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RESEARCH MEMORANDUM

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SUMMARY

In view of the promising gains in range offered by the use of hydrogen fuel in high-altitude turbojet aircraft, the effect of extremely high altitude operation on the performance and operating characteristics of two current turbojet engines was investigated using both gaseous hydrogen and JP-4 fuels. Component and over-all performance data were obtained with JP-4 fuel over a range of altitudes from about 40,000 to 80,000 feet at a flight Mach number of 0.8, and with hydrogen fuel at altitudes from about 70,000 to 90,000 feet at the same flight Mach number.

The use of hydrogen fuel provided stable engine operation to the facility altitude limit of about 90,000 feet at a flight Mach number of 0.8. In comparison, engine operation with JP-4 fuel was limited by combustion blowout at altitudes between 75,000 and 80,000 feet at a Mach number of 0.8. Furthermore, combustion with JP-4 fuel was relatively unstable at altitudes above 60,000 feet. In view of its high heating value, the specific fuel consumption obtained with hydrogen fuel was only about 40 percent of that obtained with JP-4 fuel.

At the extremely high altitude conditions, engine performance was significantly poorer than at low altitudes. The major portion of the performance losses at high altitudes was contributed by the compressor because of the low Reynolds number, and by the combustor because of low combustion efficiency. At altitudes as high as 75,000 feet, the loss in thrust amounted to about 12 percent and the rise in specific fuel consumption was as much as 12 to 35 percent. A variable-area exhaust nozzle is a definite necessity for high-altitude operation, since fixed-nozzle operation nearly doubled the thrust losses. In addition, the operating margins of the compressor and turbine shrank to almost nothing at extremely high altitudes, indicating the need for more adequate margins in the design of high-altitude engines.

INTRODUCTION

The requirement that military aircraft fly farther and higher has led to an intensive search for higher energy fuels as a means of extending aircraft performance. One such fuel that has been receiving considerable attention recently is liquid hydrogen. The analysis of reference 1 indicates the advantages and possible applications of this fuel for very long range high-altitude aircraft. It is concluded therein that within the state of the art and progress anticipated, turbojet-powered aircraft designed for liquid-hydrogen fuel may perform several important military missions that comparable aircraft using conventional hydrocarbon fuel cannot accomplish. One of the principal advantages shown in reference 1 for the hydrogen-fueled aircraft is its outstanding subsonic range capability at altitudes as high as 80,000 to 90,000 feet.

With the possibility in view of operating turbojet-powered aircraft at altitudes as high as 90,000 feet, the need exists for research information on turbojet operational characteristics with hydrogen fuel, as well as engine and component performance data at these high altitudes. To provide such information, two current turbojet engines were operated in the NACA Lewis laboratory altitude facilities at conditions corresponding to altitudes as high as about 90,000 feet at a flight Mach number of 0.8. Both engines were initially operated with JP-4 fuel up to their altitude limits. The fuel injectors were then modified for the use of gaseous-hydrogen fuel, and the engines were operated with this fuel at altitudes between about 70,000 and 90,000 feet. To obtain a better indication of the potentialities of liquid hydrogen, one of the engines was also operated with a special combustor developed at the laboratory specifically for operation with hydrogen fuel. Steady-state performance data were obtained, and some of the engine operating characteristics, such as engine operating range and compressor stall limits, were determined.

These experimental data are summarized in this report to illustrate the engine performance and operating characteristics with hydrogen fuel as compared with those with JP-4 fuel. The over-all and component performance data obtained are presented up to the very high altitude conditions attainable with hydrogen fuel. These data indicate some of the factors that should be considered in the design of future engines in order to help alleviate the adverse effect of high altitude on performance and, thereby, more fully realize the advantages offered by hydrogen fuel.

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APPARATUS

Engines

Two current turbojet engines in the 7500- to 10,000-pound-thrust class were used and are referred to in this report as engines A and B. Both are single-spool axial-flow engines incorporating current engine design practice. A few specific features and dimensions of the components pertinent to the results are described in the following paragraphs.

To indicate the approximate geometry of the compressor and turbine blading of the two engines investigated, a few compressor and turbine dimensions are presented in the following table:

	Engine A	Engine B
Compressor		
Inlet hub-tip ratio	0.51	0.55
Inlet guide-vane tip chord, in.	1.6	2.0
First-stage rotor tip chord, in.	2.1	2.2
Inlet guide-vane tip solidity	.81	.82
First-stage rotor tip solidity	.85	.94
Turbine		
Inlet stator mean chord, in.	1.4	1.8
First-stage rotor mean chord, in.	1.2	1.6
Inlet stator mean solidity	1.8	1.5
First-stage rotor mean solidity	1.5	1.4

The combustor on engine A was of the annular type and that on engine B was cannular. For hydrogen-fuel operation the combustors were modified, and the fuel was injected as shown in figure 1. The injectors used in engine A, which had a vaporizing combustor, consisted of tubes bent to inject the gaseous fuels in an upstream direction (fig. 1(b)). The fuel injectors used in engine B were merely open-end tubes that discharged a stream of gaseous fuel downstream into the combustor (fig. 1(a)).

As mentioned previously, engine A was also operated with a special combustor that was developed in a segment combustor rig at the Lewis laboratory for use specifically with gaseous-hydrogen fuel. This configuration of engine A is referred to as engine A-1. Details of this annular-type hydrogen combustor are shown in figure 2. One object of this design is to shorten the combustor length, which, if successful, offers the possibility of shortening the length of future engines intended for use with hydrogen fuel. Therefore, in the design of this combustor, advantage was taken of the fact that hydrogen has a much higher flame speed, and thus burns more rapidly than hydrocarbon fuels. These properties made it possible to shorten the length of the combustion and mixing zones to only about two-thirds that of the standard hydrocarbon-fueled combustor. Fuel was injected through orifices in two concentric manifolds located within a V-gutter flameholder, which provided a flame seat at the forward end of the primary burning zone. Secondary air was admitted in the conventional manner through large rectangular slots in the liner wall downstream of the primary burning zone.

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Installation and Techniques

Engines A and B were investigated in the altitude wind tunnel and engine A-1 in a 10-foot-diameter altitude test chamber. Air was supplied to the engine inlet at pressures and temperatures corresponding to the altitudes and flight speeds being simulated. In all cases the inlet conditions were set on the basis of 100-percent free-stream ram-pressure recovery.

Because the laboratory exhaust system will not provide a static pressure in the altitude facilities below that corresponding to an altitude of about 60,000 to 65,000 feet (120 to 140 lb/sq ft abs), a special testing technique was devised to enable simulation of much higher altitudes. Instead of expanding the turbine exhaust gas in the conventional manner through a tailpipe and exhaust nozzle to the correct static pressure for the simulated altitude, a long diffuser was installed on the engine as shown in figure 3. This diffuser was designed to diffuse efficiently the exhaust gas to a Mach number of about 0.2 before discharging it into the tunnel or test chamber. A large butterfly valve installed near the diffuser exit made it possible to vary the pressure drop across the diffuser, and thus vary the turbine-outlet temperature without changing the facility exhaust pressure.

To aid in visualizing the way this technique extended the effective altitude capability of the facilities, variations in Mach number and pressure through a diffuser and through a conventional tailpipe with choked flow in the exhaust nozzle are compared in figure 4. The Mach number progressively drops through the diffuser, reaching a value of about 0.2 at the exit as compared to 1.0 at the exit of the choked nozzle. As a result, the diffuser-exit total pressure is nearly equal to the exhaust pressure. In contrast, the total pressure at the exit of the choked nozzle is nearly twice the exhaust pressure. Thus, for the same engine speed, turbine-outlet temperature, and exhaust pressure, the turbine-outlet total pressure with the diffuser installed is only slightly more than one-half of that with the standard tailpipe and convergent nozzle. This reduction in turbine-outlet pressure effectively increases the altitude limit of the facility by nearly 15,000 feet above that for operation with the exhaust nozzle choked.

Thus, diffusing the exhaust gas in this manner before discharging it from the tailpipe permitted operation of the engine with the turbine-outlet total pressure only slightly higher than the pressure in the wind tunnel or test chamber. Consequently, most of the difference between turbine-outlet total pressure and free-stream static pressure that normally exists across the exhaust nozzle was utilized to extend the maximum simulated altitude of the facilities. It should be evident that, with this technique, minimum diffuser pressure losses and minimum diffuser-exit velocity will provide maximum gains in the altitude limits of the facilities.

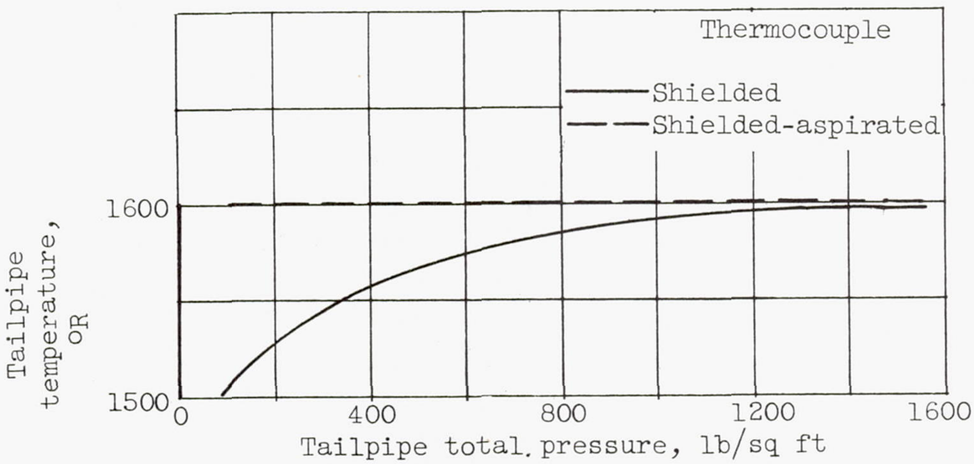
When this technique is used, thrust cannot be measured directly. Therefore, the thrust values presented herein were computed from gas flow and turbine-outlet total-pressure and total-temperature measurements. To obtain the correct thrust values, the pressure losses measured between the turbine and the exhaust nozzle of the standard tailpipe configurations at low-altitude conditions were correlated with turbine-outlet Mach number. Turbine-outlet total pressure was then adjusted at each operating condition by subtracting the tailpipe pressure loss for the corresponding turbine-outlet Mach number. A typical variation of the tailpipe pressure losses with altitude is shown in the section entitled Tailpipe.

Instrumentation

Detailed surveys of temperature and pressure were made throughout the engines to provide the measurements necessary to compute the performance of each component as well as over-all engine performance. The location of the measuring stations and the number of pressure and temperature probes installed at each station throughout the engines are indicated in figure 3. (All symbols and stations are defined in the appendix.) The instrumentation at each measuring station consisted principally of several radial survey rakes. In addition to the steady-state instrumentation, transient total-pressure measurements were obtained at the compressor inlet and outlet of engines A and B for use in determining the compressor stall lines. The JP-4 fuel flow was measured by rotometers, and gaseous-hydrogen fuel flow was measured by calibrated orifices.

One feature of the steady-state instrumentation that is of particular importance is the design of shielded thermocouples used to measure exhaust-gas temperature. It is reported in reference 2 that the radiation error of a thermocouple installed in a hot-gas stream where the gas temperature is considerably higher than the wall temperature becomes substantial at very low pressures. It is also shown that, by properly shielding and aspirating the thermocouple, essentially all the radiation error can be eliminated. A double-shielded - aspirated thermocouple is compared in figure 5 with a more conventional single-shielded thermocouple.

To illustrate the magnitude of the radiation effect on the thermocouple and, thus, the importance of using properly shielded thermocouples at very high altitudes, the variations of shielded-thermocouple and shielded-aspirated-thermocouple readings with pressure are compared in the following plot for operation at a gas temperature of 1600° R in a conventional uninsulated tailpipe: 4075



At a pressure of 100 pounds per square foot, which corresponds to the turbine-outlet pressure at an altitude of about 90,000 feet and a flight Mach number of 0.8, the shielded thermocouple read nearly 100° F lower than the shielded-aspirated thermocouple.

