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RESEARCH MEMORANDUM

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Bureau of Aeronautics, Department of the Navy

ALTITUDE-TEST-CHAMBER INVESTIGATION OF McDONNELL AFTERBURNER ON J34 ENGINE

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SUMMARY

An altitude-test-chamber investigation was conducted to determine the operational and performance characteristics of a McDonnell afterburner with a fixed-area exhaust nozzle on a J34 engine.

At rated engine speed, the altitude limit, as determined by combustion blow-out, occurred as a band of unstable operation of about 6000-foot altitude in width with minimum altitude limits from 31,000 feet at a simulated flight Mach number of 0.40 to about 45,500 feet at a simulated flight Mach number of 1.00. Considerable difficulty was experienced in attempting to establish or maintain balanced-cycle engine operation at altitudes above 36,000 feet.

The fuel-air ratio for balanced-cycle operation and lean blow-out of the afterburner, the augmented-thrust ratio, the total specific fuel consumption, and the afterburner combustion efficiency for balanced-cycle operation are summarized in the following table:

Altitude (ft)	Flight Mach number	Afterburner fuel-air ratio		Augmented-thrust ratio (a)	Total specific fuel consumption (lb/(hr)(lb net thrust))	Afterburner combustion efficiency (percent)
		Balanced-cycle operation	Lean blow-out			
20,000	0.60	0.041	0.011	1.80	2.43	75
20,000	1.00	.036	.012	2.33	2.24	81
35,000	1.00	.047	.012	2.03	2.59	55

[a]Augmented-thrust ratios were obtained from augmented thrust at limiting turbine-outlet temperature and normal thrust at turbine-outlet temperature as much as 150° F lower than limiting value.

Satisfactory afterburner ignition was obtained over a range of flight Mach numbers from 0.32 to 0.60 at altitudes from 10,000 to 30,000 feet and engine speeds from 10,000 to 12,500 rpm.

INTRODUCTION

At the request of the Bureau of Aeronautics, Department of the Navy, an investigation has been conducted in a 10-foot altitude test chamber at the NACA Lewis laboratory to determine the operational and performance characteristics of an afterburner manufactured by the McDonnell Aircraft Corporation and used with a J34 turbojet engine.

The altitude blow-out limits at various engine operating conditions are presented for a range of simulated flight Mach numbers from 0.50 to 1.00. The operational range of afterburner fuel flow (and fuel-air ratio), the augmented thrust ratio, the specific fuel consumption, and the afterburner combustion efficiency are given for approximately zero-ram conditions at an altitude of 5000 feet and for a range of flight Mach numbers from 0.60 to 1.00 at altitudes of 20,000, 30,000, and 35,000 feet. The ignition characteristics of the afterburner, the stability of afterburner operation, and the several structural failures that occurred during the investigation are also discussed.

APPARATUS AND INSTRUMENTATION

A sketch of the McDonnell afterburner is shown in figure 1. The afterburner has an over-all length of 51 inches and a maximum internal diameter of $25\frac{3}{4}$ inches. The original configuration had a fixed exhaust-nozzle area with an equivalent diameter of $18\frac{5}{8}$ inches. This diameter was modified to an equivalent diameter of 18 inches by welding on a conical extension piece that increased the length of the inner shell about 7/8 inch. The modified configuration was used throughout the experiments. Shell cooling of the burner section is accomplished by an ejector cooling jacket, which uses the exhaust jet to induce a flow of cooling air over the burner inner shell. In this installation, the cooling air for the burner cooling jacket was obtained from the test section of the altitude chamber at a pressure approximately equal to the altitude ambient pressure. A radiation shield is installed between the burner shell and the cooling jacket. Three sheltered zones, which serve as flame seats, are provided by a step in the outer shell, a step in the inner cone, and a blunt end on the inner cone. The total flame-seating surface represents 22.6 percent of the cross-sectional area within the burner shell. Fuel is provided from two manifolds, both of which are located near the turbine-outlet flange. One manifold is installed around the outer shell of the burner and the other inside the inner cone as shown in figure 1. The fuel is injected perpendicular to the gas stream through 27 tubes of 1/8-inch diameter in each manifold. All the tubes are uniformly pinched at the ends to provide an injection orifice. Independent control of the fuel to each manifold is provided.

The ignition system is composed of two spark plugs and a pilot fuel-supply line to each plug. One spark plug is located at the step in the outer shell and a second spark plug is located at the step in the inner cone. Both spark plugs are the single-electrode type for which the spark occurs between the electrode and the adjacent burner structure. A separate pilot fuel line is connected to each fuel manifold and the two lines are arranged to inject fuel a short distance upstream of the flame seats and in line with both the inner and outer spark plugs. The pilot fuel line for the inner spark plug is shown in figure 1. Ignition of the rich fuel-air mixture thus provided at both spark plugs results in small pilot flames from which the main fuel supply is ignited. No provision is made to shut off the pilot fuel flow after combustion of the main fuel supply is established.

The investigation of the performance of the McDonnell afterburner was conducted with the burner installed on a modified J34 turbojet engine having a rated engine speed of 12,500 rpm and a rated thrust of 3000 pounds at zero-ram sea-level conditions. For the investigation of the performance of the engine without the afterburner, the engine was equipped with a short tail pipe and an NACA-designed variable-area exhaust nozzle. This nozzle was locked in the position that provided limiting turbine-outlet temperature at static sea-level conditions at rated engine speed.

The fuel used in both the afterburner and the engine was 62-octane gasoline.

The general arrangement of the engine setup in the altitude test chamber is shown in figure 2. A photograph of the engine with the afterburner installed in the test section of the chamber is presented in figure 3. Details of the engine setup and the altitude chamber are given in reference 1.

A periscope was installed in such a manner that operation of the afterburner could be observed from the control room.

The engine fuel flow, the afterburner fuel flow, the air flow, the engine speed, and the pressure and temperature measurements at various stations in the engine, the afterburner, and the test chamber were measured with standard instrumentation. The engine thrust was balanced and measured through a lever arrangement by a null-type air-pressure diaphragm. The forces introduced into the thrust-measuring system by the pressure differences across the two baffles in the test chamber (fig. 2) were determined by calibration. An indication of the turbine-outlet temperature was provided by three stagnation-type thermocouples installed by the engine manufacturer. The readings of these thermocouples were calibrated against the calculated turbine-inlet temperature (based on engine fuel-air ratio, compressor-outlet temperature, and an assumed engine combustion efficiency of 97.5 percent) in order to provide an indication of the engine manufacturer's specified maximum turbine-inlet temperature. In addition to the engine manufacturer's thermocouples, three rakes of nine thermocouples each were installed about $1\frac{1}{2}$ inches downstream of the turbine outlet. An average reading of 1590° R from these 27 thermocouples corresponded to the limiting turbine-outlet temperature indicated by the manufacturer's thermocouples. The manufacturer's thermocouples were used to provide a convenient indication of turbine-outlet temperature during each run, but an average of the 27 thermocouples was used for actual temperature measurements presented herein.

PROCEDURE

Altitude Blow-Out Limits

The altitude blow-out limit of operation of the afterburner was investigated over a range of flight Mach numbers from 0.50 to 1.00 at rated engine speed of 12,500 rpm. The afterburner was usually started at an altitude of approximately 10,000 feet, a ram pressure ratio of 1.07 across the engine, and an engine speed of 10,500 rpm. After ignition, the fuel flow to the individual burner manifolds was adjusted to a ratio that would provide the most satisfactory combustion, as viewed through the periscope.

As soon as stable operation of the afterburner was established, the engine speed was increased to the rated value; while a fixed pressure differential was maintained across the engine, the pressure altitude was increased until blow-out occurred. This procedure was repeated for a range of fixed pressure differences to simulate variation in flight Mach number. The engine-inlet total air temperatures were preset to values corresponding to NACA standard air at the altitude and Mach number at which blow-out was expected to occur on the basis of information obtained in preliminary exploratory runs. Engine-inlet temperature was from 10° to 30° F higher than the desired NACA standard value at blow-out.

Balanced-cycle engine operation (limiting turbine-outlet temperature at rated engine speed) was maintained as nearly as possible during the run by adjustment of engine fuel flow and afterburner total fuel flow. Blow-out was detected by the change in noise level of the engine, the surge of operating conditions, such as altitude pressure and engine speed, the decrease of turbine-outlet temperature, and by visual observation of the flame. The blow-out limit was considered to be the altitude at which the fuel ceased to burn at either the outer flame seat or at the two flame seats on the inner cone. Partial failure of combustion in the inner burning region, that is, at one flame seat or the other, could not be detected.

Operational Range and Augmented Thrust Performance

Data for both maximum fuel-flow operating limits and augmented thrust performance were simultaneously obtained while operating the engine on balanced cycle at given conditions of pressure altitude and flight Mach number. The minimum fuel-flow operating limits were then obtained by reducing the total afterburner fuel, while maintaining altitude and flight Mach number constant, until lean blow-out occurred. Data were recorded at the instant of blow-out. The method of blow-out detection used was the same as discussed in the previous paragraph. The thrust increases afforded by the afterburner were evaluated by comparison of the augmented performance data with normal or nonburning performance data.

The operational range of the afterburner and the augmented and normal performance of the engine were determined at an engine speed of 12,500 rpm and at the flight conditions listed in the following table:

Approximate pressure altitude (ft)	Augmented performance (balanced-cycle operation)	Flight Mach number Lean blow-out	Normal performance
5,000	Approximately 0	0.08	Approximately 0
20,000	0.60, 0.86, and 1.01	0.62	0.60, 0.86, and 1.00
30,000	0.61, 0.86, and 1.01	0.60, 0.85, and 0.99	---------------
35,000	0.84 and 0.99	0.98	1.00
40,000	----------------	1.12	---------------
50,000	----------------	----------------	0.85

The engine-inlet-air total temperatures were generally maintained within 10° F of the value corresponding to NACA standard air at the altitude and the flight Mach number of operation. Data calculations were based on a 100-percent ram recovery at the engine inlet.

The values of combustion efficiency presented are defined as the ratio of the actual temperature rise in the afterburner divided by the theoretical temperature rise for complete combustion of the fuel. The theoretical temperature rise was determined from the charts of reference 2 and the actual temperature rise was determined from the measured turbine-outlet temperature and the effective exhaust-nozzle-outlet temperature calculated according to the method discussed in the appendix.

RESULTS AND DISCUSSION

Altitude Limits

A preliminary investigation of the altitude limits of the afterburner indicated that balanced-cycle operation (limiting turbine-outlet temperature at rated engine speed) could not be obtained at any simulated flight conditions at an altitude of 20,000 feet or above with the afterburner exhaust-nozzle area of the original configuration ($18\frac{5}{8}$- in. equivalent diameter). With the modified-afterburner exhaust nozzle (18-in. diameter), balanced-cycle operation could be obtained at altitudes up to 36,000 feet. When this altitude range was approached, the turbine-outlet temperatures, however, tended to decrease with increasing altitude regardless of whether the afterburner fuel flow was maintained constant or increased. When this condition occurred, the afterburner fuel flow could usually be decreased several hundred pounds per hour, while altitude was maintained constant,with little or no effect on turbine-outlet temperature.

