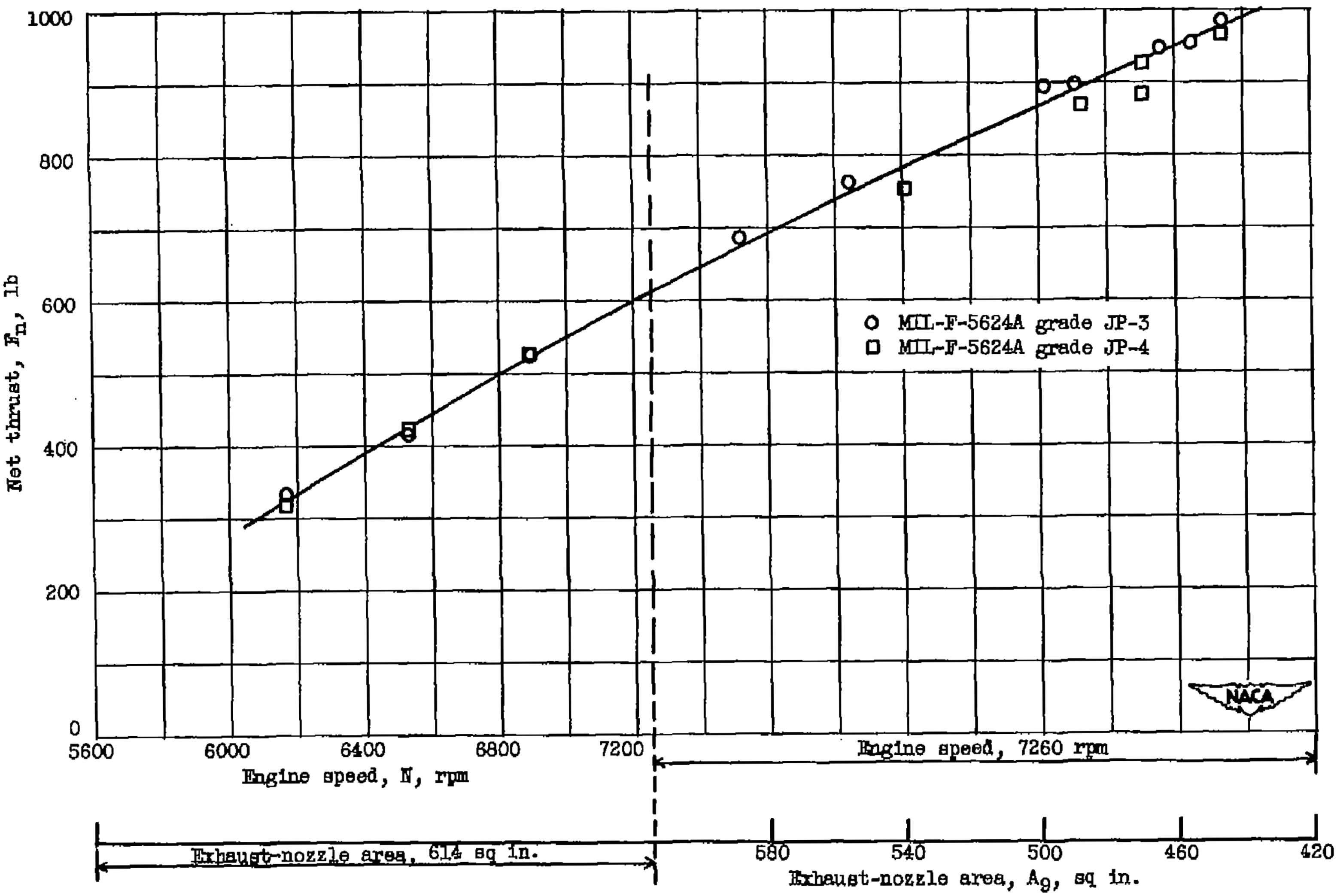

Figure 24. - Effect of two grades of fuel on net thrust. Altitude, 45,000 feet; flight Mach number, 0.20; engine-inlet air temperature, 0° F.