The turbine-outlet instrumentation used varied from one engine to another, with both types of thermocouples being used. However, all turbine-outlet temperature measurements were corrected for the radiation error in order to correspond to temperatures measured by the shielded-aspirated thermocouples.

PROCEDURE

Performance data using JP-4 fuel were obtained on engines A and B over a range of altitudes from about 40,000 to 80,000 feet at a Mach number of 0.8. Using gaseous-hydrogen fuel, the performance data of all three engines were obtained at altitudes between 70,000 and 90,000 feet at a Mach number of 0.8. Compressor stall lines were also obtained on engines A and B over the entire range of altitudes investigated. At each flight condition the engines were operated over a range of corrected speeds from about 87 percent or stall limited speed to 109 percent of rated speed. By modulating the diffuser-exit valve, turbine-outlet temperature was varied at each engine speed from the limiting value down to the minimum temperature obtainable with the diffuser valve wide open. Engine-inlet temperature was maintained at about -20° F for all operations.

The engines were started and operated in the conventional manner when using JP-4 fuel. However, some special techniques were employed and special precautions observed when operating with hydrogen fuel. Prior to starting the engines, the gaseous-hydrogen lines were completely purged with helium. The starting sequence consisted of first windmilling the engine to about 20 percent of rated speed, then energizing the standard spark ignition system, and finally opening the fuel valve. Starting the engine in this manner resulted in smooth and reliable ignition with no false starts. Starts were generally made between altitudes of 45,000 and 55,000 feet.

In the starting sequence it is of particular importance that the ignition system be energized prior to introducing the hydrogen fuel into the combustor. In one case where the fuel was introduced before the spark was energized, ignition occurred with a loud report. Although subsequent inspection of the engine revealed no damage, this method of starting is, of course, undesirable.

RESULTS AND DISCUSSION

Altitude Operating Limits

Because gaseous hydrogen has very wide combustion stability limits, its use as a turbojet fuel provides a substantial increase in altitude operating limits above those obtained using the conventional JP-4 fuel, as shown in figure 6. The maximum operable altitude of both engines A and B, when operating with JP-4 fuel under ideal and carefully controlled conditions, occurred between altitudes of 76,000 and 82,000 feet at a flight Mach number of 0.8. However, with hydrogen fuel the engines could be operated in the conventional manner up to the altitude limit of the facilities, which was about 90,000 feet at a Mach number of 0.8. At this flight condition the pressure in the combustors was about 450 pounds per square foot. The minimum operable speed of both engines above an altitude of about 60,000 feet was limited by compressor stall. Two maximum-speed limits are shown in figure 6. One is rated mechanical speed, which is obtainable at high altitudes only when the exhaust-nozzle area is increased to prevent overheating the turbine. The other limit is maximum engine speed as restricted by turbine-outlet temperature when operating with a fixed-area exhaust nozzle. The exhaust-nozzle area represented by this limit was the area that would provide limiting turbine-outlet temperature at rated speed and an altitude of 40,000 feet with JP-4 fuel used. The reason for the difference in constant-nozzle-area operating limits for the two fuels is associated with differences in exhaust-gas properties, as will be discussed further in the section Over-all Engine Performance.

Hydrogen fuel provided much more stable combustion at high altitudes than did JP-4 fuel. For example, when operating with JP-4 fuel at altitudes above 60,000 to 65,000 feet, combustion was erratic and very careful throttle manipulation was required to minimize the possibility of combustion blowout. Even when carefully controlling the fuel flow, random blowouts occurred at altitudes above about 60,000 feet. Therefore, it might be stated that the practical altitude operating limit with JP-4 fuel was at an altitude of about 60,000 feet, even though combustion could be sustained at altitudes up to 80,000 feet.

In contrast, when operating with hydrogen fuel, the fuel flow could be quickly modulated in a manner normally used during low-altitude operation without causing blowout. There were even a number of instances where combustion continued through compressor surge encounters and recoveries.

Over-All Engine Performance

Increases in altitude up to the engine or facility operating limits resulted in performance substantially below the values predicted on the basis of low-altitude data, assuming an absence of any adverse altitude or Reynolds number effects. The performance losses encountered with engines A and B are indicated in figure 7 for operation with both fuels at a flight Mach number of 0.8. This figure shows the variations of maximum corrected net thrust and corrected specific fuel consumption with altitude for fixed- and variable-exhaust-nozzle-area operation and the exhaust-nozzle-area variation required for rated-speed and limiting-temperature operation at all altitudes. Values of corrected net thrust and nozzle area are referenced to those for rated speed and limiting turbine-outlet temperature operation at an altitude of 40,000 feet.

One important comparison illustrated in figure 7 is the large reduction in specific fuel consumption provided by the hydrogen fuel below that for JP-4 fuel. In general, the specific fuel consumption was reduced as much as 60 to 65 percent below that obtained with JP-4 fuel. When operating at a given condition, the maximum net thrust obtained with hydrogen fuel was higher than that obtained with JP-4 fuel. Likewise, the exhaust-nozzle area for rated-speed and limiting-temperature operation was smaller when operating with hydrogen fuel. These differences are due to differences in both fuel-air ratio and properties of combustion products of the two fuels. The differences in gas properties affect turbine and tailpipe operating points, so that the turbine-outlet Mach numbers and tailpipe pressure losses are reduced with hydrogen. As is shown in the section Tailpipe, turbine-outlet Mach numbers and tailpipe losses were higher for engine A than for engine B, thus increasing the thrust more for engine A when hydrogen was used in place of JP-4 fuel.

Another important comparison shown in figure 7 is the relative performance between fixed- and variable-exhaust-nozzle operation. Operation with a fixed-exhaust-nozzle area up to an altitude of 80,000 feet resulted in net thrust losses as great as 20 to 30 percent of the corrected reference thrust with similar increases in specific fuel consumption. These performance losses with a fixed-area exhaust nozzle are attributable to two factors: the reduction in component performance due to the adverse effects of altitude or Reynolds number; and the resultant shift in the engine operating point to reduced engine speeds for limiting turbine-outlet temperature operation. Shifting the engine operating point to rated speed while increasing exhaust-nozzle area to hold turbine-outlet temperature constant made it possible to regain about 10 percent in net thrust or about one-third to one-half of the thrust loss at an altitude of 80,000 feet. This shift in operating point was accompanied by a very slight reduction in specific fuel consumption, thus indicating little change in component or cycle efficiency. It is therefore evident that the variable-area exhaust nozzle is an important component of an engine designed for high-altitude operation.

These gains in thrust that resulted from shifting the operating point to rated speed by increasing exhaust-nozzle area are mainly attributable to a shift in the compressor operating point to a higher airflow condition. This shift is illustrated on the compressor maps for both engines in figure 8. The rated-speed and limiting-temperature operating condition at an altitude of 40,000 feet is indicated by point A (figs. 8(a) and (c)). The shift in compressor operation to the reduced-speed condition shown by point B (figs. 8(b) and (d)) at an altitude of 70,000 feet resulted from the adverse effect of altitude on compressor and turbine performance, as mentioned previously. Opening the exhaust nozzle to permit rated-speed operation then shifted compressor operation to point C, which was accompanied by an increase in corrected air flow of about 8 to 10 percent, a slight increase in pressure ratio, and a reduction of about 0.02 to 0.04 in compressor efficiency.

From the over-all performance data thus far presented, it is readily apparent that large reductions in specific fuel consumption are attainable by the use of hydrogen fuel. However, when engine operation was extended to the extremely high altitudes afforded by the use of hydrogen fuel, large losses in thrust and specific fuel consumption occurred. The thrust losses became further amplified for engine operation with a fixed-area exhaust nozzle, where maximum engine speed at high altitudes was limited below rated speed to avoid overheating the turbine. Thus, a variable-area exhaust nozzle becomes a necessary engine component to enable attainment of maximum thrust when operating at extremely high altitudes.

Component Performance

Because the over-all performance losses associated with operation at extremely high altitudes are large, the data should be examined in further detail to determine the extent to which each component contributes to these losses, and whether there are any possible avenues for alleviating the large altitude effects in future engines. The portion that each component contributed to the losses in over-all engine performance at high altitudes is shown in figure 9 for operation with JP-4 fuel at rated speed and limiting turbine-outlet temperature. More than half of the thrust losses at high altitudes resulted from reduced compressor performance, which consisted principally of a drop in air flow. The thrust losses directly chargeable to reduced turbine efficiency were relatively small, although, as will be discussed in the section Tailpipe, the reduced turbine efficiency was also reflected in an increase in tailpipe pressure loss.

Reductions in combustion efficiency accounted for one-half to two-thirds of the rise in specific fuel consumption at high altitudes. The remainder of this rise was contributed about equally by compressor efficiency, turbine efficiency, and tailpipe pressure loss.

Similar data comparing the contributions of each component to the specific-fuel-consumption losses for cruise thrust operation are shown in figure 10. Also shown in this figure are the engine-speed reductions required at altitude for operation at minimum specific fuel consumption. The losses in specific fuel consumption at the cruise condition for engine A were about the same as those at maximum thrust with the combustion efficiency accounting for about one-half of the loss. However, the losses for engine B at cruise conditions were nearly twice those at maximum thrust, principally because of the greater altitude sensitivity of the compressor and turbine at the reduced engine speeds required to minimize specific fuel consumption.

The magnitude of this engine-speed reduction for engine B amounted to nearly 10 percent over an altitude range from 40,000 to 75,000 feet (fig. 10(b)), whereas the magnitude of the speed reduction for engine A was less than half as much (fig. 10(a)). These reductions in engine speed at altitude were required principally to avoid the high tailpipe pressure losses at high altitudes when the engine was operating at full speed and reduced turbine temperature. Even though engine A had the highest tailpipe pressure losses, the speed reduction at high altitudes was greatest for engine B, which resulted from the comparative insensitivity of the air flow of compressor B to engine-speed changes near rated speed.

From these data it is apparent that the greatest gains in performance at altitude can be obtained by improvements in compressor and combustor performance. Although turbine efficiency contributes only 2 or 3 percent of the loss in thrust and specific fuel consumption at an altitude as high as 75,000 feet, any improvements in turbine efficiency will also be reflected by reduced tailpipe pressure losses.

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Having indicated the relative influence of the various component performance variables on the high-altitude performance losses of these two engines, the succeeding discussion will present the performance of each component over a range of altitudes. In addition, the data for each component are examined with a view toward indicating some of the important factors that might be considered in the design of future engines to alleviate the adverse altitude effects and provide higher altitude operating ceilings.

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<u>Compressor</u>. - The variations of compressor efficiency and corrected air flow with altitude are shown in figure 11 for both engines at rated corrected speed. Increasing the altitude from 40,000 to 85,000 feet reduced the compressor efficiencies about 9 percentage points and lowered the corrected air flow by 10 to 15 percent.

Effects of altitude on the stall-limit lines of the two compressors are shown in figure 12. These data show a very significant depression of the stall line and, thus, reduction in pressure ratio margin, as the altitude was increased. For example, increasing the altitude from 40,000 to 80,000 feet reduced the pressure ratio margin of both engines by about 70 percent at rated corrected speed. It should also be noted that as altitude was increased, the low-speed end of the stall line intersected the steady-state operating line at progressively higher engine speeds, thus restricting operation only to very high engine speeds at high altitudes as shown in figure 6.

Because variations in compressor performance with altitude are associated with Reynolds number effects, compressor performance might rightfully be presented as a function of Reynolds number. However, because of the complex nature of the flow through the many compressor stages, no Reynolds number identified with a specific location within the compressor can be correctly referred to as the Reynolds number on which the performance variation with altitude is solely dependent. Generally, there is little increase in Reynolds number from the first to the last stages of a compressor, because the increase in density through a compressor is nearly offset by the increase in viscosity and reduction in compressor blade dimensions. Therefore, because the first stage plays a critical role in determining over-all compressor performance, and because the first-stage inlet conditions are most easily measured, using a first-stage-rotor Reynolds number as a representative value for the complete compressor is becoming conventional when presenting over-all compressor performance variations with Reynolds number. This Reynolds number is based on relative tip velocity and mean blade chord of the first-stage rotor blades.