Although the altitude limit of satisfactory afterburner operation (balanced-cycle operation) was approximately an altitude of 36,000 feet, the burner would operate at higher altitudes. The altitude blow-out limits of the afterburner with the modified 18-inch-diameter exhaust nozzle over a range of flight Mach numbers at an engine speed of 12,500 rpm are shown in figure 4. Blow-out usually occurred during a small adjustment in one of the operating variables and fell within a band of unstable operation of about 6000-foot altitude in width. Temporary limitation of the laboratory service equipment at the time of this investigation made it impossible to obtain altitude blow-out points at flight Mach numbers over 0.80. The curve separating the regions of blow-out and stable operation indicated that the lowest altitude at which blow-out would be encountered varied from about 31,000 feet at a flight Mach number of 0.40 to about 45,500 feet at a flight Mach number of 1.00. The engine and afterburner operating conditions for the altitude blow-out points shown in figure 4 are given in table I.

As indicated in table I, the turbine-outlet temperatures at blow-out at some conditions were as much as 200° F below the temperature 1590° R required for balanced-cycle operation of the engine. Under these conditions, a long luminous flame was apparent, which indicated that a considerable portion of the fuel burned after leaving the engine exhaust nozzle. As the afterburner fuel flow was decreased, this tail of flame retracted within the burner shell.

The inlet-air temperatures prevailing during these experiments, which are included in table I, were from 10° to 30° F higher than NACA standard air with the greater deviations occurring in the low Mach number region of the data. Other investigations have indicated that turbine-outlet pressure is an important factor affecting blow-out. Because the turbine-outlet pressures would be decreased by the high inlet-air temperatures, the altitude blow-out regions indicated in figure 4 may be slightly lower than would occur at standard inlet-air temperatures.

Operational Range of Afterburner

Fuel flow. - The operational range of the afterburner fuel flow over a range of flight Mach numbers at altitudes of 20,000, 30,000, 35,000, and 40,000 feet, as well as at an altitude of 5000 feet and a Mach number of approximately 0, is presented in figure 5 for an engine speed of approximately 12,500 rpm. The altitudes given are approximate.

The upper limit of fuel flow represents the condition of maximum permissible turbine-outlet temperature (balanced-cycle operation) and is significant only for the nozzle size used in these investigations. The lower limit of fuel flow represents the condition of lean blow-out. Both the upper and lower limits of fuel flow increased with flight Mach number and decreased with increasing altitude. For all the conditions investigated, the spread between maximum and minimum fuel flow at a given altitude varied between 2500 and 3000 pounds per hour.

Fuel-air ratio. - By use of the data of figure 5 and the corresponding air-flow data, the curves of figure 6 were prepared to show the operational range of the afterburner fuel-air ratio. The afterburner fuel-air ratio is defined as the fuel flow to the afterburner divided by the amount of air flow associated with the unburned oxygen at the afterburner inlet.

The maximum fuel-air ratios (fuel-air ratio necessary to maintain balanced-cycle operation) increased with altitude and decreased with increasing flight Mach number. For an altitude of 20,000 feet, the upper limit of afterburner fuel-air ratio varied from about 0.041 to 0.036 as the flight Mach number was increased from 0.60 to 1.00. The highest fuel-air ratio encountered in this investigation was 0.051, which occurred at an altitude of 35,000 feet and a flight Mach number of 0.84. As the Mach number at this altitude was increased to 1.00, the fuel-air ratio decreased to 0.047. At an altitude of 5000 feet and a Mach number of 0, the maximum fuel-air ratio was about 0.031. An increase in the exhaust-nozzle area would have permitted operation at higher fuel-air ratios at low altitudes; as was previously noted, however, operation with a larger nozzle was unsatisfactory at the higher altitudes because of the inability to maintain limiting turbine-outlet temperature for balanced-cycle operation. In order to obtain both maximum thrust at take-off conditions and balanced-cycle engine operation at altitude conditions, a variable-area exhaust nozzle would therefore be required.

The minimum fuel-air ratios were nearly independent of both altitude and flight Mach number within the range of the investigation and had a value of approximately 0.012 (fig. 6). Minimum fuel-air-ratio data at an altitude of 20,000 feet are incomplete because of a structural failure of the afterburner, which will be subsequently discussed.

Performance with Afterburning

Augmented and normal net thrust. - The variation of augmented net thrust for balanced-cycle operation and normal net thrust over a range of simulated flight Mach numbers at altitudes of 20,000, 30,000, and 35,000 feet and at 5000 feet for a Mach number of 0 is presented in figure 7. Additional data for the augmented performance points are included in table II. For all altitudes, the augmented engine thrust rapidly increased with increasing flight Mach number; at an altitude of 20,000 feet, the augmented thrust increased from 2520 to 3350 pounds as Mach number was increased from 0.60 to 1.00.

Normal-engine-performance data were not available for an altitude of 30,000 feet and for a flight Mach number of 0.85 at 35,000 feet; in order to include an augmented-to-normal net-thrust comparison at these conditions in the discussion, the normal performance at 30,000 feet and flight Mach number of 0.85 at 35,000 feet was found by interpolation of plots of performance variables for altitudes of 20,000, 35,000, and 50,000 feet. These data were selected at an indicated engine speed of 12,500 rpm from curves of performance plotted against engine speed. Normal engine thrust (without the afterburner installed) increased slightly with flight Mach number within the range of the investigation and decreased as the altitude was increased.

Turbine-outlet temperature. - The turbine-outlet-temperature conditions that prevailed during both normal and augmented performance are presented in figure 8. Because of the operating characteristics of the engine with a fixed exhaust nozzle, the turbine-outlet temperatures during normal engine performance decreased from the limiting value at 5000 feet at a flight Mach number of 0 to a value 150° F below the limiting temperature for high altitudes at a Mach number of 1.00. During afterburner operation, however, the turbine-outlet temperature was nearly constant at the limiting value. Because of this temperature difference, a part of the thrust increase obtained during afterburning is a result of the increased turbine-outlet pressure associated with the increased turbine-outlet (and hence turbine-inlet) temperature.

Augmented thrust ratio. - The ratio of the augmented net thrust with afterburning to the normal-engine net thrust (hereinafter designated augmented-thrust ratio) over a range of flight Mach numbers at altitudes of 20,000, 30,000, and 35,000 feet and for a flight Mach number of 0 at an altitude of 5000 feet is presented in figure 9. Because the engine speeds for the normal and augmented performance differed slightly, the normal-engine thrust used to calculate the augmented thrust ratio was determined from plots of normal performance against engine speed for the engine speed that prevailed during the corresponding augmented performance point. Thus, the augmented-thrust ratios of figure 9 do not correspond exactly to the thrust data presented in figure 7.

The augmented-thrust ratio rapidly increased with increasing flight Mach number and decreased as the altitude was raised over the range of flight conditions investigated. Over a range of flight Mach numbers from 0.60 to 1.00 at an altitude of 20,000 feet, for example, the augmented-thrust ratio increased from 1.80 to 2.33 (fig. 9); increasing the altitude to 35,000 feet at a flight Mach number of 1.00, however, decreased the augmented-thrust ratio to about 2.03. At an altitude of 5000 feet and a flight Mach number of 0, a thrust ratio of 1.27 was obtained. This relatively low value of augmented-thrust ratio at this flight condition is attributed to the limitation imposed on the maximum operable afterburner fuel-air ratio (limiting turbine-outlet temperature) by the exhaust-nozzle size used in this investigation.

Total specific fuel consumption. - The total specific fuel consumption based on net thrust (lb/(hr)(lb net thrust)) over the range of flight Mach numbers and altitudes of this investigation is presented in figure 10 for normal engine performance and for augmented performance in the balanced-cycle condition. The specific fuel consumption for afterburning operation decreased with increasing flight Mach number and increased rapidly with altitude, for example, increasing the Mach number from 0.60 to 1.00 at an altitude of 20,000 feet decreased the specific fuel consumption from 2.43 to 2.24 pounds per hour per pound net thrust (the minimum specific fuel consumption obtained); increasing the altitude to 35,000 feet at Mach number 1.00, however, increased the specific fuel consumption to 2.59. The values of specific fuel consumption at an altitude of 20,000 feet, which are for balanced-cycle operation, are from 34 to 72 percent higher than the normal-engine specific fuel consumption in the range of flight Mach numbers shown.

At an altitude of 5000 feet and a flight Mach number of 0, the specific fuel consumption was about 2.10 pounds per hour per pound net thrust, which is about twice the normal specific fuel consumption.

Afterburner combustion efficiency. - The afterburner combustion efficiencies over a range of flight Mach numbers at altitudes of 20,000, 30,000, and 35,000 feet and for a flight Mach number of 0 at an altitude of 5000 feet are presented in figure 11. Combustion efficiency increased with flight Mach number and decreased rapidly with altitude, for example, increasing the Mach number from 0.60 to 1.00 at an altitude of 20,000 feet, increased the efficiency from 75 to 81 percent (the maximum efficiency obtained); increasing the altitude to 35,000 feet at Mach number of 1.00, however, reduced the efficiency to 55 percent. At an altitude of 5000 feet and a flight Mach number of 0, a combustion efficiency of 75 percent was obtained.

The previous discussion of the difficulty of maintaining balanced-cycle operation above an altitude of about 36,000 feet and the decrease in combustion efficiency at 35,000 feet shown in figure 11 indicate that the afterburner operation at altitudes above 35,000 feet was unsatisfactory. Because of the assumptions made in the calculation of combustion efficiency, the values may be from 3.5 to 7.5 percent too high, as discussed in the appendix.

Operational Characteristics

Afterburner fuel regulation. - The double-fuel-manifold configuration used in this afterburner made it necessary to establish a satisfactory ratio of the fuel flow between the two manifolds over the range of simulated flight conditions investigated. Investigations conducted by the manufacturer showed that a fuel ratio between the outer and inner manifolds of 6:5 provided the most satisfactory operation at sea-level zero-ram conditions. Accordingly, the 6:5 ratio was selected to be used for the investigation at an altitude of 5000 feet. A partial failure of the fuel throttle to the inner manifold during the performance run caused the ratio of outer- to inner-manifold fuel flow to be 7:10 as indicated in table II. Brief preliminary investigations showed that balanced-cycle operation could not be reached with the 6:5 ratio at higher altitudes. For all investigations at altitudes above 5000 feet, therefore, a ratio of outer- to inner-manifold fuel flow that provided uniform combustion with flame filling the exhaust nozzle was established by visual observation of the afterburner flame. The fuel ratios used varied from approximately 1 at 20,000-foot altitude and Mach number 1.01 to 3.5 at 30,000-foot altitude and a Mach number of 0.61 (table II).