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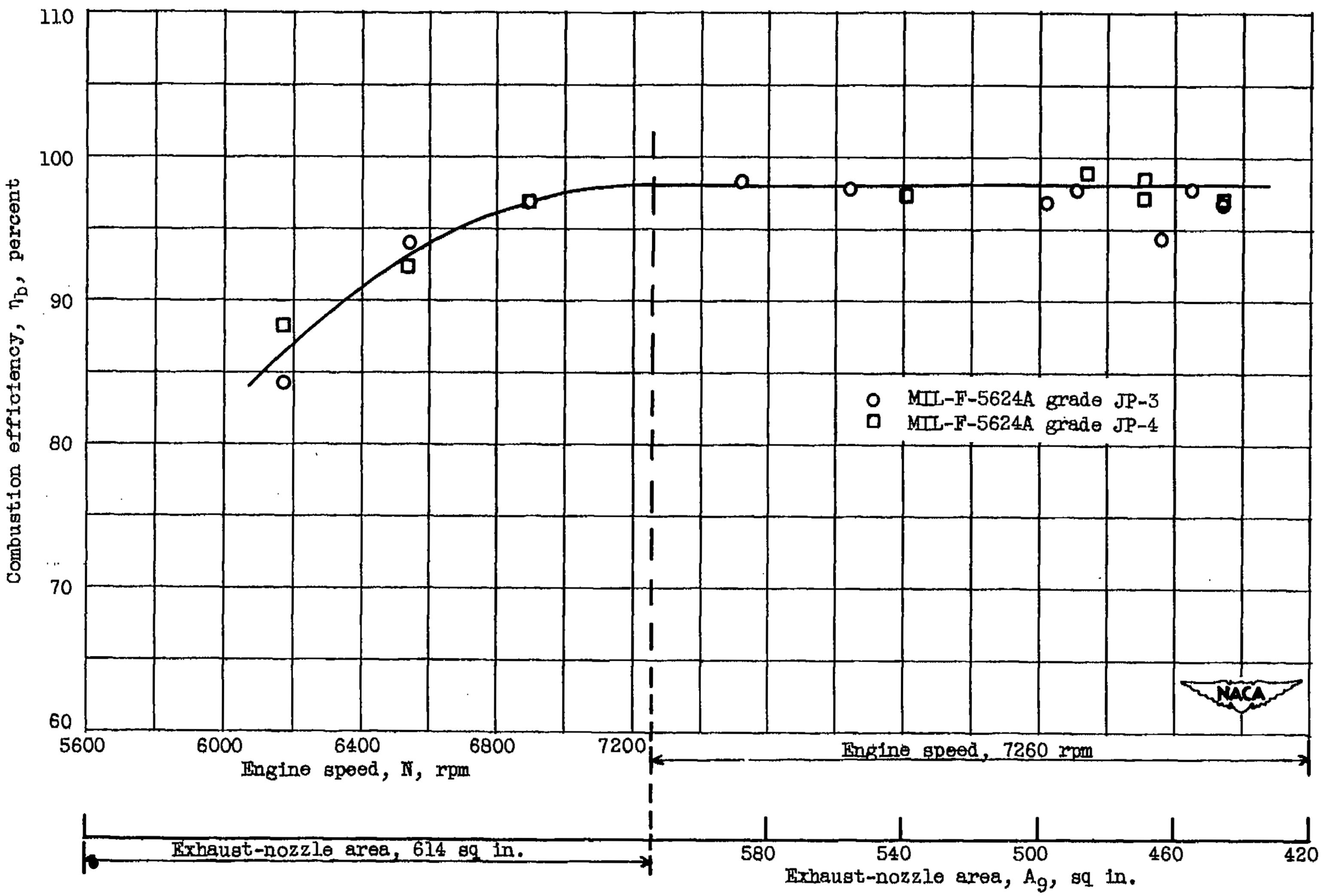

Figure 25. - Effect of two grades of fuel on combustion efficiency. Altitude, 45,000 feet; flight Mach number, 0.20; engine-inlet air temperature, 0° F.