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The variation of compressor performance with this first-stage-rotor Reynolds number is shown in figure 13 for operation at rated corrected speed. To enable comparisons of the performance trends with Reynolds number for compressors having significant variations in geometry, data are included for two other compressors in addition to those of engines A and B. Engine C is a 10,000-pound-thrust engine of current design, and compressor D is the experimental NACA compressor of reference 3. The significant geometry differences among the compressors are in first-stage blade chords and tip solidities as indicated on the figure. It should also be noted that compressor D is a transonic compressor, while the others are subsonic compressors. Although the rate at which the efficiency and air flow decrease with Reynolds number varies among the five compressors, the general trends are similar, with the performance falling rapidly at Reynolds numbers below 200,000 to 300,000. As Reynolds number was reduced, the air flow and efficiency of the transonic compressor decreased a greater amount than those of the others. Consequently, when referred to the sea-level performance values, the large Reynolds number effects would become even more evident.

Although the general trend of performance with Reynolds number is the same for all the compressors, there were, however, large differences among the compressors in the actual quantitative performance loss at a given Reynolds number. These differences in Reynolds number sensitivity illustrate the danger of using directly the data presented herein to predict the altitude or Reynolds number effects on similar size compressors of different or less conventional design.

Some general observations can be made from these data concerning possible design considerations that might make future compressors less sensitive to high-altitude operation. It appears that a general trend toward larger size engines should reduce the altitude effects on compressors, which may be explained by the fact that the Reynolds number varies proportionately with some significant dimension within the compressor. It was demonstrated in the experiment of reference 4 that the altitude effect on a given compressor is purely a function of Reynolds number. Consequently, if engine size is increased, the Reynolds number at a given operating condition will be correspondingly raised. The performance will thereby improve in much the same manner as the performance trends with Reynolds number (fig. 13). This point is further supported by reference 5, wherein the performance of five aerodynamically similar compressor rotors and stators was successfully correlated with Reynolds number. However, the data presented herein for several compressors of different aerodynamic design failed to generalize with Reynolds number, which suggests that the performance variations with altitude or Reynolds number are also a function of some geometric or aerodynamic variable of the compressor. Consequently, there appears to be some compressor design variable in addition to increased size that will relieve the altitude effect on performance.

Turbine. - The variations of corrected turbine gas flow and efficiency with altitude are shown in figure 14 for both engines operating at rated engine speed and limiting turbine temperature. Performance variations with altitude are remarkably similar for the two engines. As the altitude was increased from 40,000 to 85,000 feet, the turbine efficiency dropped $6\frac{1}{2}$ to $7\frac{1}{2}$ percentage points and the corrected gas flow decreased 7 percent. As has been found in other investigations, turbine performance was less sensitive to altitude than was compressor performance.

No attempt is made to present turbine performance as a function of Reynolds number as was done for the compressor. Such a correlation, which should logically be made at constant corrected turbine speed and pressure ratio or corrected work output, becomes virtually impossible when the turbine data are obtained in an engine. The reason for this difficulty is that as the altitude is increased, the pressure ratios or corrected work outputs at which the turbine can be operated at a given corrected turbine speed likewise increase. Consequently, large and somewhat questionable extrapolations would be necessary to correlate turbine performance with Reynolds number at constant turbine operating conditions.

Effects of altitude on the turbine work limit are shown in figure 15. The turbine work limit, which is sometimes referred to as turbine limiting-loading, is the maximum work output that can be obtained with given turbine-inlet conditions. The characteristics of turbine flow that so limit the work output are explained in reference 6.

The margin between the turbine operating line and work limit shown in figure 15 rapidly diminishes above altitudes of 50,000 to 60,000 feet. This reduction is due to two factors: (1) The limiting-loading line or available enthalpy drop per pound of gas decreases at altitude because of the reduced turbine efficiency; and (2) the operating line or work per pound of air required to drive the compressor increases at altitude because of the reduced compressor efficiency. Hence, an altitude limit of the turbine is eventually reached. At high altitudes the turbine operating lines of both engines rapidly approached the work limit of the turbine. The reduction in turbine loading and, thus, the increased turbine operating margin with hydrogen fuel, results from favorable changes in gas properties. These changes in gas properties enable the turbine to extract a given amount of work per pound of air flow with a lower temperature drop across the turbine and, thus, a lower value of corrected enthalpy drop. The reduced turbine loading resulting from this reduction in corrected enthalpy drop permits operation to higher altitudes with hydrogen fuel before the turbine work limit is encountered. 4075

It appears that for hydrogen-fuel operation neither engine would be capable of operating much above an altitude of about 90,000 feet. Thus, it is important in the design of very high altitude engines that attention be given to providing sufficient margin between the turbine operating line and work limit to enable satisfactory operation, even in the environment of very low Reynolds numbers.

Although turbine performance was not correlated with Reynolds number, the study of turbine data obtained at altitude on a large number of engines has led to the conclusion that turbine performance and operating margin are adversely affected by Reynolds number in much the same manner as in the compressor. Therefore, turbine data also strongly suggest that continued development toward larger size engines might well alleviate the turbine performance losses at high altitudes.

Tailpipe. - Closely associated with the loss in turbine efficiency at altitude is the increase in tailpipe total-pressure loss shown in figure 16. For both engines there was approximately a twofold increase in pressure loss as altitude was raised from 40,000 to 85,000 feet. This rise in tailpipe losses resulted from increased turbine-outlet Mach numbers at altitude (fig. 16) as the turbine approached the work limit. The compressibility effects associated with higher Mach numbers throughout the tailpipe resulted in a rise in tailpipe drag coefficient of about 75 percent for engine A and 40 percent for engine B between altitudes of 50,000 and 75,000 feet. Consequently, the tailpipe pressure loss increased not only proportionally with the velocity head through the tailpipe, but also in proportion to the increased drag coefficient.

Combustor. - It was shown in figure 10 that about one-half of the increase in specific fuel consumption at high altitudes when using JP-4 fuel was attributable to reduced combustion efficiency. When hydrogen fuel is considered for high-altitude operation, one important question is whether its combustion efficiency will also fall off at high altitudes in a similar manner. In answering this question, two arrangements for the use of hydrogen fuel should be considered. One arrangement, which might be applicable in a dual-fuel aircraft that burned JP-4 fuel at low altitudes and hydrogen at high altitudes, is the use of hydrogen fuel in a standard combustor. The other arrangement is the use of hydrogen fuel in a combustor designed solely for this fuel. The following discussion describes the performance of hydrogen fuel with these two combustor arrangements.

The variation of combustion efficiency for the standard combustor with altitude and combustor-inlet pressure is shown in figure 17 for both engines operating with JP-4 and hydrogen fuels at rated engine speed and limiting turbine-outlet temperature. Combustion efficiencies
4075 for both fuels dropped rapidly at altitudes above about 65,000 feet and combustor-inlet pressures below about 1500 pounds per square foot. With engine A, the efficiencies obtained with hydrogen fuel were from 2 to 6 percentage points higher than those obtained with JP-4 fuel at altitudes between 70,000 and 75,000 feet. However, with engine B, efficiencies with hydrogen fuel were 2 to 4 percentage points lower than those obtained with JP-4 fuel at altitudes where both fuels were used. At an altitude of 85,000 feet the efficiency with hydrogen fuel had dropped to as low as 84 percent for engine A and 79 percent for engine B.

Although the efficiency levels were not greatly different for the two fuels, the combustion limit of JP-4 fuel occurred at a combustor-inlet pressure of about 700 pounds per square foot, whereas the hydrogen fuel, as mentioned previously, still burned stably at pressures as low as about 450 pounds per square foot (fig. 17). The ability of hydrogen fuel to burn stably at very low pressures in a conventional turbojet combustor is also indicated in reference 7. The combustor of reference 7 was operated with hydrogen fuel to a pressure as low as 233 pounds per square foot where combustion was stable, although the combustion efficiency was as low as 70 to 75 percent.

The fact that the combustion efficiencies obtained with hydrogen fuel were not appreciably different from those obtained with JP-4 fuel might be due to the conventional turbojet combustor's not being properly matched to the burning characteristics of the hydrogen fuel. Some experiments were conducted in a combustor facility to develop a combustion chamber that would accommodate and take full advantage of the high flame speed of the hydrogen fuel (ref. 8). Two basic factors were considered in the design of this combustor: (1) to design the primary burning zone around the highly reactive combustion characteristics of the fuel; and (2) to shorten the combustor length to take advantage of the high flame speed and, thus, rapid burning of the hydrogen fuel.

A full-scale hydrogen combustor (fig. 3) designed on the basis of reference 8 and only about one-half as long as the standard combustor was installed and operated in engine A, as described in reference 9. The performance of this combustor is compared with that obtained with the conventional combustors of engines A and B in figure 18. The combustion efficiencies of the short combustor varied from 88 to 97 percent at combustor-inlet pressures of 500 to 1200 pounds per square foot, respectively. These efficiencies were about 3 percentage points higher than those obtained with hydrogen fuel in the standard combustor of engine A and 10 percentage points higher than those obtained with the standard combustor of engine B. These gains were obtained with a configuration representing the first effort at designing a hydrogen combustor. Thus, it would appear that further research in this area might yield additional gains in combustion efficiency.

A most important contribution of this work with the hydrogen combustor is the demonstration that combustor length can be shortened appreciably, in this case, by about one-half. Future engines designed to operate at high altitudes with hydrogen fuel might well take advantage of this length reduction to afford a substantial reduction in engine weight.

Increased turbine temperature. - Recently obtained turbine stator-blade temperature measurements at altitudes up to about 80,000 feet indicate a possibility of raising turbine-inlet temperature at high altitudes by as much as 50° to 150° F without overheating the turbine. Such an increase in temperature would, of course, provide a substantial gain in thrust at high altitudes. The data are presented in figure 19, which shows the reduction in stator-blade temperature with altitude for a constant turbine-inlet temperature and the corresponding increase in turbine-inlet temperature with altitude for a constant stator-blade temperature. As altitude was increased from 40,000 to 75,000 feet the stator-blade temperature decreased about 140° for a turbine-inlet temperature of 2100° R. Conversely, the turbine-inlet temperature could be increased about 150° over the same altitude range for a stator-blade temperature of 2000° R. Little effect was observed below an altitude of about 40,000 feet.

It is believed that this variation of blade temperature with altitude results from a decrease in convective heat transfer to the blade from the gas stream while the radiation from the blade remains constant. Thus, the stator blades would seek a continually lower equilibrium temperature as altitude is increased.

If the trends of figure 19 are valid for the turbines (rotors as well as stators) of engines A and B, the turbines could have been operated at inlet temperatures 140° higher at 75,000 feet, which would have offset the entire altitude effect on correct thrust at this altitude. However, further investigations are required to determine whether this relation exists on other engines and whether the rotor-blade temperatures are correspondingly lower at altitude.

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CONCLUDING REMARKS

Operation with hydrogen fuel substantially extended the altitude operating limits while reducing specific fuel consumption to about 40 percent of that obtained with JP-4 fuel. Combustion blowout limited operation with JP-4 fuel at an altitude of about 80,000 feet and Mach number of 0.8, although combustion was relatively unstable and random blowouts occurred at altitudes above 60,000 feet. In contrast, the hydrogen fuel burned stably at altitudes as high as 90,000 feet, which was the altitude limit of the facility. At these high altitudes the compressor was operating dangerously close to its stall limit, and the turbine was rapidly approaching its work limit. If engine operation could be extended above the facility altitude limit of 90,000 feet, the operating ceiling of the engine would be imposed by either compressor stall or the turbine work limit. Thus, engines intended to utilize the extreme altitude operation afforded by hydrogen fuel should be designed with increased compressor and turbine operating margins.

Altitude effects on engine performance associated with increases in altitude up to about 75,000 feet at a Mach number of 0.8 imposed net thrust losses as great as 12 percent and increases in specific fuel consumption of 12 to 35 percent. These performance losses were for operation with a variable-area exhaust nozzle, which allowed rated speed and temperature operation at all altitudes. With a fixed-area nozzle, maximum engine speed was so restricted at high altitudes by turbine temperature that the loss in maximum thrust was nearly doubled. This difference clearly illustrates the need for a variable-area exhaust nozzle on engines designed to operate at extreme altitudes.