Ratios of outer- to inner-manifold fuel flow less than those selected resulted in rough burning at the inner cone and a considerable increase in afterburner vibration. Ratios greater than those selected resulted in a thinning of the flame in the central portion of the burner and, in extreme cases, intermittent burning at the inner cone. During operation with a properly proportioned fuel flow to the two manifolds, the afterburner operated very smoothly with a bluish transparent flame of uniform intensity. At altitudes above 36,000 feet, the exhaust flame was considerably elongated during operation with high fuel flows associated with attempts to establish balanced-cycle operation. Under these conditions traces of yellow were observed in the otherwise blue flame.

Slight flickering of the flame was occasionally noticed during operation near the altitude blow-out limit. Lean blow-out points at stabilized altitude conditions were characterized by a progressive failure of combustion in the outer burning region, starting at the top of the burner.

Ignition characteristics. - Throughout the investigation, very little difficulty was experienced with afterburner ignition. Although the complete determination of ignition characteristics was beyond the scope of the present investigation, an occasional attempt was made to ignite the afterburner at conditions other than those previously mentioned (altitude, 10,000 ft; ram pressure ratio, 1.07; and engine speed, 10,500 rpm). By use of the same afterburner exhaust-nozzle area that was used throughout the investigation (18 in. equivalent diameter), successful starts were made over a range of flight Mach numbers from 0.32 to 0.60 at altitudes from 10,000 to 30,000 feet and engine speeds from 10,000 to 12,500 rpm. No attempts were made to ignite the afterburner at other conditions. Almost all instances of unsuccessful ignition that were encountered could be attributed to either starting technique or malfunctioning of some component of the ignition system.

In order to minimize pressure and air-flow fluctuations in the service equipment, the afterburner manifolds were separately ignited. The fuel flow during single manifold ignition was usually below 2000 pounds per hour for the conditions investigated.

Afterburner-shell temperatures. - The variation of afterburner-shell (inner-skin) temperatures over a range of flight Mach numbers at altitudes of 20,000, 30,000, and 35,000 feet is shown plotted against the distance from the shell and inner-cone flame seats in figure 12. For these runs, the pressure at the cooling-jacket inlet was from 1 to 4 inches of water higher than the ambient altitude pressure; the air available for cooling, which entered the test section through a valve in the bottom of the altitude chamber, reached a temperature of about 660° R before entering the cooling jacket.

The maximum shell temperatures recorded during the investigation were from 2085° to 2110° R and were obtained just downstream of the flame seats at the bottom of the burner shell. A temperature drop of about 200° F occurred along the length of the burner shell for about two-thirds of the conditions; at altitudes of 30,000 and 35,000 feet, however, an over-all temperature increase of as much as 75° F was measured on the bottom surface. The shell temperatures were as much as 550° F lower at the top of the burner than at the bottom. The temperatures generally decreased with increasing altitude and increased with increasing flight Mach number at the higher altitudes. At an altitude of 20,000 feet, however, increasing flight Mach number had little effect on shell temperatures.

Structural failure. - Three types of structural failure of the afterburner occurred during the investigation. The first of these was a failure of the flow divider that separates the inner and outer burning regions. Considerable distortion and the eventual failure of the flow divider with the formation of several circumferential cracks and tears occurred after 1 hour of engine operation, during which time the afterburner had not been operated. A replacement part of the same material and design as the original flow divider was made and installed at this laboratory. No further difficulty of this type was experienced throughout the investigation.

The second type of failure occurred after $11\frac{1}{2}$ hours of engine operation including $1\frac{1}{2}$ hours of afterburner operation. A hole was burned in the bellows slip joint near the bottom of the burner shell and appeared to be caused by a collection of liquid fuel in the bellows during engine starts and afterburner false starts. In order to insure against repeated failure of the bellows, a ring was welded into the burner to close the space between the outer shell and the expansion joint. When this ring, which rendered the expansion joint inoperative, was installed, no further difficulty was encountered.

A third type of failure occurred after $18\frac{1}{2}$ hours of engine operating time, of which $3\frac{1}{4}$ hours were afterburner operation. Failure of the burner shell, the radiation shield, and the cooling jacket occurred at the location of an instrumentation patch plate. This patch plate probably not only impaired shell cooling at this point but also provided a slight obstruction to the gas flow that acted as a flame seat or a point of localized heating. Although the burned-out areas were repaired, failure occurred again after a total time of about 22 hours of engine operation and 4 hours of afterburner operation. The location and conditions of both of these failures are illustrated in figures 13 and 14. The first failure was located near the center of the patch seen in figure 14; the second failure occurred along the lower weld of the first repair patch. Both burn-out failures occurred during high Mach number afterburning runs at an altitude of 20,000 feet; maximum afterburner fuel flows for the entire investigation prevailed for these runs. These failures illustrate the importance of an unobstructed cooling-air flow over the outside of the burner shell and the necessity of maintaining a smooth inner surface of the burner shell for satisfactory afterburner operating life.

SUMMARY OF RESULTS

An altitude-test-chamber investigation of the operational and performance characteristics of the McDonnell afterburner with an

18-inch-diameter exhaust nozzle on a J34 engine at rated engine speed gave the following results.

1. The altitude limit, as determined by combustion blow-out, occurred as a band of unstable operation of about 6000-foot altitude in width with minimum altitude limits from 31,000 feet at a flight Mach number of 0.40 to about 45,500 feet at a flight Mach number of 1.00. Considerable difficulty was experienced in attempting to establish or maintain balanced-cycle engine operation at altitudes above 36,000 feet.

2. The fuel-air ratio for balanced-cycle operation and lean blow-out of the afterburner, the augmented thrust ratio, the total specific fuel consumption, and the afterburner combustion efficiency for balanced-cycle operation are summarized in the following table:

Altitude (ft)	Flight Mach number	Afterburner fuel-air ratio		Augmented-thrust ratio (a)	Total specific fuel consumption (lb/(hr)(lb net thrust))	Afterburner combustion efficiency (percent)
		Balanced-cycle operation	Lean blow-out			
20,000	0.60	0.041	0.011	1.80	2.43	75
20,000	1.00	.036	.012	2.33	2.24	81
35,000	1.00	.047	.012	2.03	2.59	55

[a]Augmented-thrust ratios were obtained from augmented thrust at limiting turbine-outlet temperature and normal thrust at turbine-outlet temperature as much as 150° F lower than limiting value.

3. Satisfactory afterburner ignition was obtained over a range of flight Mach numbers from 0.32 to 0.60, at altitudes from 10,000 to 30,000 feet, and engine speeds from 10,000 to 12,500 rpm (rated speed).

4. Maximum afterburner-shell temperatures were about 2110° R; temperatures measured at the top of the burner shell were as much as 550° F lower than those measured at the bottom. For these conditions, the cooling air at the inlet of the cooling jacket was about 660° R and at a pressure of 1 to 4 inches of water above ambient altitude pressure. Localized overheating resulted in

structural failure of the afterburner shell and a collection of liquid fuel in the expansion bellows during engine and afterburner starts caused a failure of the afterburner expansion joint.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
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APPENDIX - EQUATIONS USED IN CALCULATIONS

The following symbols are used in the equations in this appendix:

- A exhaust-nozzle-throat area, sq ft
- F_j jet thrust, lb
- g gravitational constant, ft/sec^2
- p exhaust-nozzle-throat static pressure, lb/sq ft
- p_0 ambient static pressure, lb/sq ft
- R gas constant, ft-lb/(lb)($^{\circ}$R)
- T effective exhaust-gas total temperature, $^{\circ}$R
- t exhaust-gas static temperature, $^{\circ}$R
- V_j jet velocity (sonic), ft/sec
- W gas flow, lb/sec
- γ ratio of specific heats of exhaust gas
- ρ density of gas at exhaust-nozzle throat, lb/cu ft

The equation for the effective exhaust-nozzle total temperature T was derived from consideration of the equations presented in the following paragraphs.

For sonic velocity at the nozzle outlet, the jet thrust may be expressed as

$$F_j = \frac{W}{g} V_j + (p - p_0) A \tag{1}$$

The expression for sonic jet velocity is

$$V_j = \sqrt{\frac{2\gamma}{\gamma+1} g\ RT} \tag{2}$$

From consideration of continuity the gas flow is

$$W = \rho A V_j = \frac{pAV_j}{Rt} \tag{3}$$

The equation for the ratio of total to static temperature, for a Mach number of 1.00 is written

$$\frac{T}{t} = 1 + \frac{\gamma-1}{2} = \frac{\gamma+1}{2} \qquad (4)$$

These four equations may be combined and manipulated to eliminate the unknown variables p, t, and V_j, which lead to the following equation for the effective exhaust-nozzle total temperature

$$T = \frac{\gamma g \left(F_j + p_0 A\right)^2}{2W^2 R\ (\gamma+1)}$$

In the previous equations, the area coefficient has been assumed equal to unity because of the uncertainty involved in the selection of an applicable coefficient. A survey of the available information indicated that the area coefficients for conditions similar to those of this investigation may be as low as 0.97. As compared with temperatures calculated using an area coefficient of 0.97, exhaust-nozzle total temperatures calculated with an area coefficient of 1.0 were 1.5 to 2.0 percent too high at altitudes of 20,000 feet and above, which would result in combustion efficiencies from 3.5 to 4.5 percent too high. A similar calculation at 5000-foot altitude indicated a temperature 3 percent too high and an efficiency 7.5 percent too high.

REFERENCES

1. Dowman, Harry W., and Reller, John O.: Altitude-Test-Chamber Investigation of a Solar Afterburner on the 24C Engine. I - Operational Characteristics and Altitude Limits. NACA RM No. SE8G02, Bur. Aero., 1948.

2. Turner, L. Richard, and Lord, Albert M.: Thermodynamic Charts for the Computation of Combustion and Mixture Temperatures at Constant Pressure. NACA TN No. 1086, 1946.