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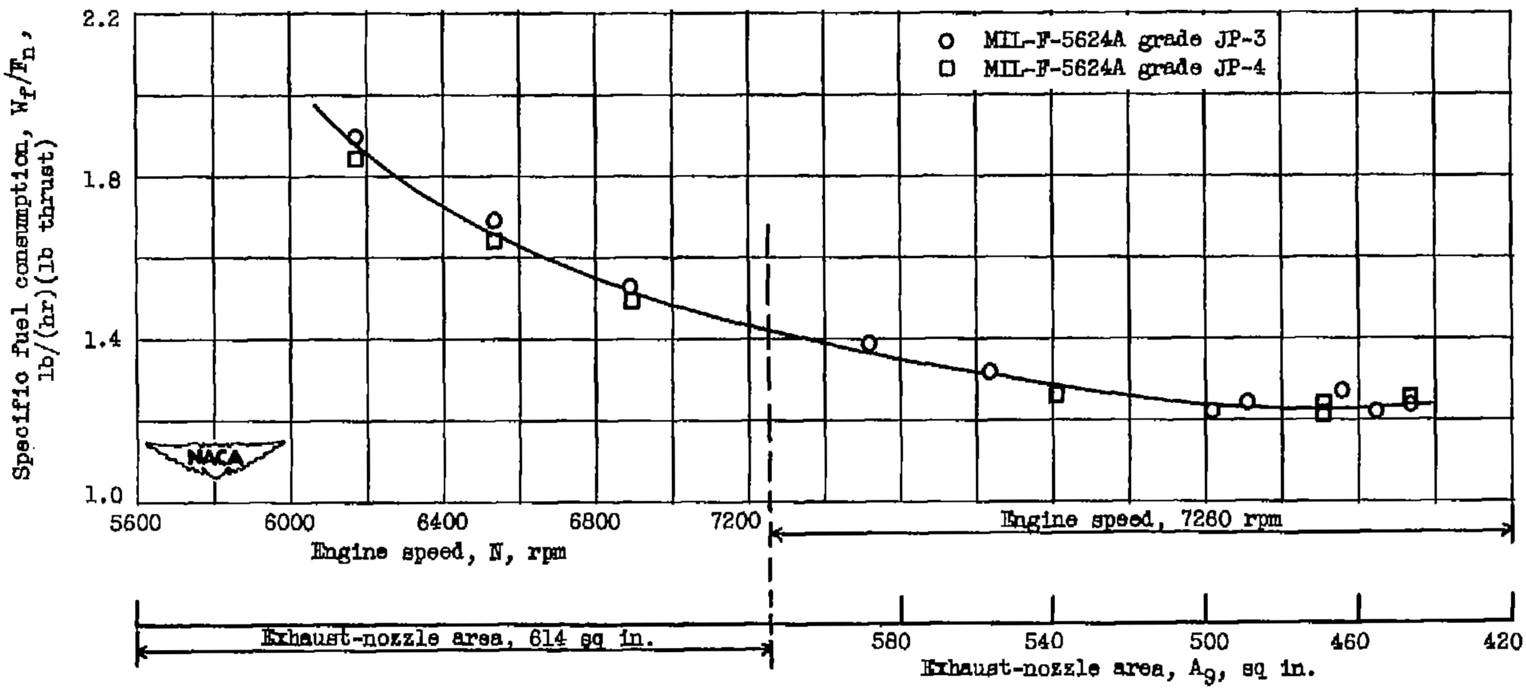

Figure 26. - Effect of two grades of fuel on specific fuel consumption. Altitude, 45,000 feet; flight Mach number, 0.20; engine-inlet air temperature, 0° F.

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SECURITY INFORMATION

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