The compressor contributed the major portion of the thrust loss at high altitudes, and the combustor caused most of the rise in specific fuel consumption. The air flow and efficiency of a given compressor correlated with a first-stage Reynolds number. Such correlation means that large engines are likely to suffer less than small ones from high-altitude operation. However, the rate of performance loss with Reynolds number differed from one compressor to another, even though they were of very similar size. Consequently, it appears that the performance variations with altitude are also a function of some aerodynamic design variable of the compressor, which means that increased engine size is not the sole factor that will relieve the altitude effect on performance.

Operation with hydrogen fuel in the standard combustors provided combustion efficiencies for one engine that were 2 to 6 percent higher than the efficiencies obtained with JP-4 fuel, but for the other engine were 2 to 4 percent lower than those obtained with JP-4 fuel. A combustor specifically designed to operate with hydrogen fuel provided significant increases in combustion efficiency over the values obtained when burning hydrogen fuel in the standard-engine combustors. In addition, this combustor was only about one-half the length of conventional turbojet combustors, thus offering the possibility of reductions in the length and, thus, weight of future engines.

The possibility of realizing a substantial thrust gain at high altitudes was indicated by turbine stator-blade temperature measurements that showed a reduction in turbine stator-blade temperature of 140° F as the altitude was increased to 80,000 feet at rated turbine-inlet temperature. Further investigation is, of course, required to determine whether such a variation exists in the rotor blades, and whether it is common to other engines.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, June 11, 1956

APPENDIX - SYMBOLS

A area, sq ft

c chord length

F thrust, lb

H total enthalpy, Btu/lb

M Mach number

N rotational speed, rpm

P total pressure, lb/sq ft

Re Reynolds number

V velocity, ft/sec

w weight flow, lb/hr

β function of γ, $\frac{0.74}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}$

γ ratio of specific heats

δ ratio of total pressure to NACA standard sea-level pressure of 2116 lb/sq ft

θ ratio of total temperature to NACA standard sea-level temperature of 518.7° R

η efficiency

μ absolute viscosity, lb-sec/sq ft

ρ density, slugs/cu ft

Subscripts:

a air

B combustor

C compressor

cr critical

f fuel

g gas

N exhaust nozzle

n net

T turbine

tp standard-engine tailpipe

0 free stream

1 compressor inlet

2 compressor outlet

3 combustor outlet

4 turbine outlet

5 tailpipe diffuser

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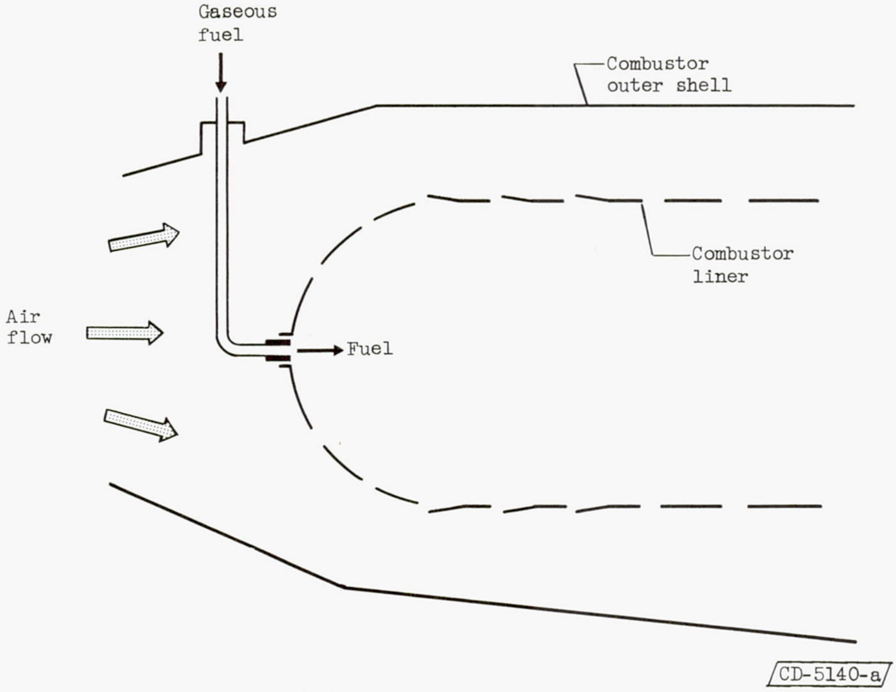
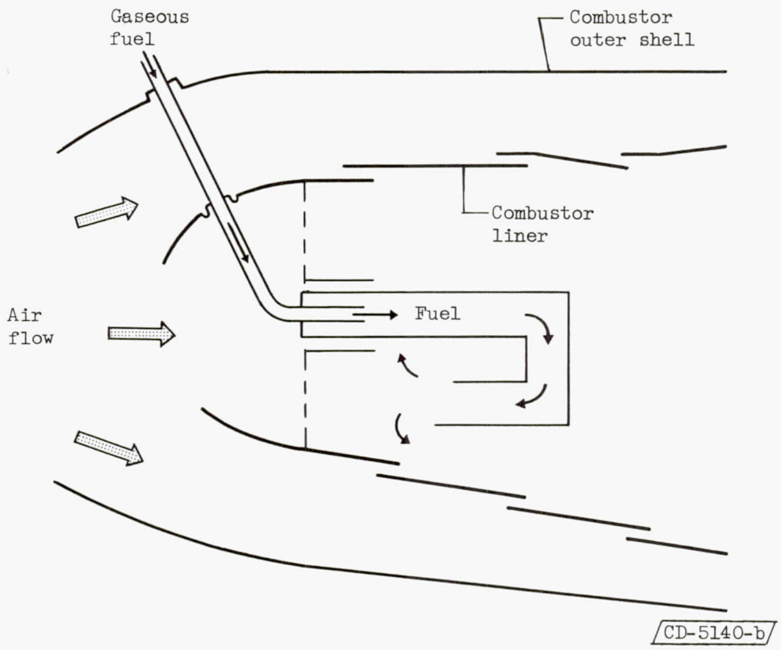


(a) Engine B; cannular combustor.



(b) Engine A; annular combustor.

Figure 1. - Schematic drawing of fuel-injector installations for operation with gaseous-hydrogen fuel.

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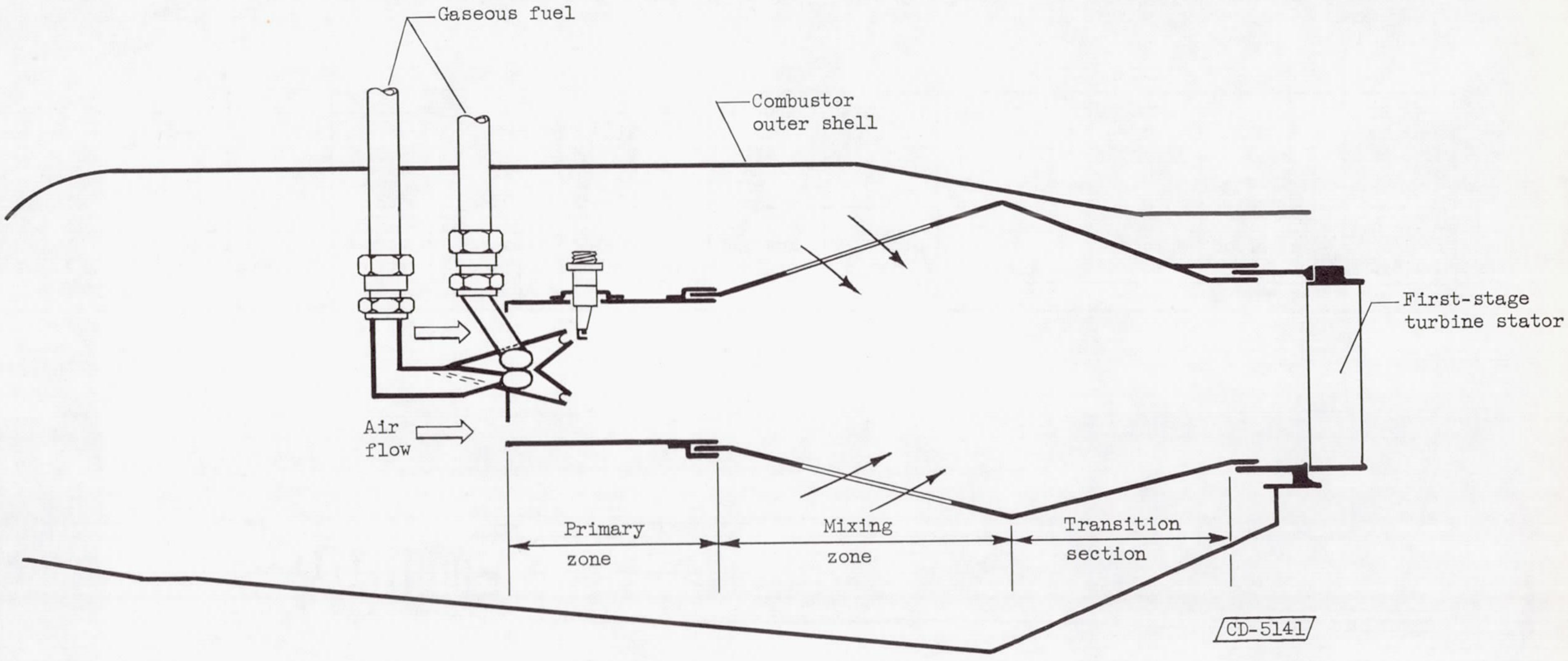
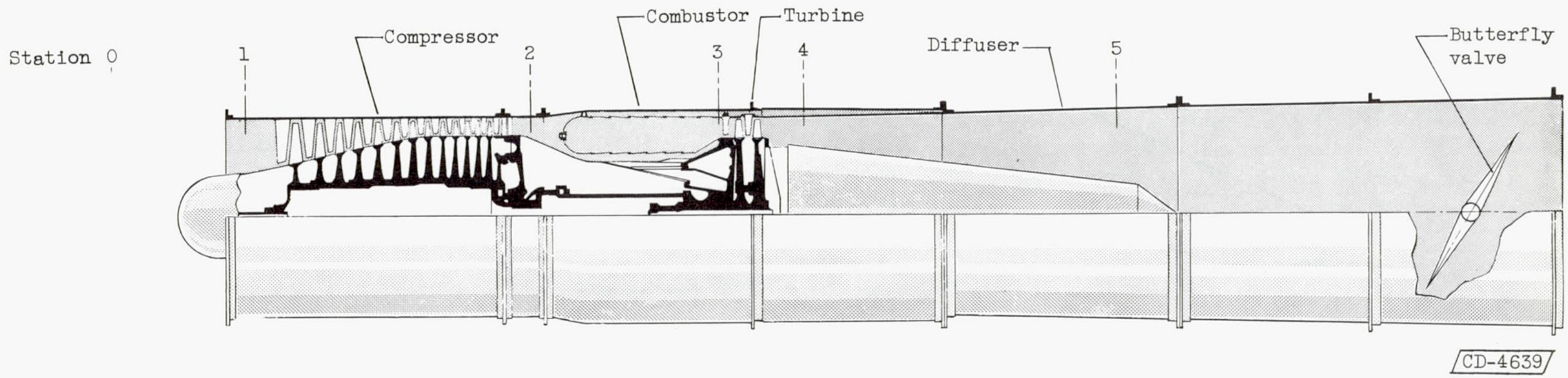


Figure 2. - Cross section of annular combustor designed for gaseous-hydrogen fuel and evaluated in engine A-1.



Station	Engine A			Engine B			Engine A-1		
	Total pressure	Static pressure	Temperature	Total pressure	Static pressure	Temperature	Total pressure	Static pressure	Temperature
1	28	16	16	28	16	16	9	--	8
2	20	--	16	20	--	20	20	--	20
3	24	--	14	16	--	4	20	--	20
4	32	--	16	32	--	8	26	4	45
5	32	--	27	32	--	32	20	4	32

Figure 3. - Typical engine and tailpipe-diffuser arrangement showing location and amount of instrumentation.