TABLE I - OPERATING CONDITIONS AT ALTITUDE BLOW-OUT

[Engine speed, 12,500 rpm]

Flight Mach number	Altitude (ft)	Inlet-air total temperature (°R)	Turbine-outlet temperature (°R)	Afterburner fuel flow (lb/hr)		
				Outer manifold	Inner manifold	Total
0.54	34,400	448	1465	--------	--------	2540
.58	36,800	449	1523	2610	590	3200
.60	39,200	453	1465	2050	520	2570
.64	41,600	448	1487	1470	1260	2730
.66	36,800	445	1383	--------	--------	2540
.77	43,400	456	1442	1440	770	2210
.80	38,500	454	1598	--------	--------	1690

TABLE II - PERFORMANCE WITH AFTERBURNING

Altitude (ft)	Flight Mach number	Engine speed (rpm)	Jet thrust (lb)	Net thrust (lb)	Engine fuel flow (lb/sec)	Afterburner fuel flow (lb/sec)			Air flow (lb/sec)	Turbine-outlet temperature (°R)	Engine-inlet temperature (°R)	Engine-inlet total pressure (in. Hg abs.)
						Total	Inner manifold	Outer manifold				
5,100	0	12,481	3253	3253	0.7050	1.201	0.708	0.493	49.42	1563	503	24.70
20,000	.60	12,470	3256	2548	.5522	1.149	.438	.711	36.18	1584	470	17.67
20,100	.86	12,466	4038	2881	.6072	1.266	.378	.889	42.34	1575	507	22.14
20,100	1.01	12,434	5106	3491	.7117	1.436	.703	.733	50.02	1563	522	26.18
30,000	.61	12,499	2134	1676	.3786	.927	.207	.722	24.54	1588	455	11.32
29,900	.86	12,405	3024	2238	.4700	1.018	.238	.781	29.76	1585	464	14.39
30,000	1.01	12,500	3571	2530	.5222	1.188	.327	.861	33.53	1608	485	16.81
34,700	.84	12,528	2309	1694	.3708	.953	.359	.594	24.15	1607	455	11.22
33,500	.99	12,446	3050	2147	.4439	1.109	.304	.806	29.96	1600	461	14.01

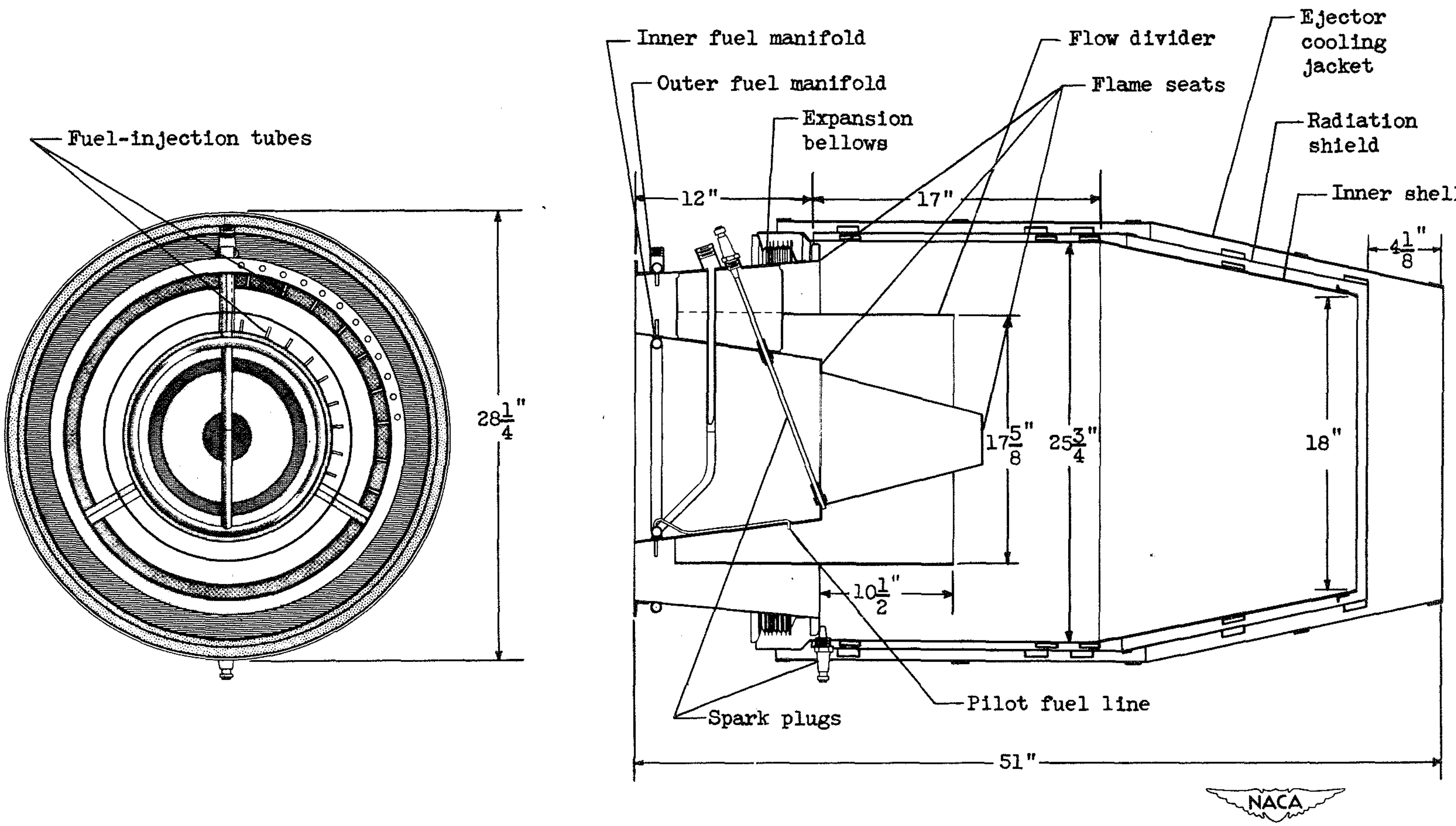

Figure 1. - Diagrammatic sketch of McDonnell afterburner.

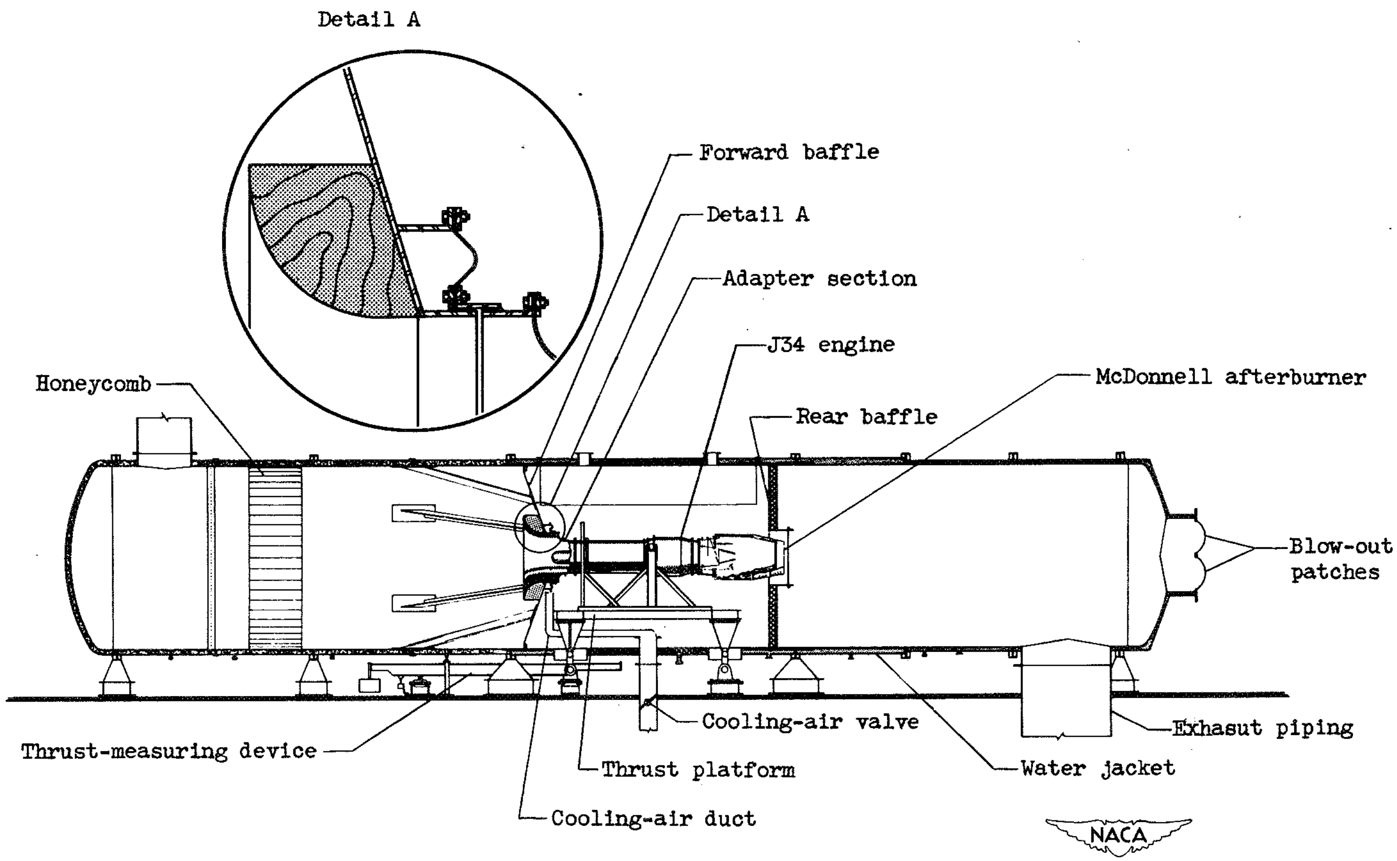

Figure 2. - Diagrammatic sketch of altitude test chamber with J34 engine and McDonnell afterburner installed.

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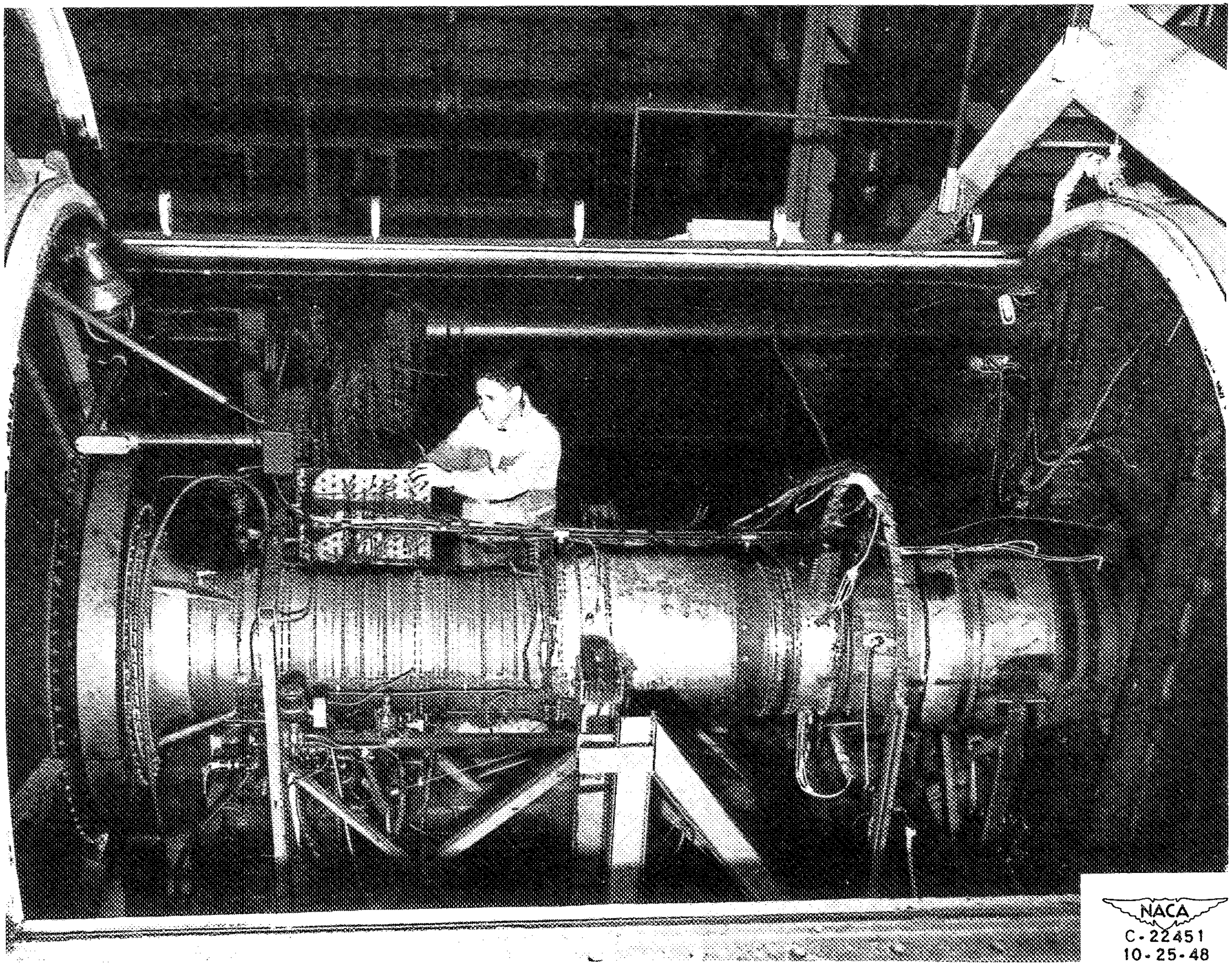

Figure 3. - Side view of J34 engine and McDonnell afterburner in altitude test chamber.