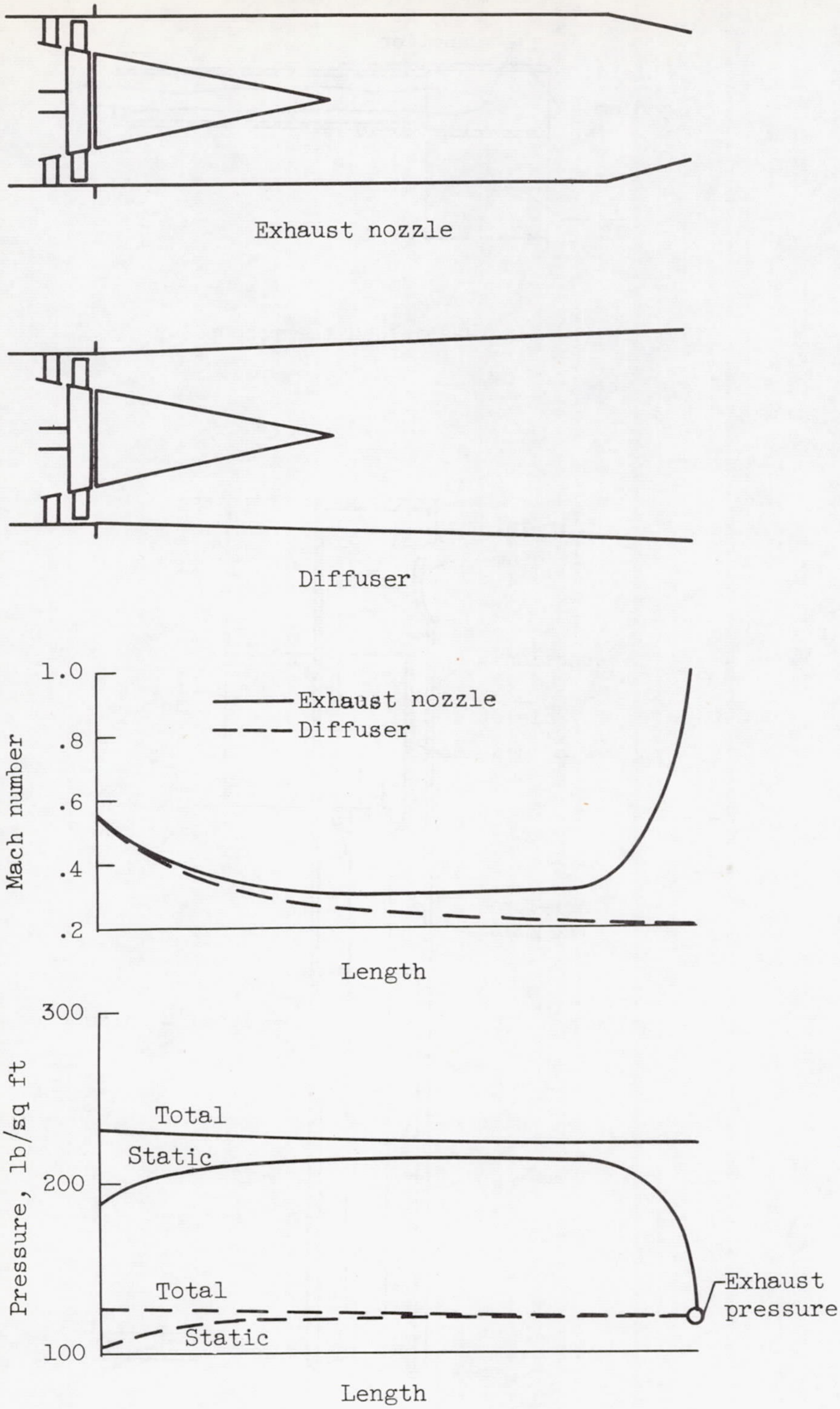
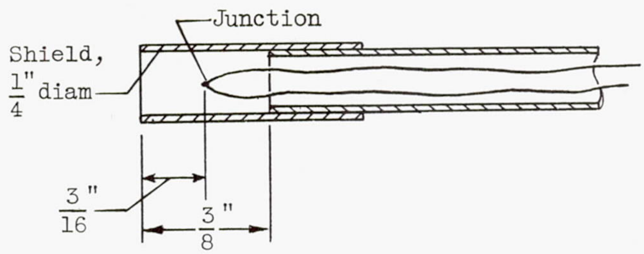
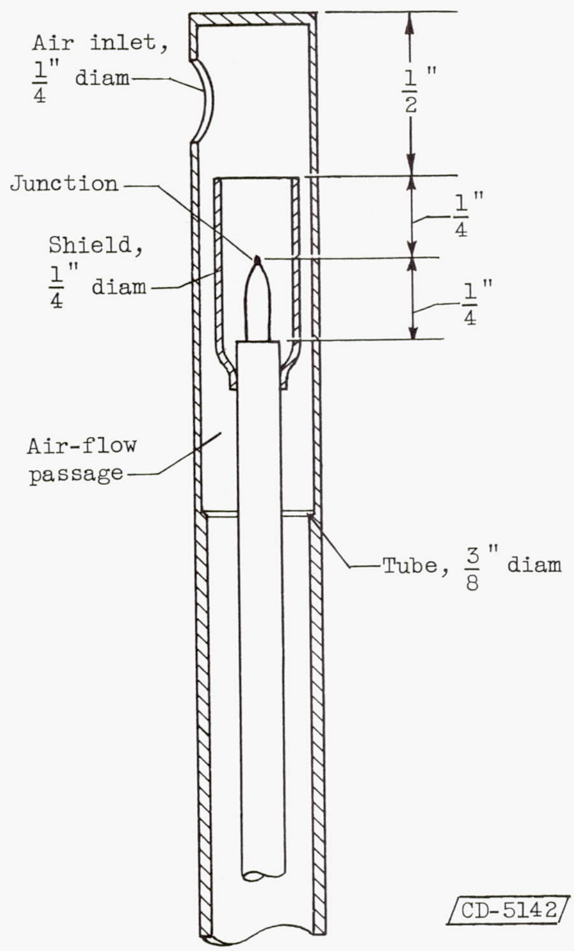


Figure 4. - Comparison of Mach number and pressure variations through tailpipe diffuser and through conventional tailpipe with choked exhaust nozzle.

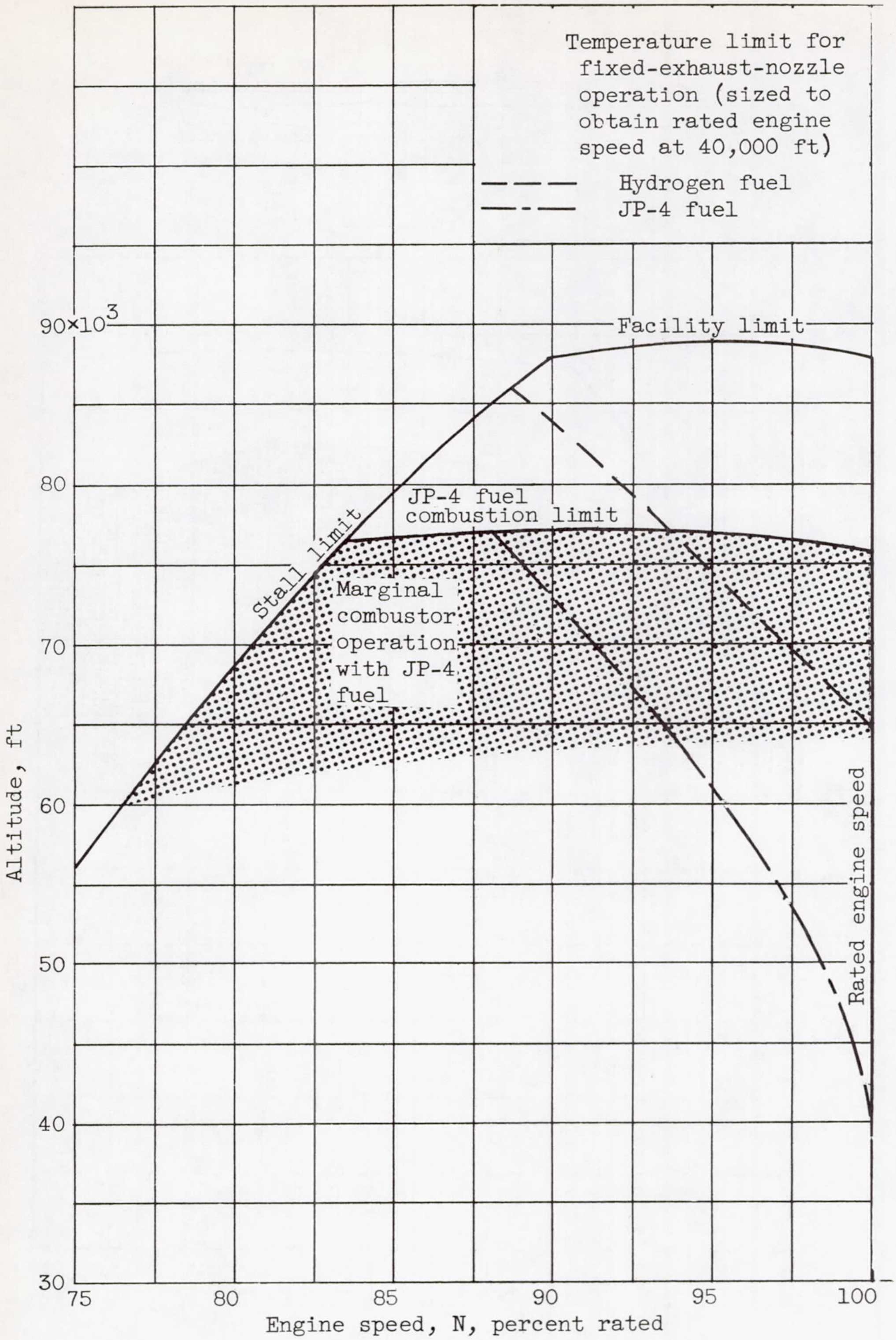


(a) Single-shielded thermocouple.



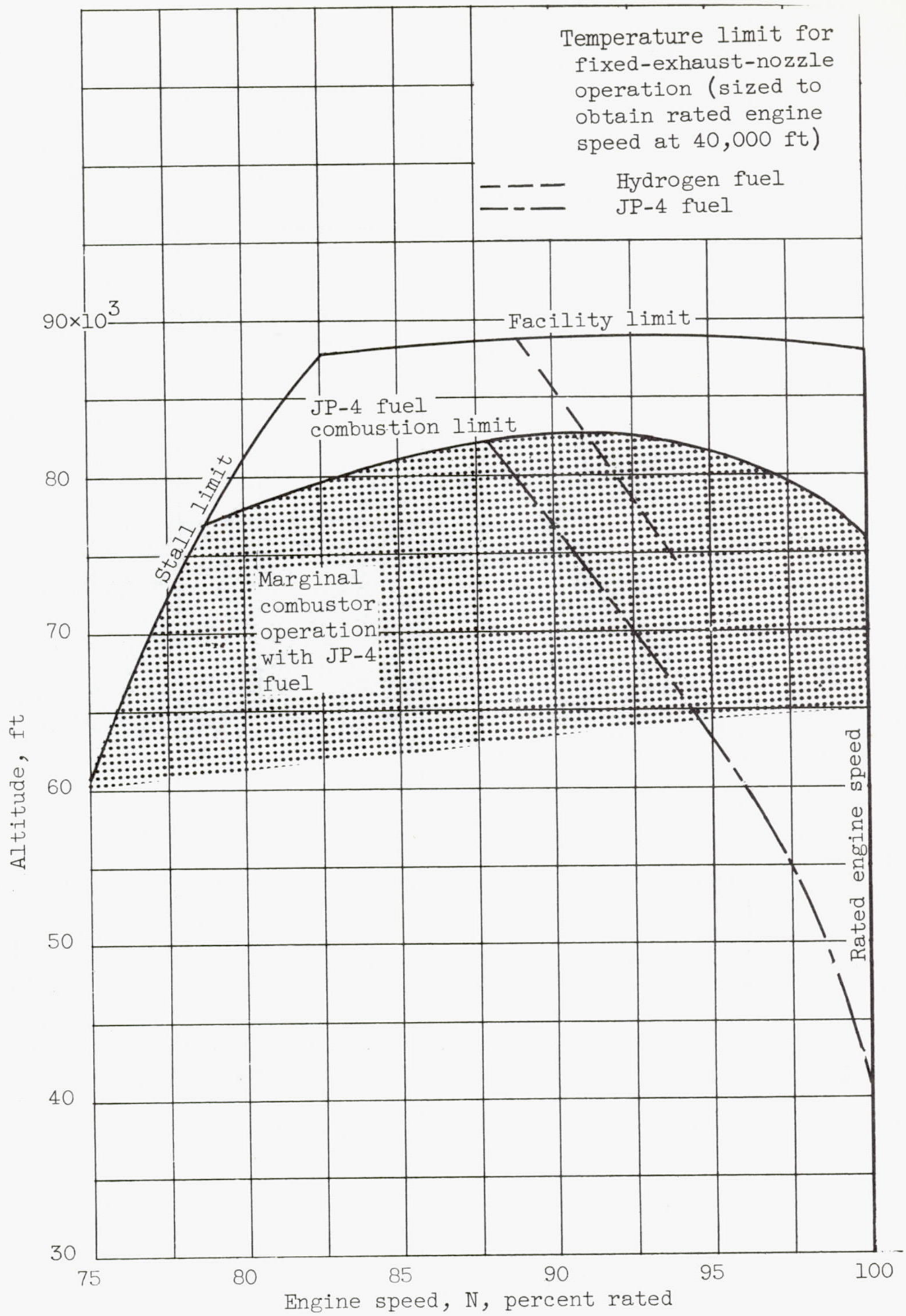
(b) Double-shielded - aspirated thermocouple.

Figure 5. - Details of tailpipe thermocouples.



(a) Engine A.

Figure 6. - Altitude operating limits at flight Mach number of 0.8.



(b) Engine B.

Figure 6. - Concluded. Altitude operating limits at flight Mach number of 0.8.

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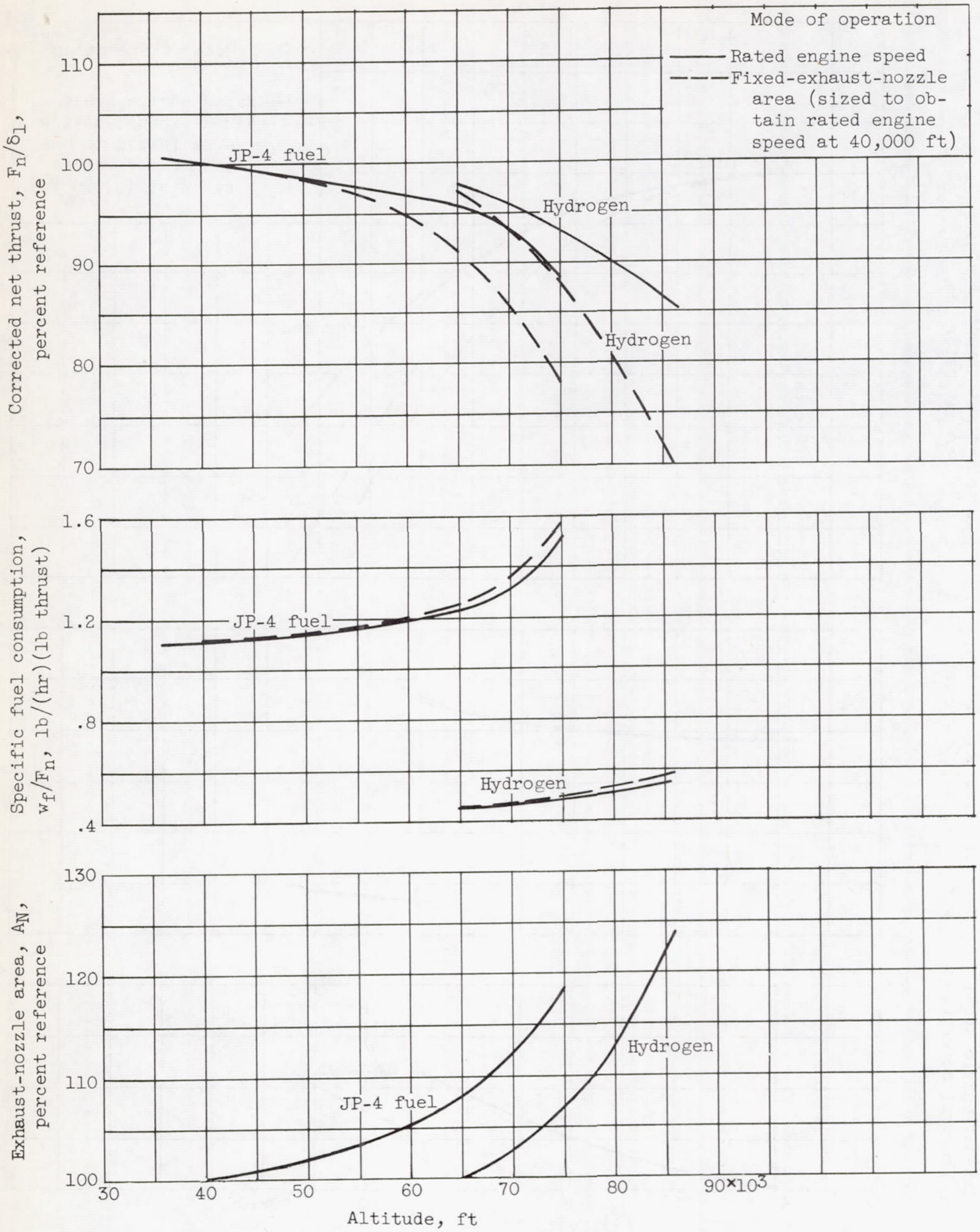

(a) Engine A.

Figure 7. - Altitude effect on performance for two modes of operation at rated exhaust-gas temperature. Flight Mach number, 0.8.

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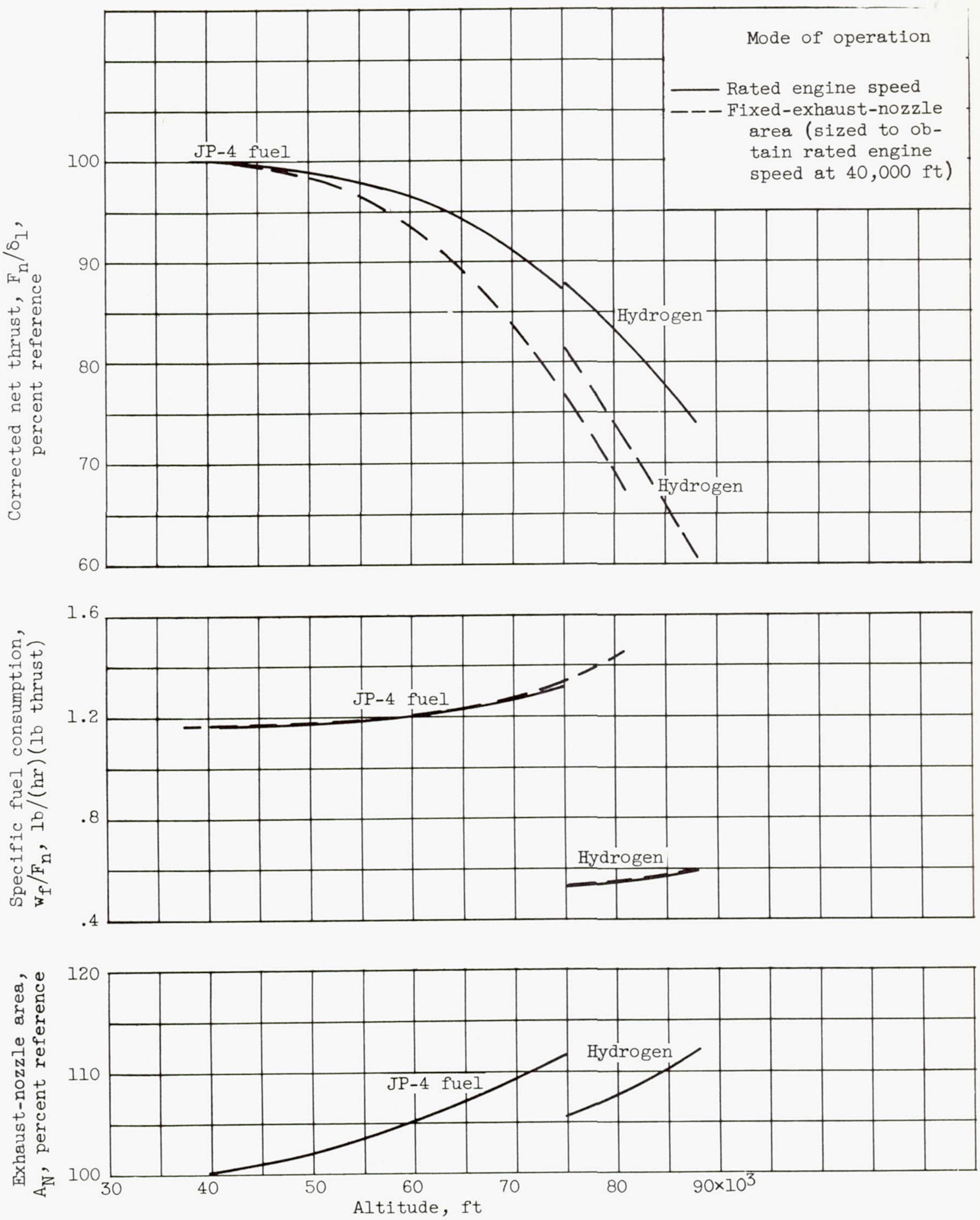

(b) Engine B.

Figure 7. - Concluded. Altitude effect on performance for two modes of operation at rated exhaust-gas temperature. Flight Mach number, 0.8.