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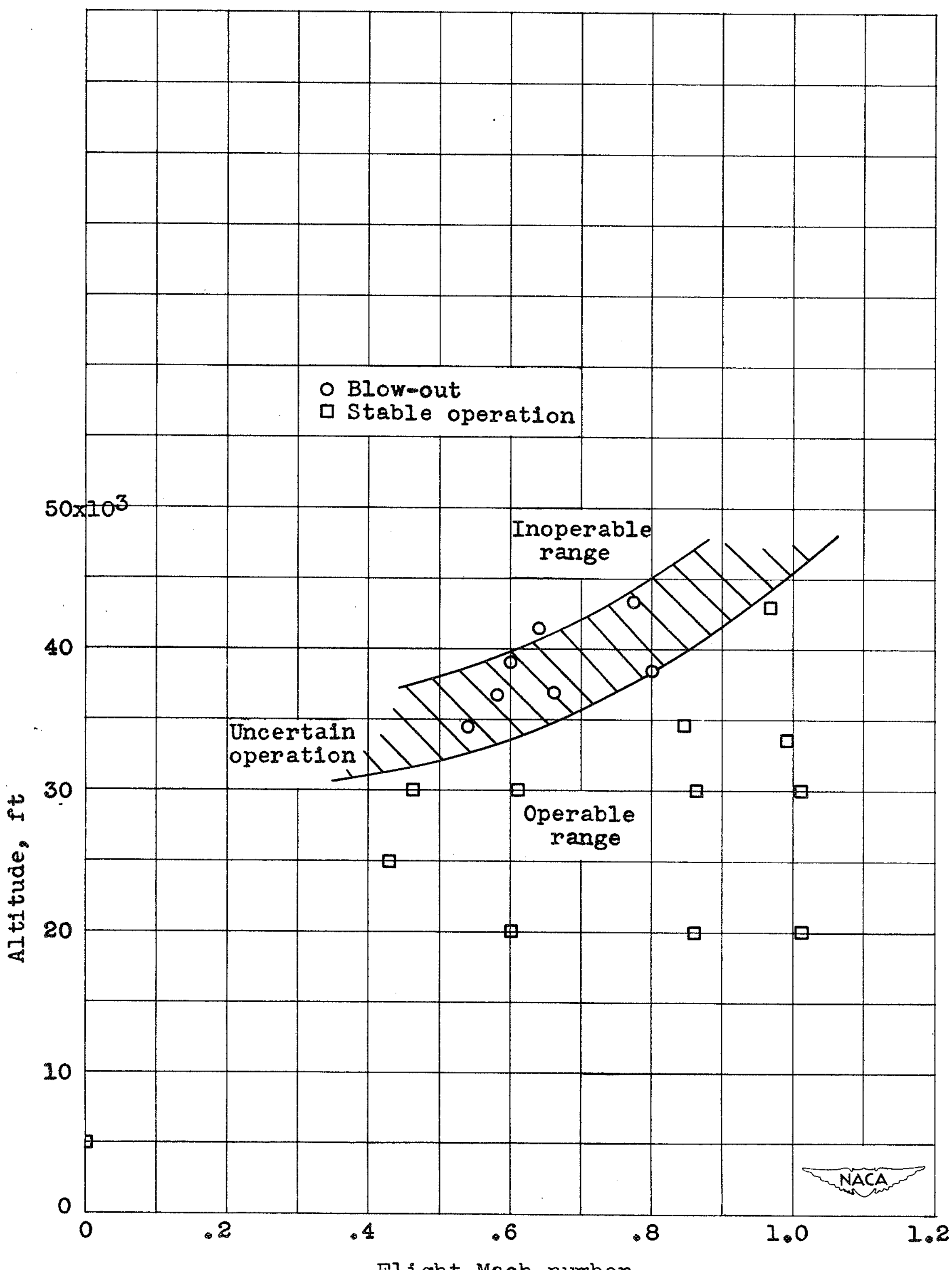


Figure 4. - Altitude blow-out limits of afterburner at engine speed of approximately 12,500 rpm.

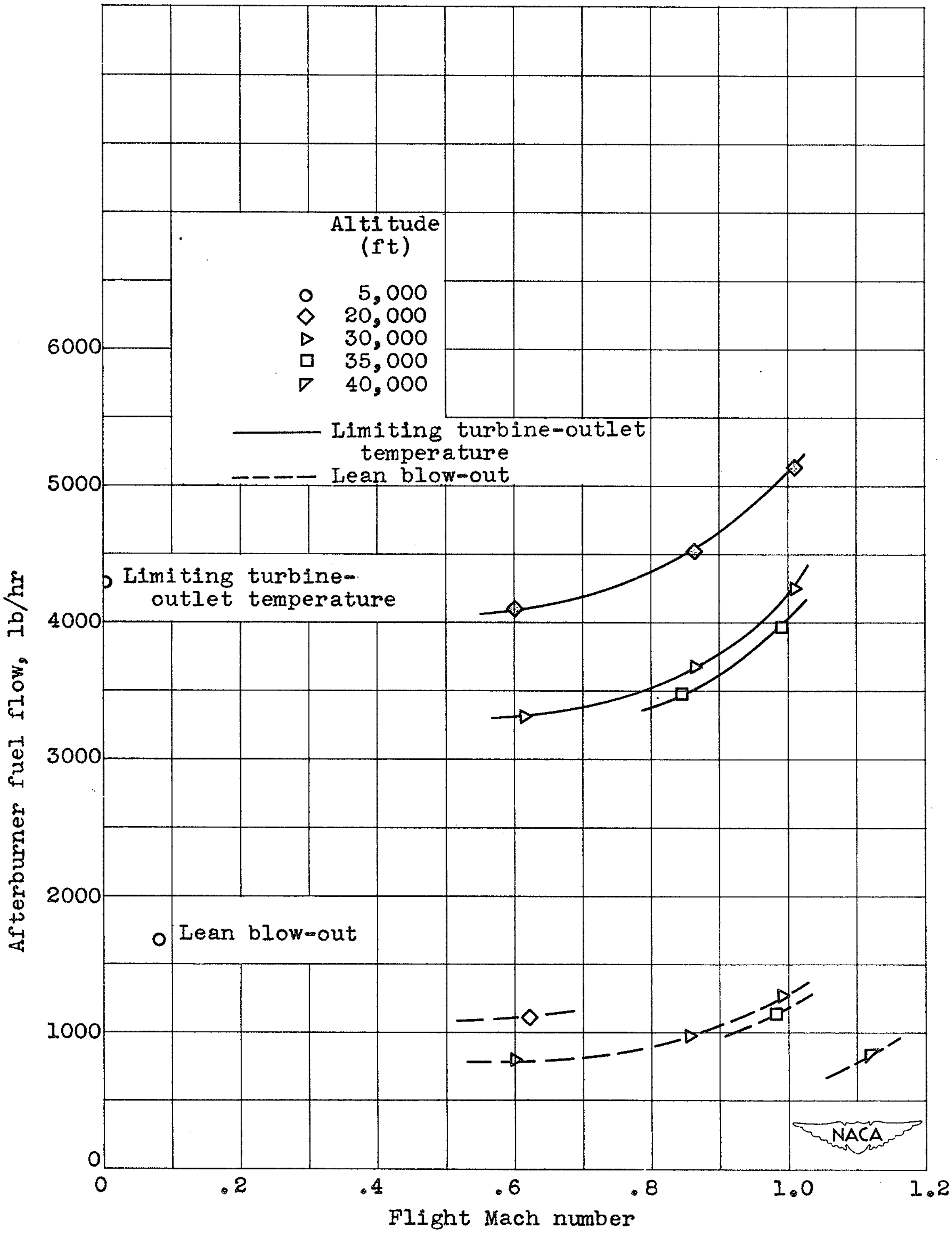


Figure 5. - Fuel-flow operational range of afterburner at engine speed of approximately 12,500 rpm.