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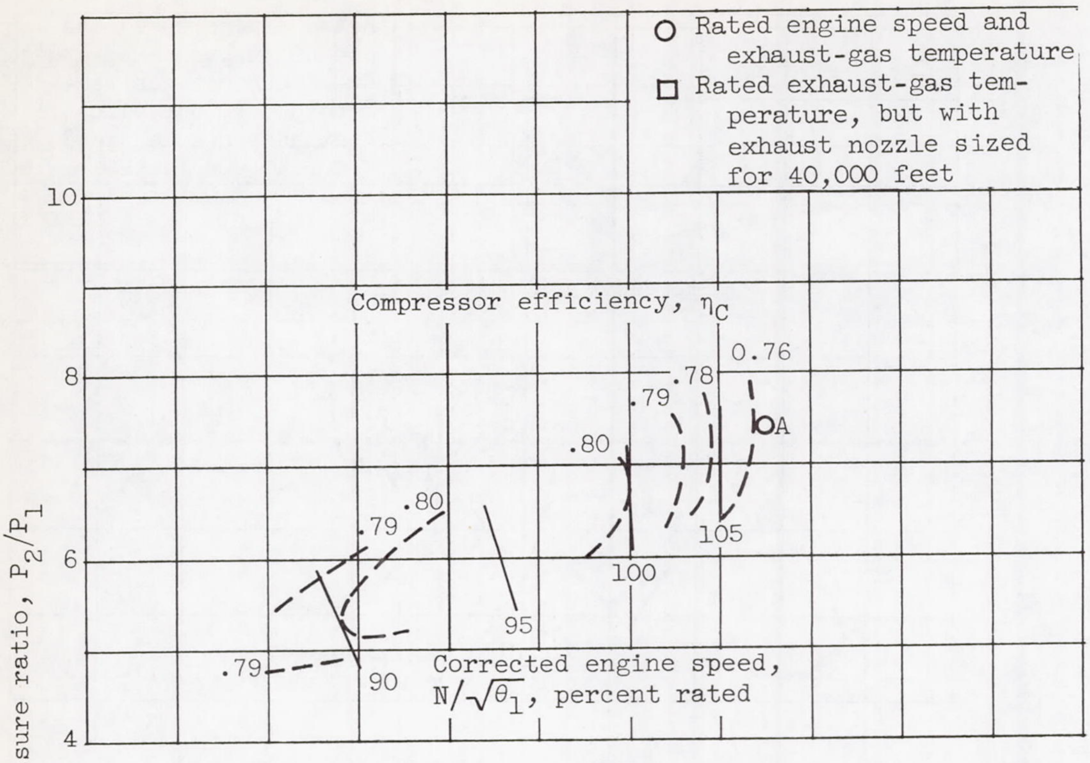
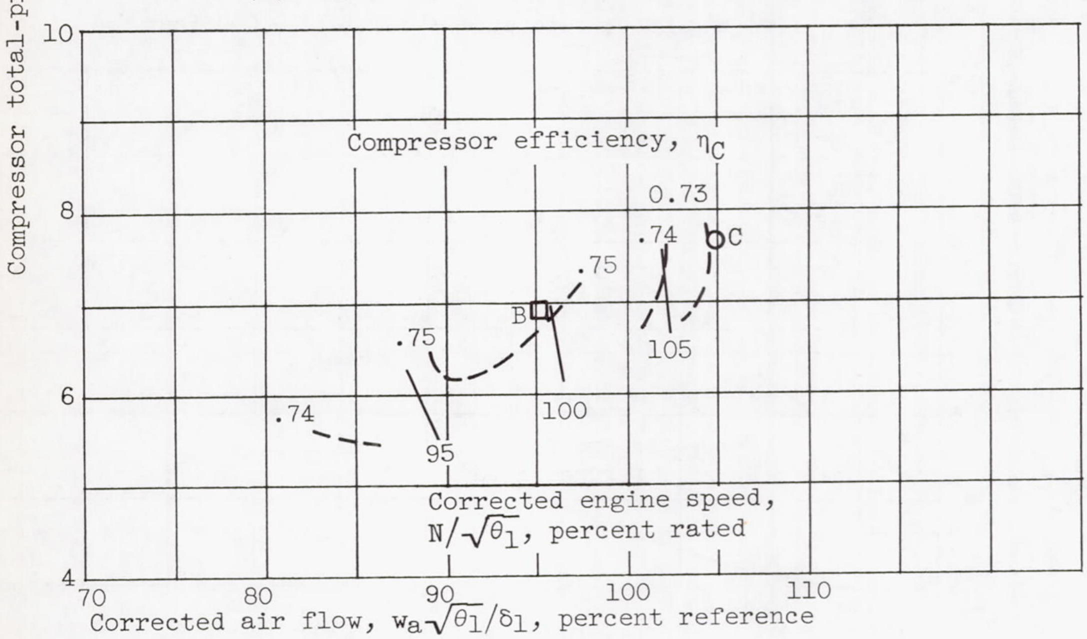


(a) Engine A; reference altitude, 40,000 feet.



(b) Engine A; altitude, 70,000 feet.

Figure 8. - Compressor maps with rated operating points superimposed. Flight Mach number, 0.8.

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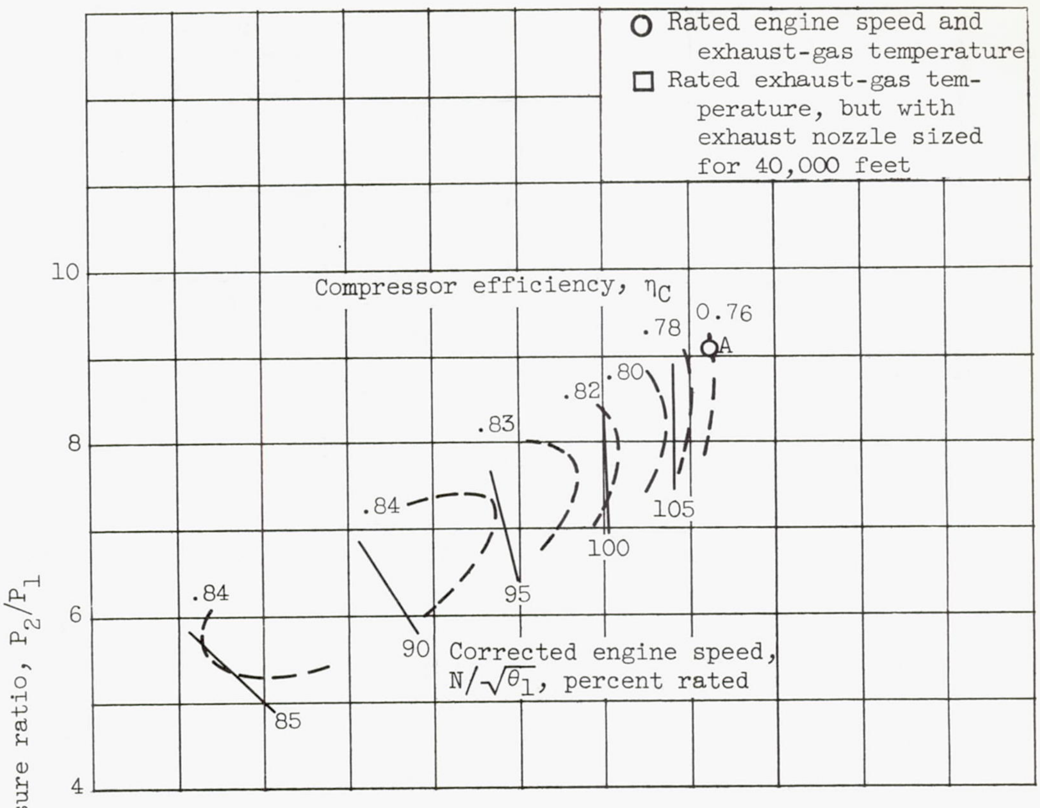
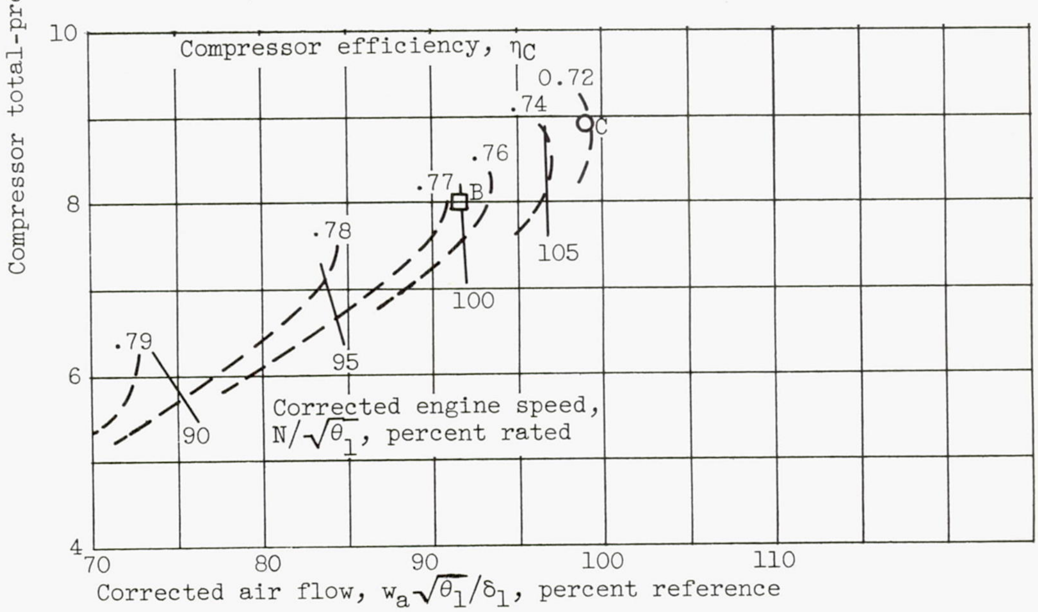
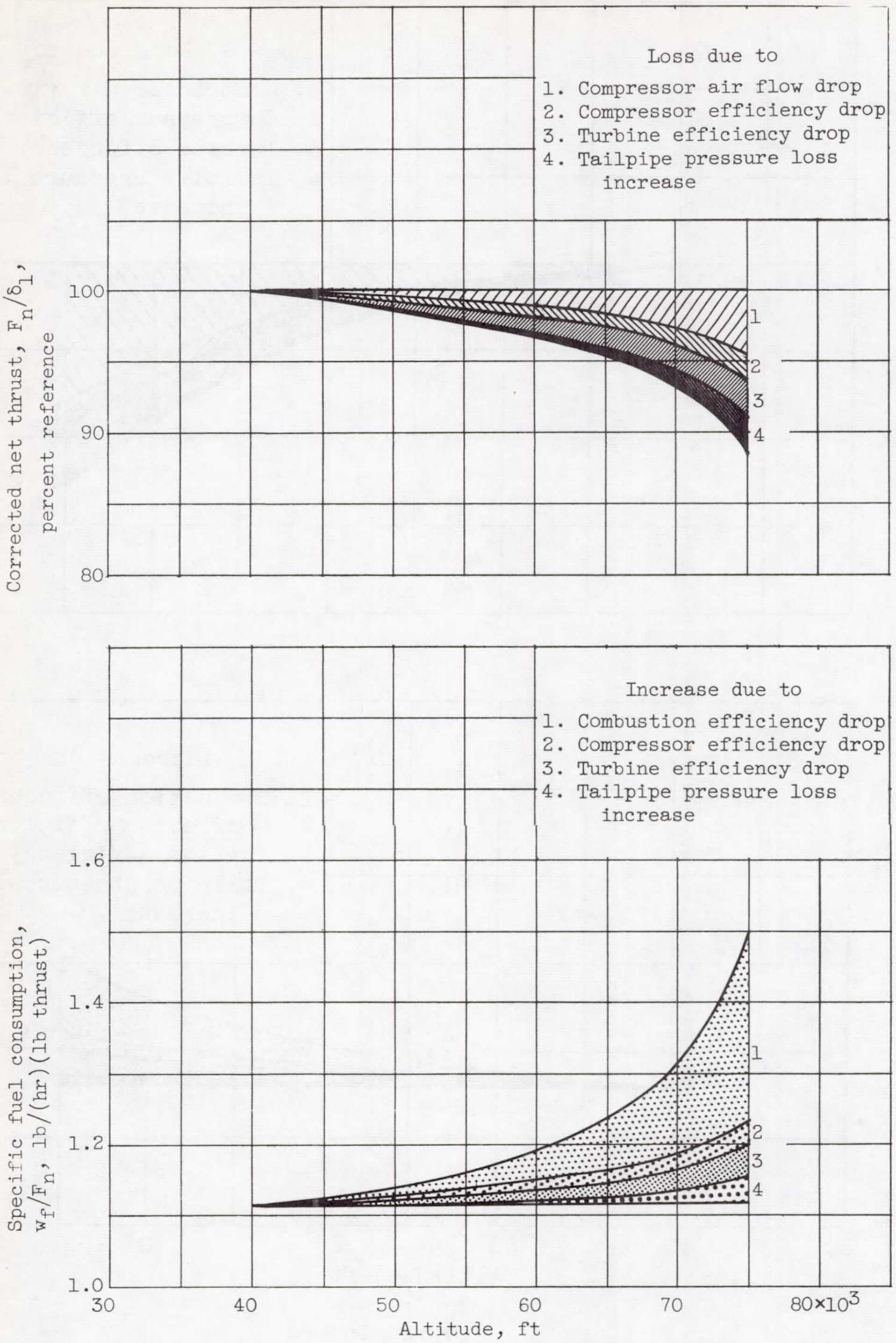


(c) Engine B; reference altitude, 40,000 feet.