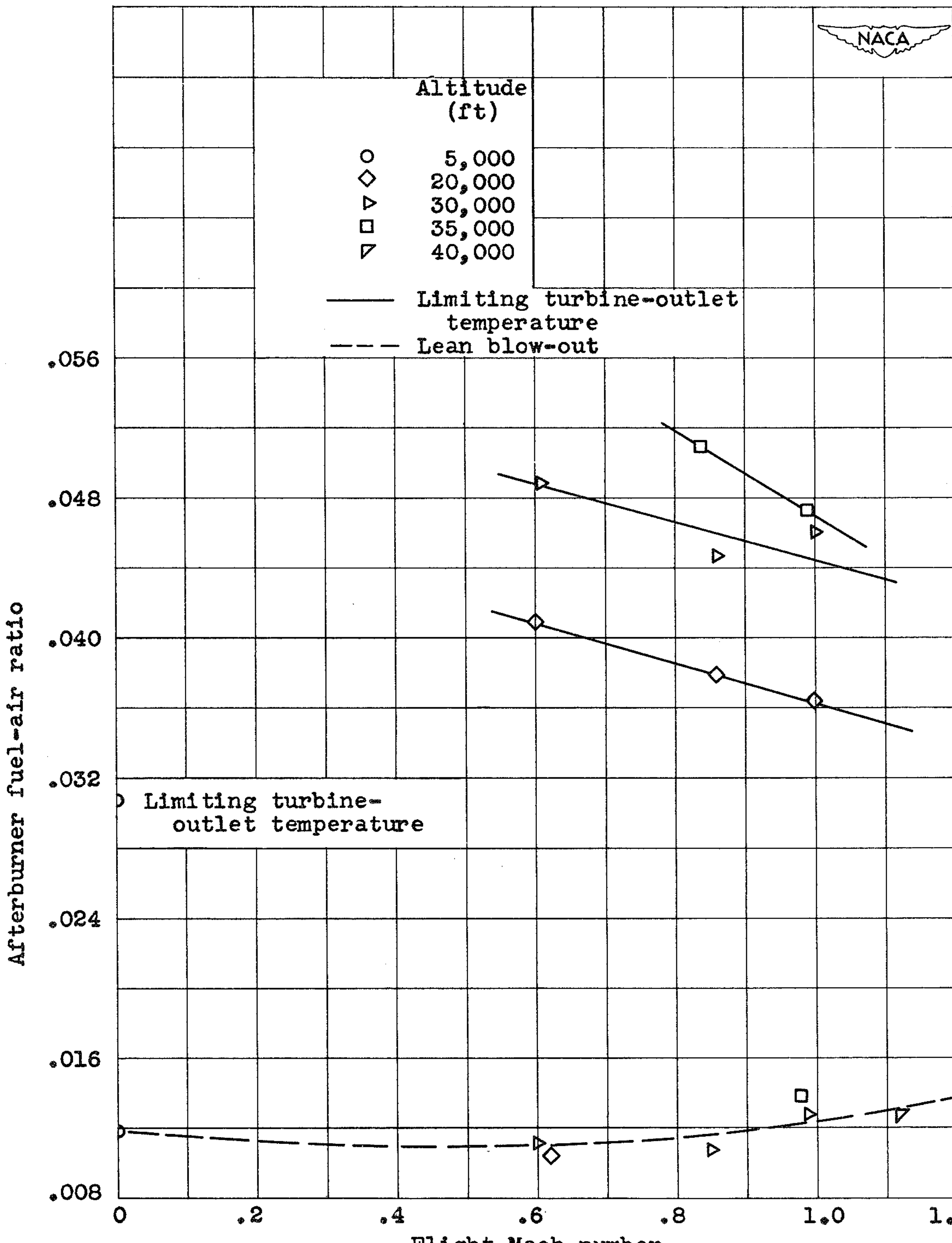


Figure 6. - Fuel-air ratio operational range of afterburner at engine speed of approximately 12,500 rpm.

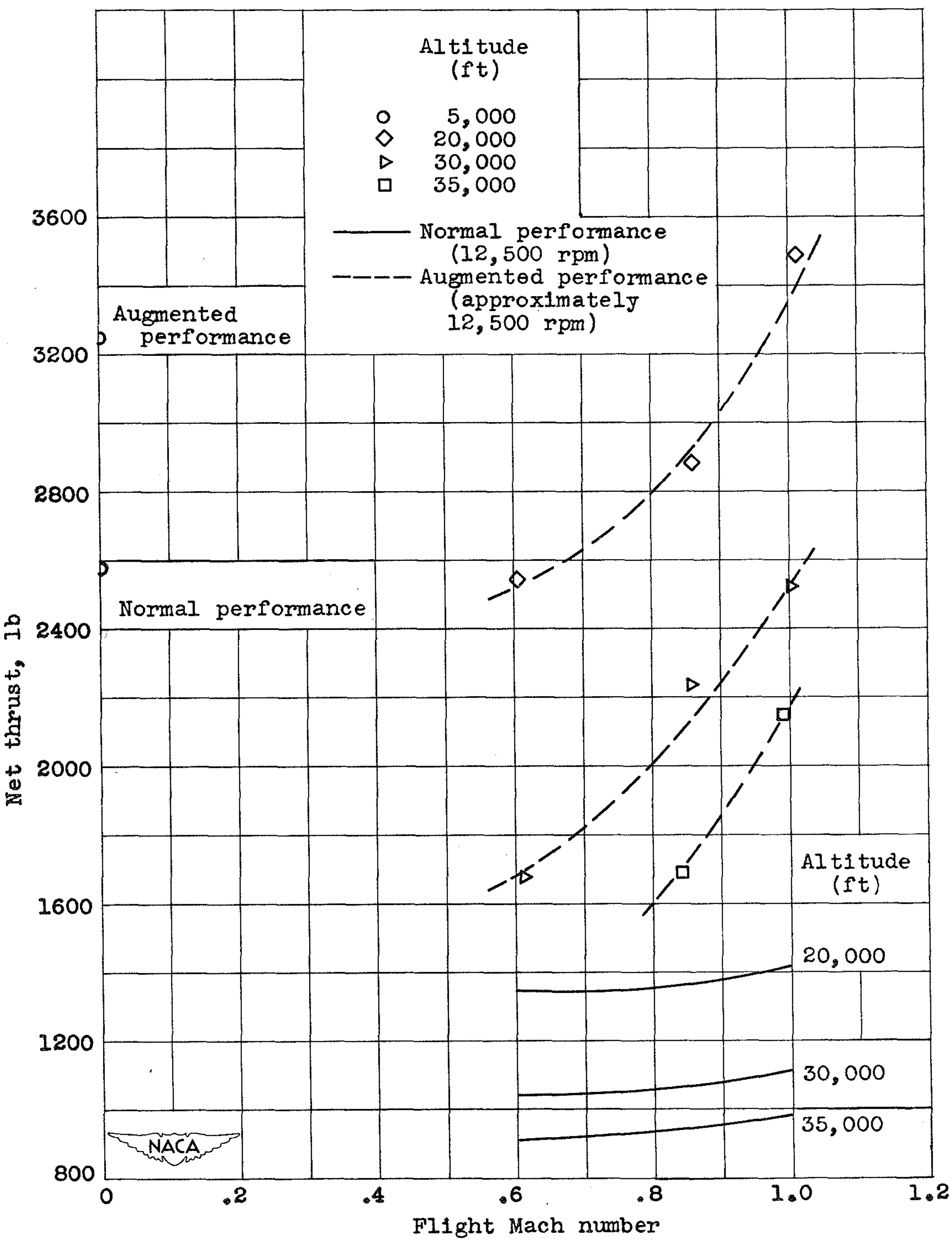


Figure 7. - Augmented net thrust with afterburner and normal net thrust.

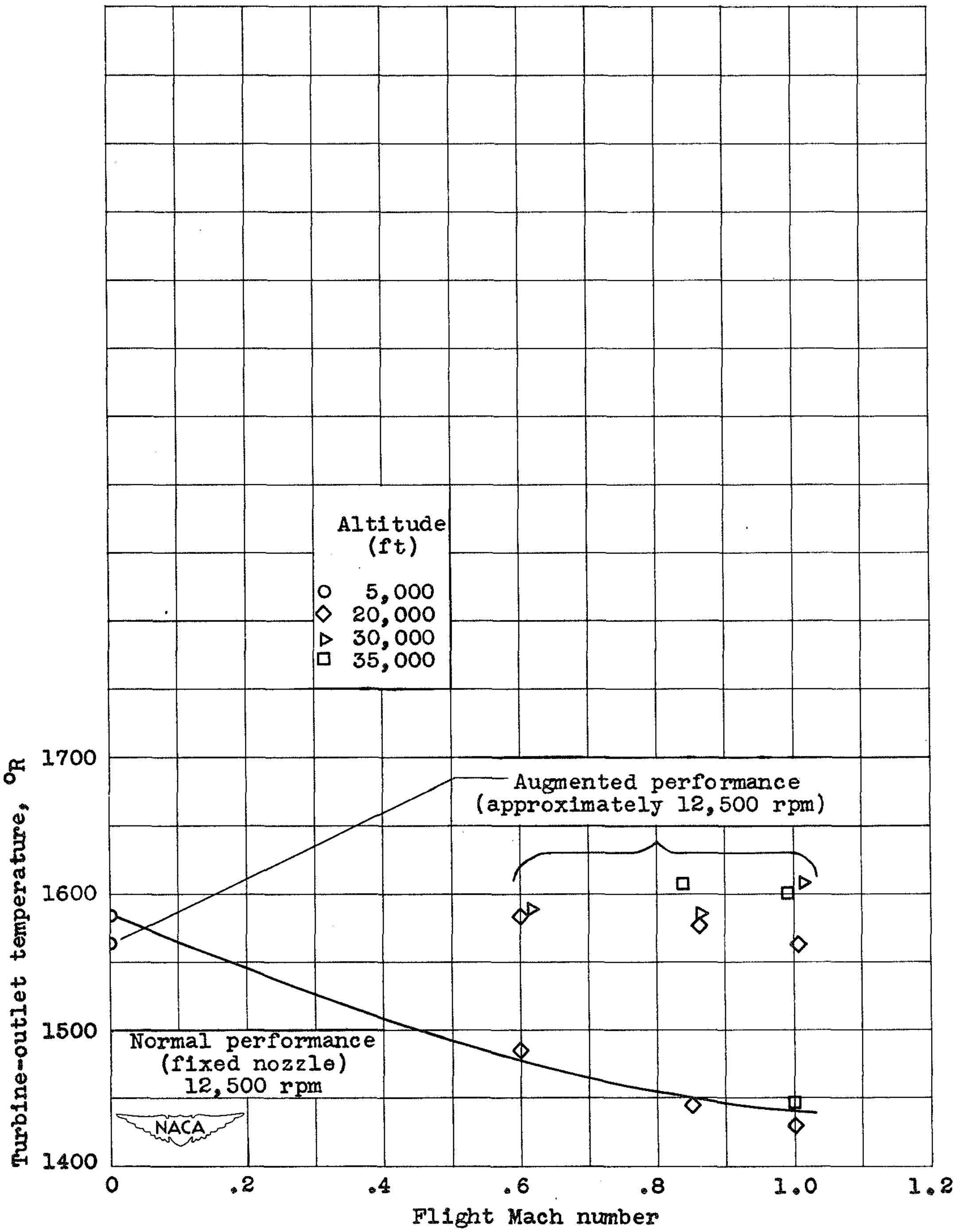


Figure 8. - Turbine-outlet temperature for augmented and normal operation.

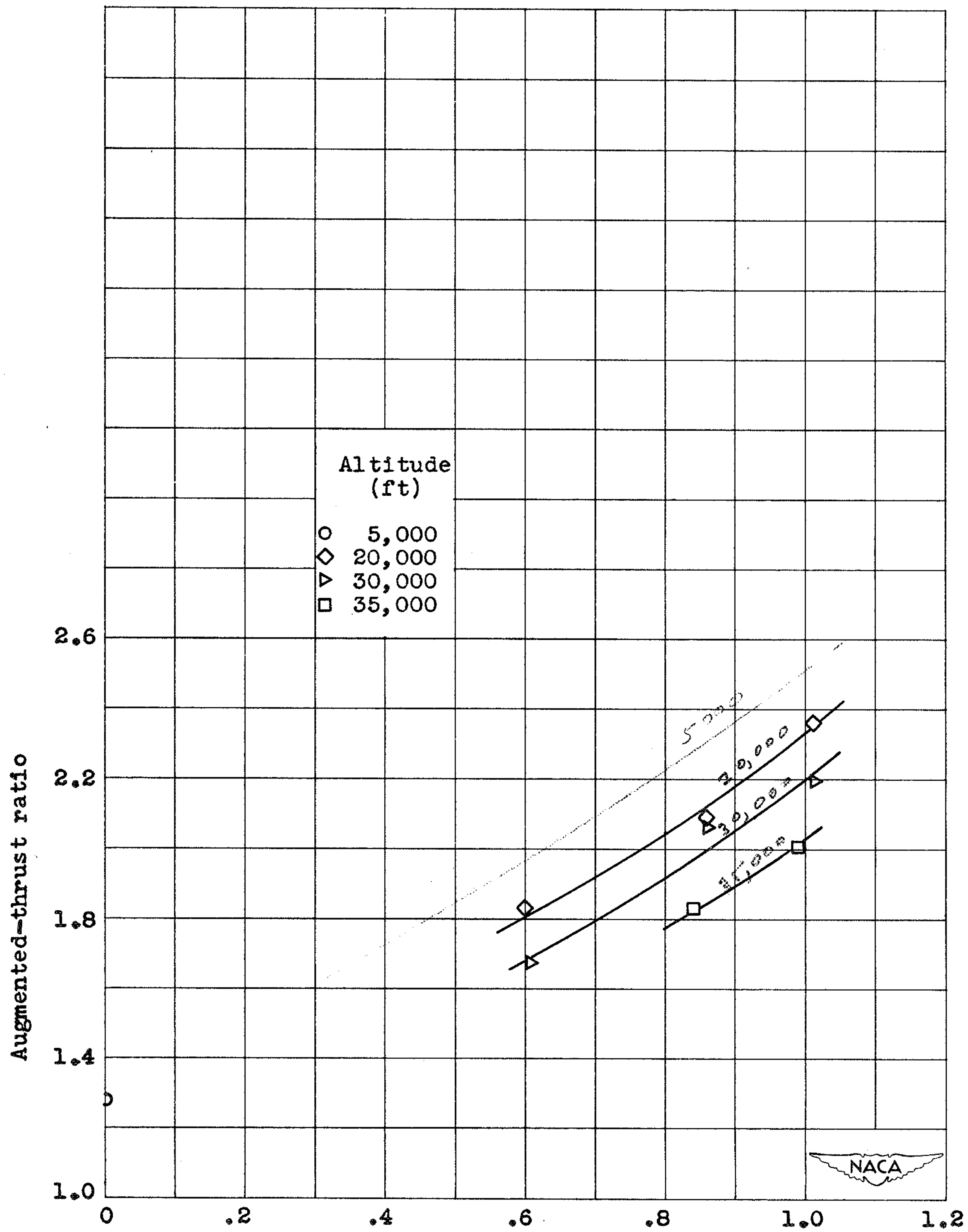


Figure 9. - Augmented-thrust ratio at engine speed of approximately 12,500 rpm.

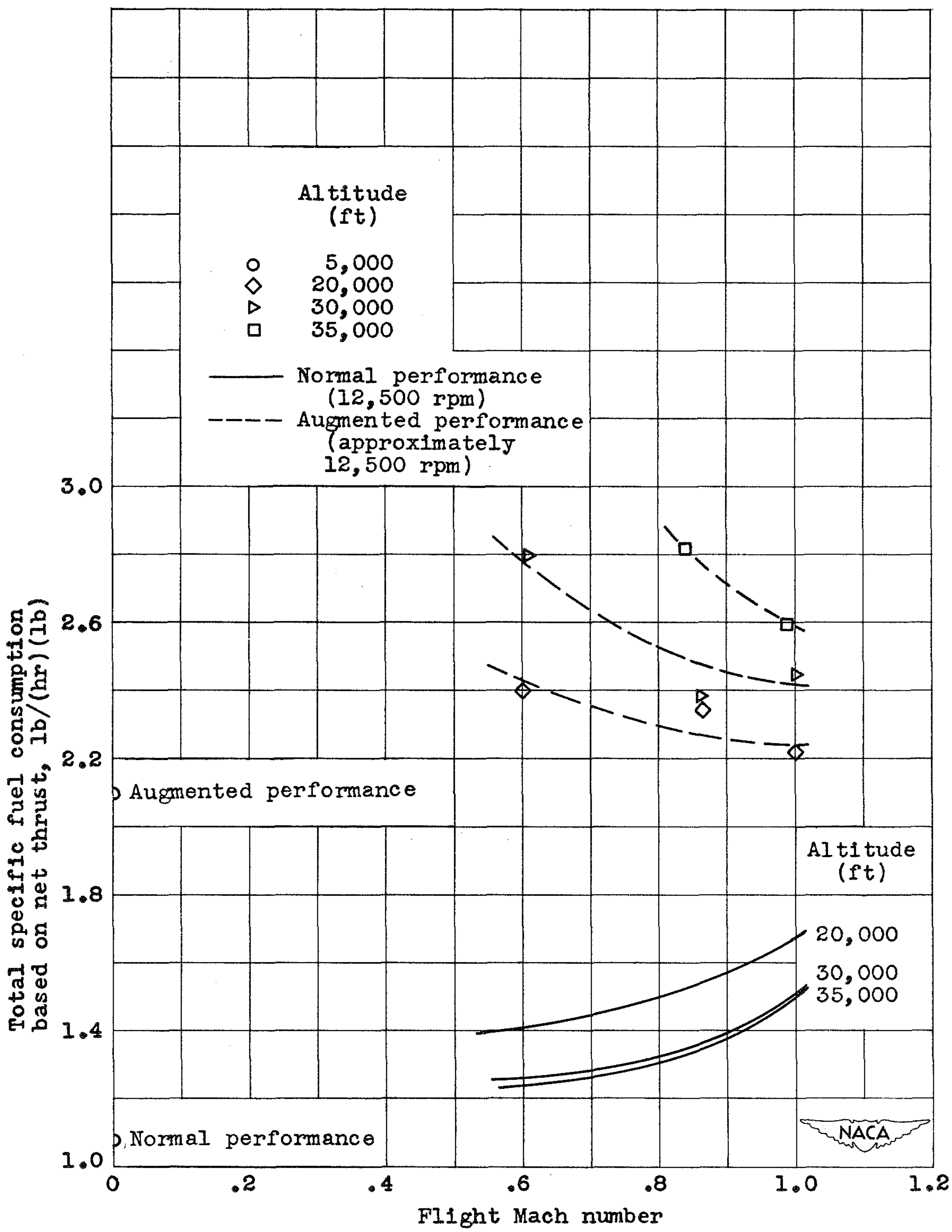


Figure 10. - Total specific fuel consumption for augmented and normal operation.

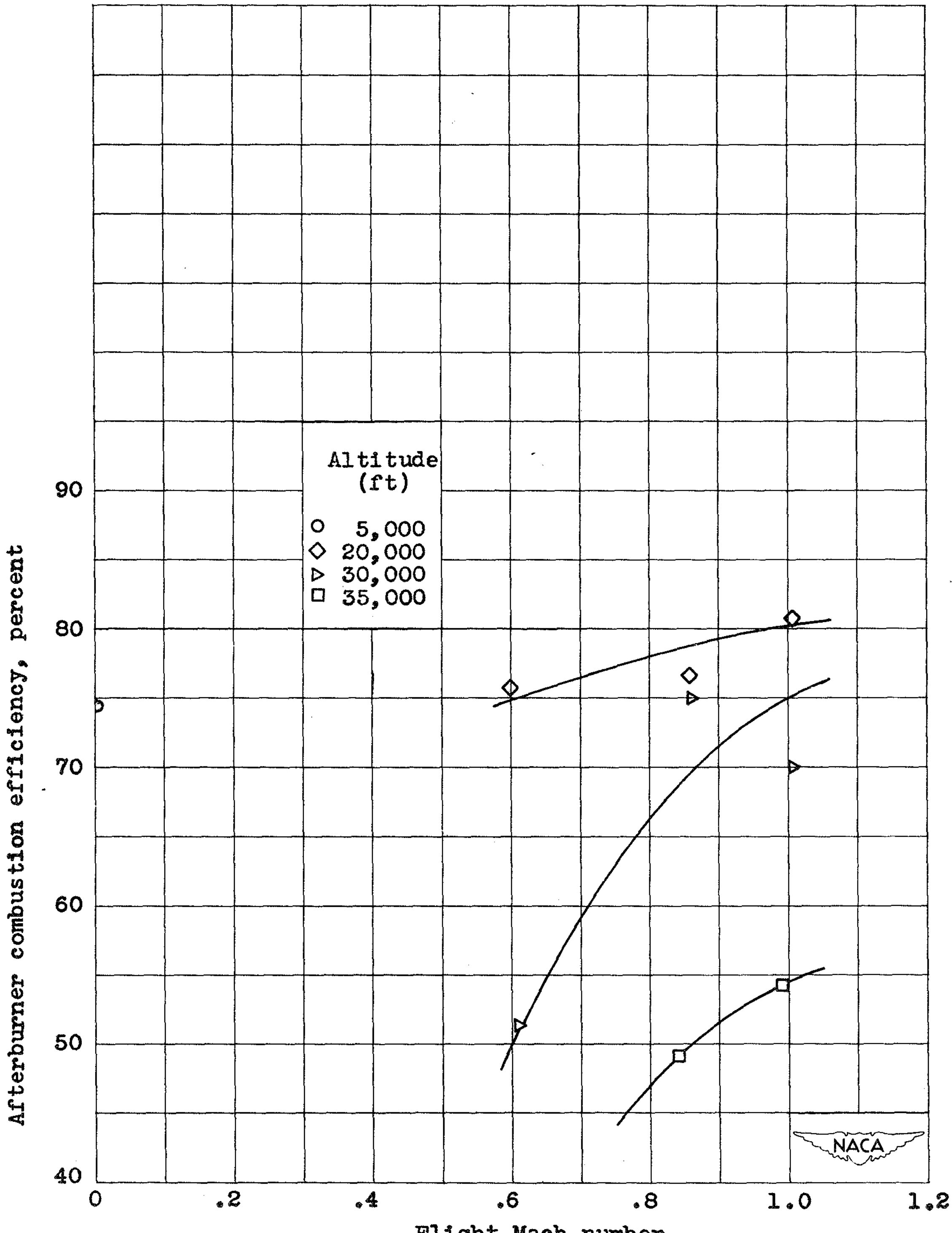
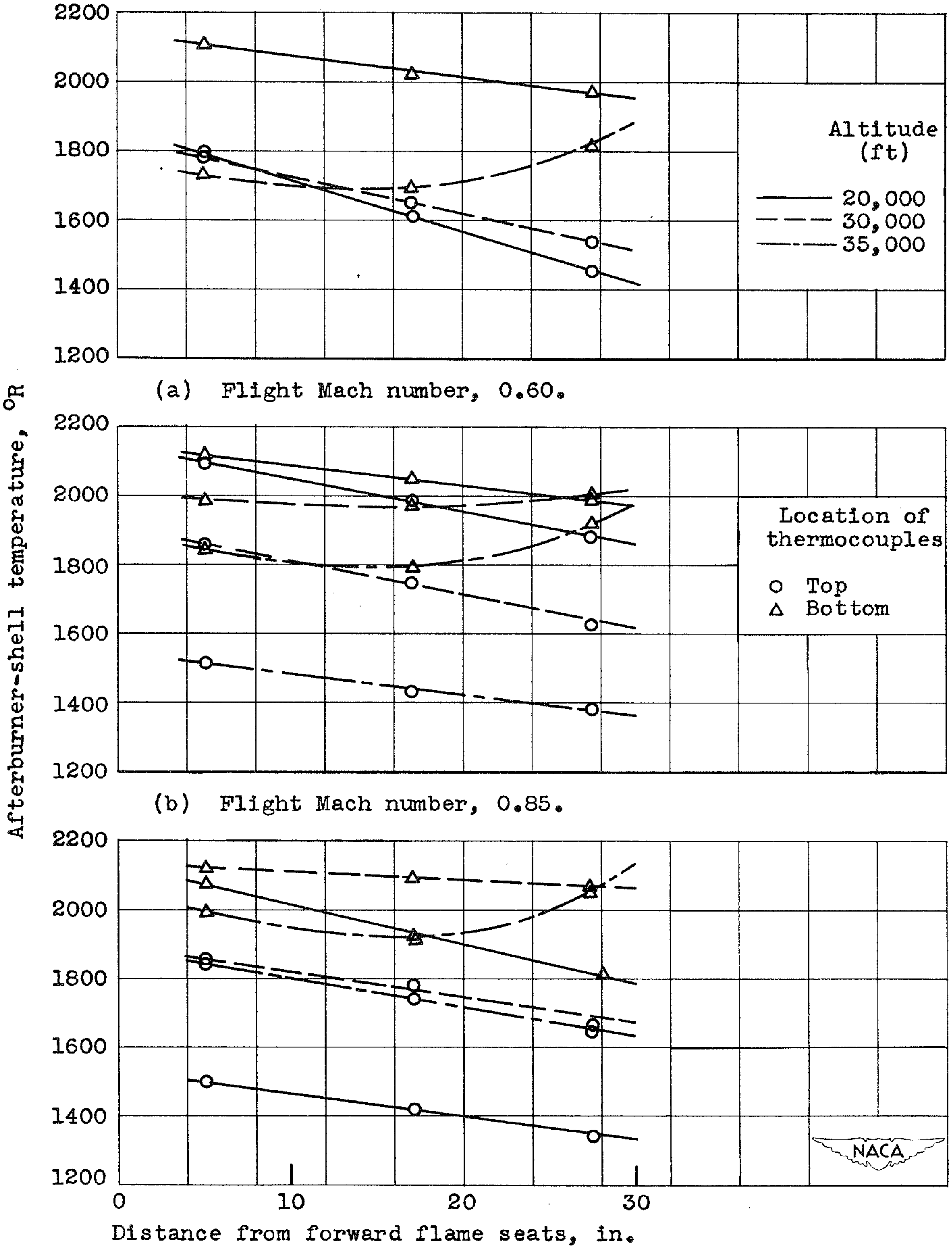


Figure 11. - Combustion efficiency of afterburner at an engine speed of approximately 12,500 rpm.



(c) Flight Mach number, 1.00.

Figure 12. - Afterburner-shell temperatures at engine speed of approximately 12,500 rpm.

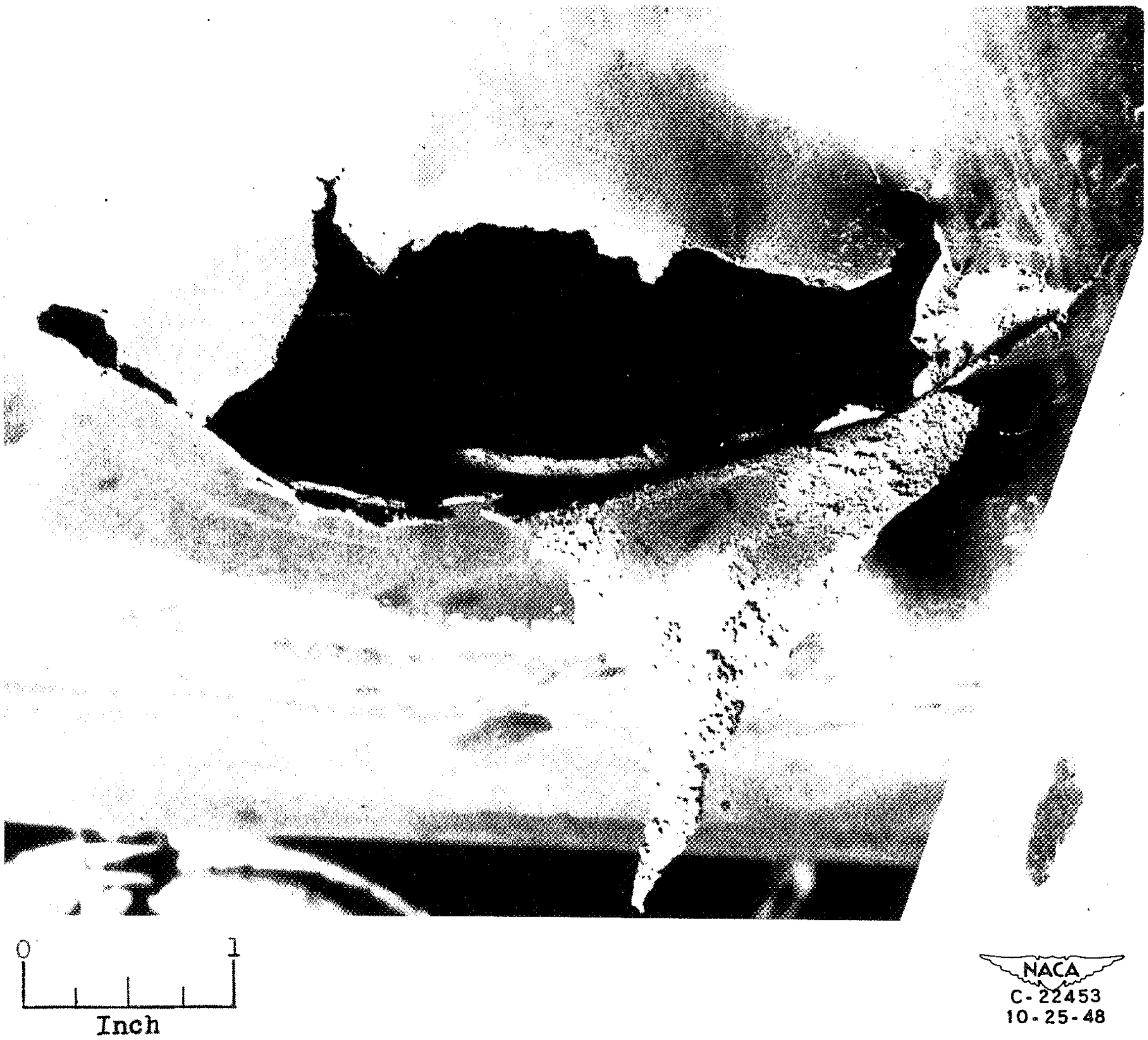


Figure 13. - External view of failure of McDonnell afterburner on J34 engine.

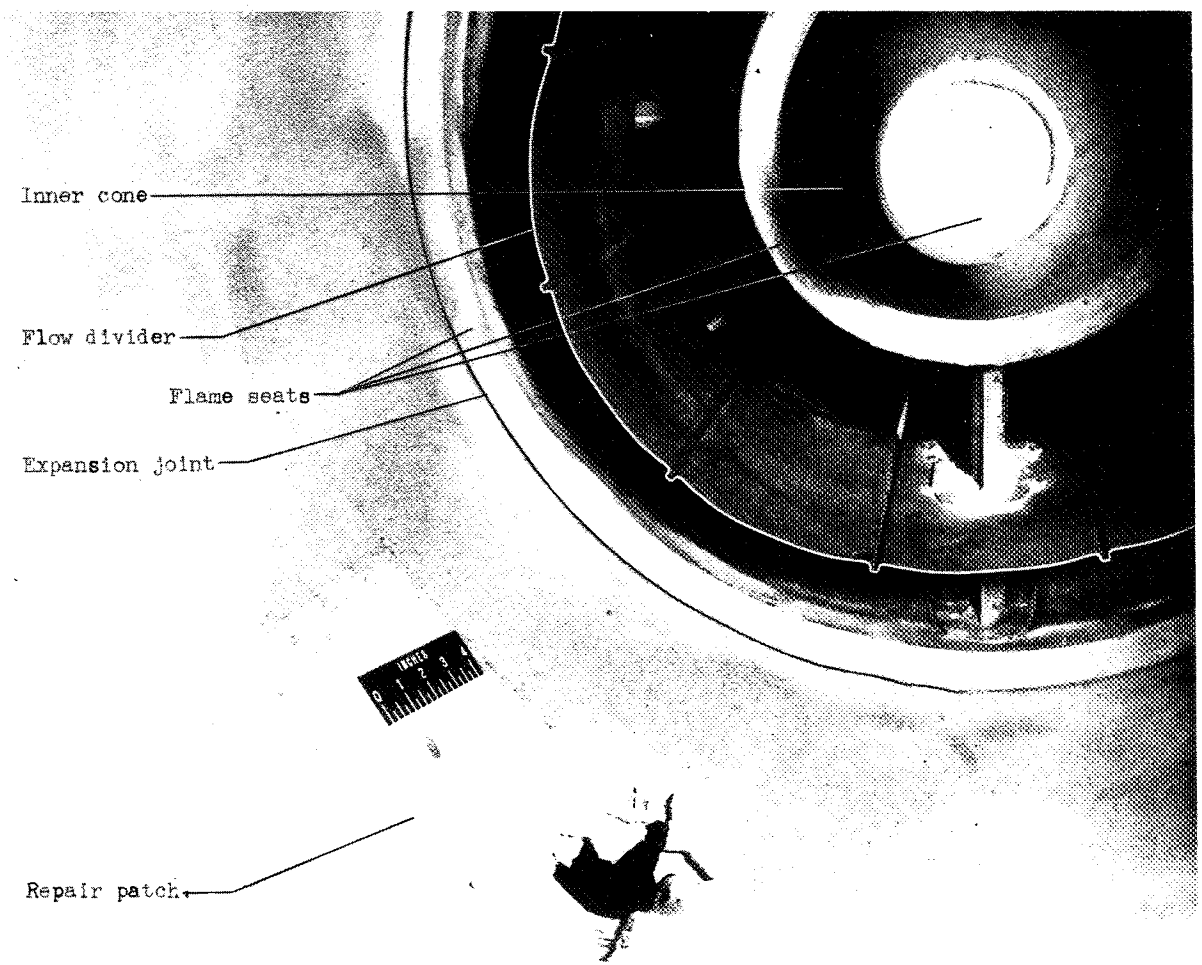


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Figure 14. - Internal view of failure of McDonnell afterburner on J34 engine.