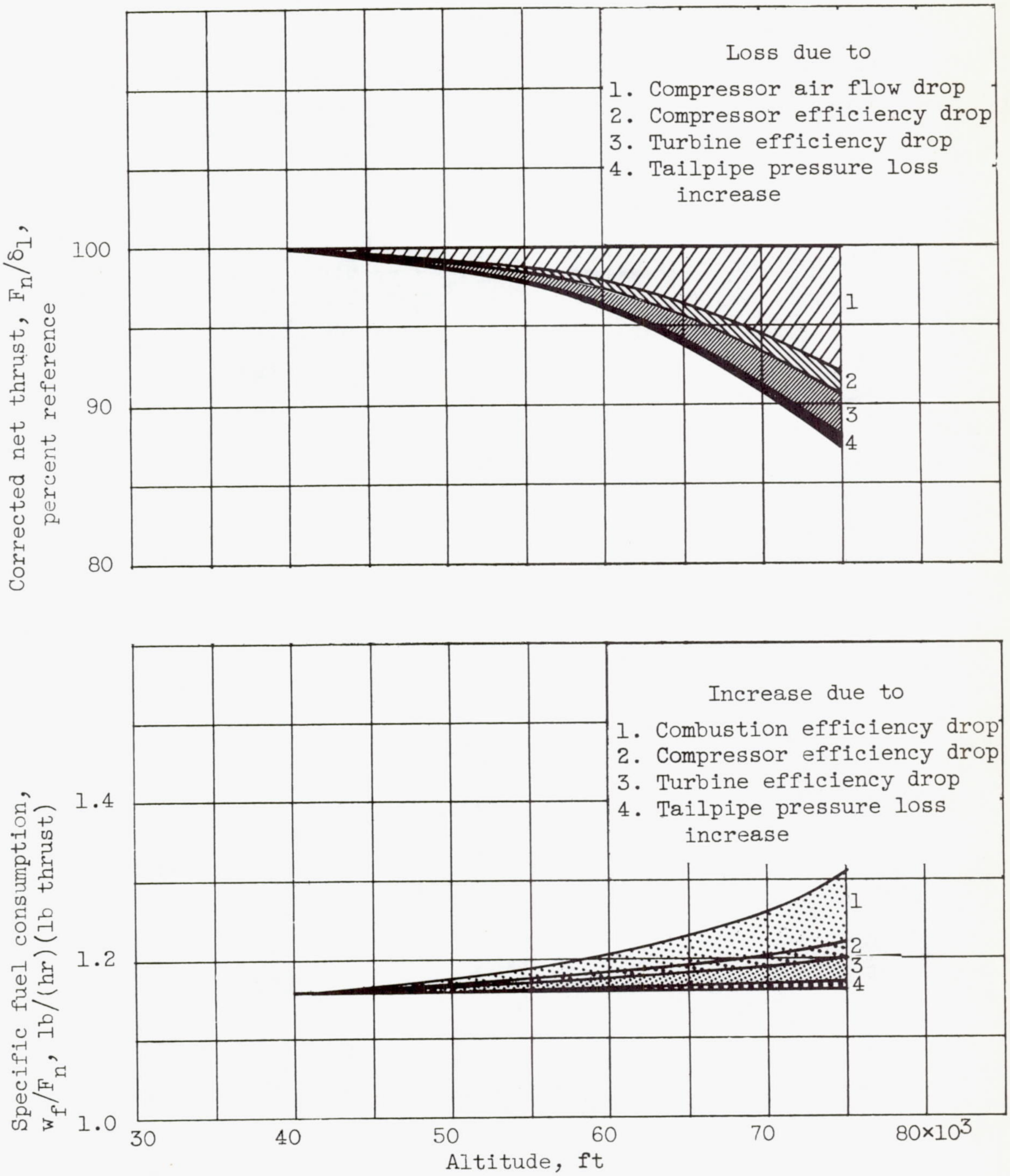
(d) Engine B; altitude, 70,000 feet.

Figure 8. - Concluded. Compressor maps with rated operating points superimposed. Flight Mach number, 0.8.



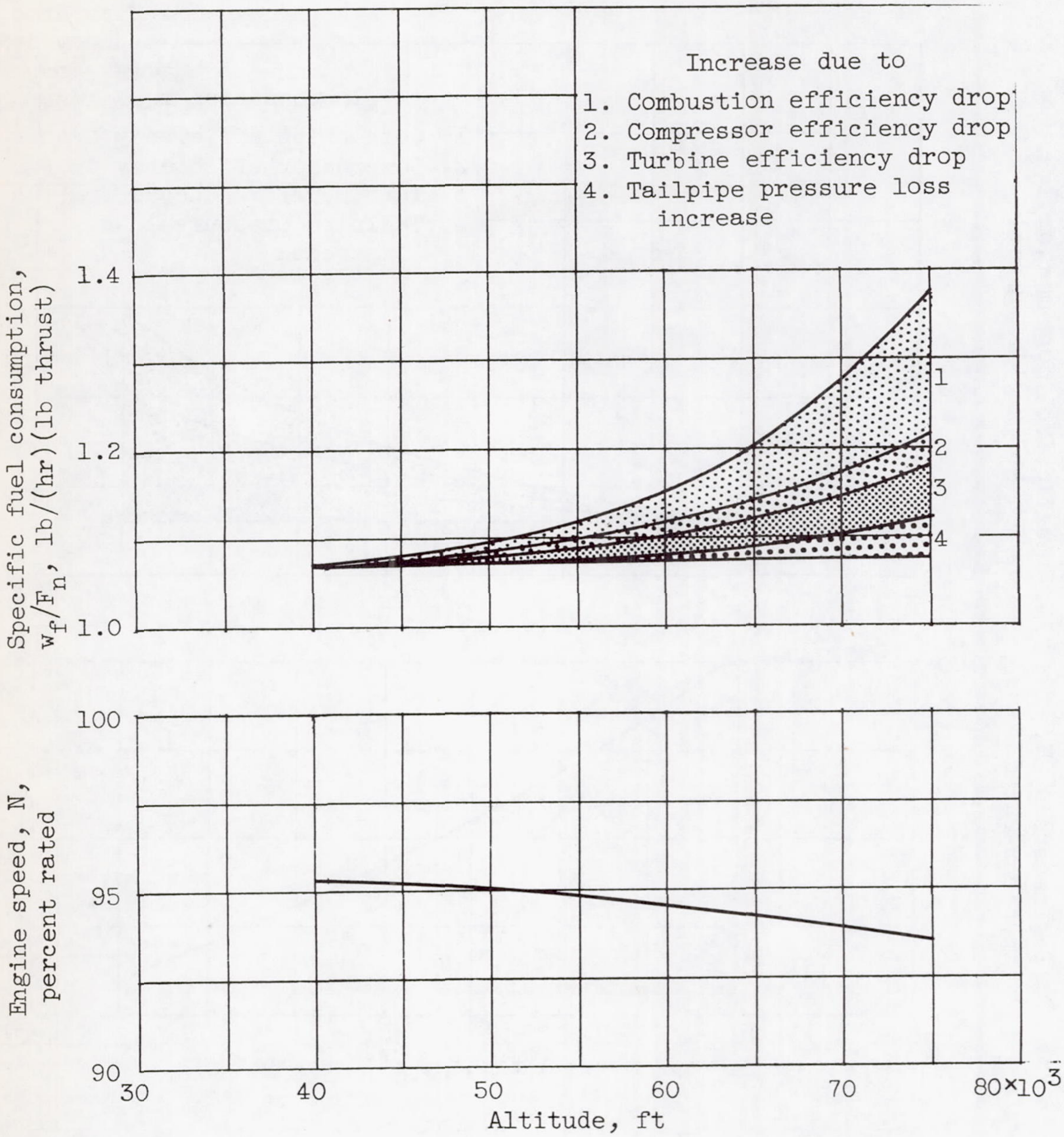
(a) Engine A.

Figure 9. - Contribution of components to altitude performance losses. Rated engine speed; rated turbine-outlet temperature; flight Mach number, 0.8; JP-4 fuel.



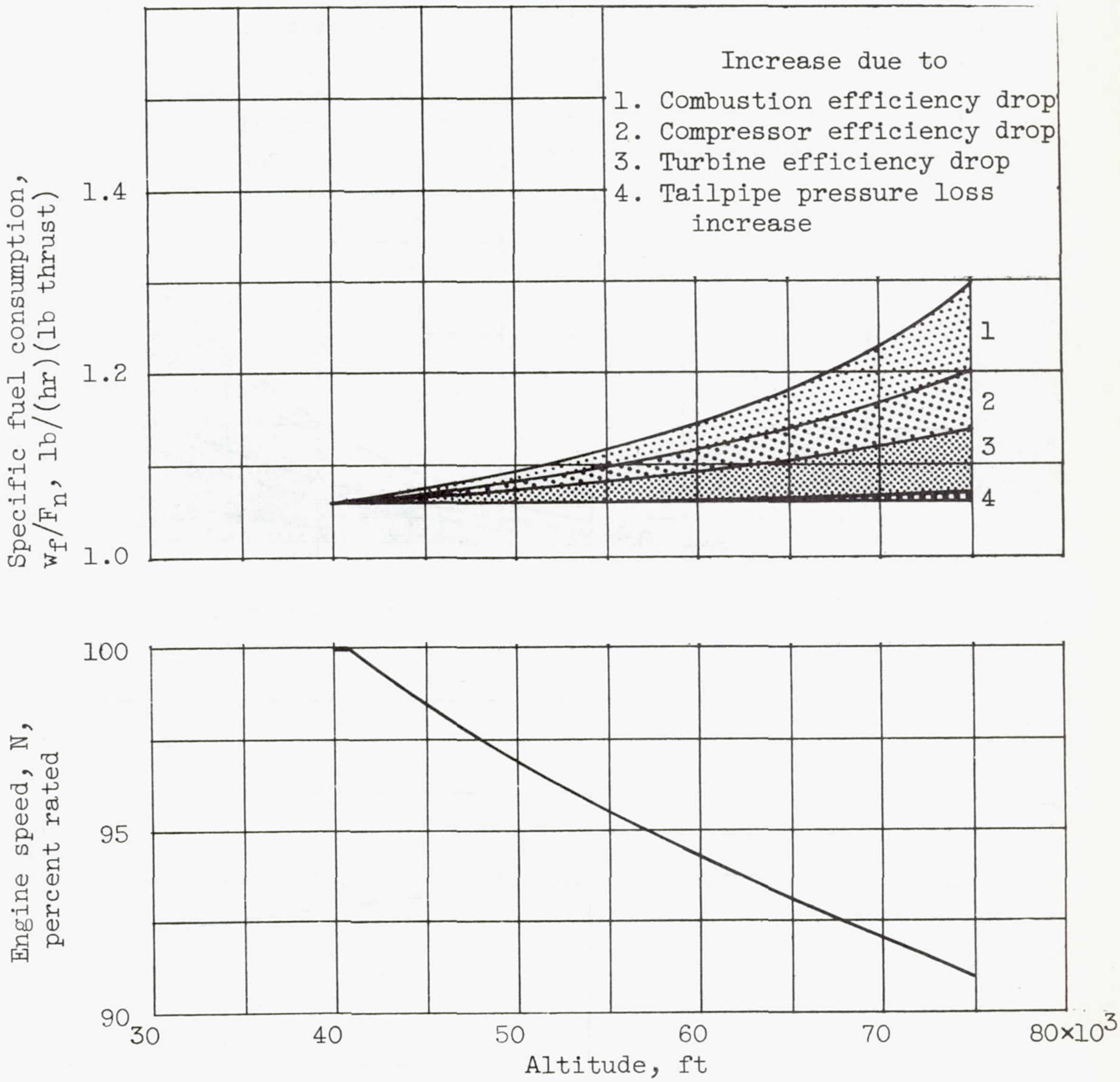
(b) Engine B.

Figure 9. - Concluded. Contribution of components to altitude performance losses. Rated engine speed; rated turbine-outlet temperature; flight Mach number, 0.8; JP-4 fuel.



(a) Engine A.

Figure 10. - Contribution of components to performance losses at 75 percent maximum thrust.



(b) Engine B.

Figure 10. - Concluded. Contribution of components to performance losses at 75 percent maximum thrust.

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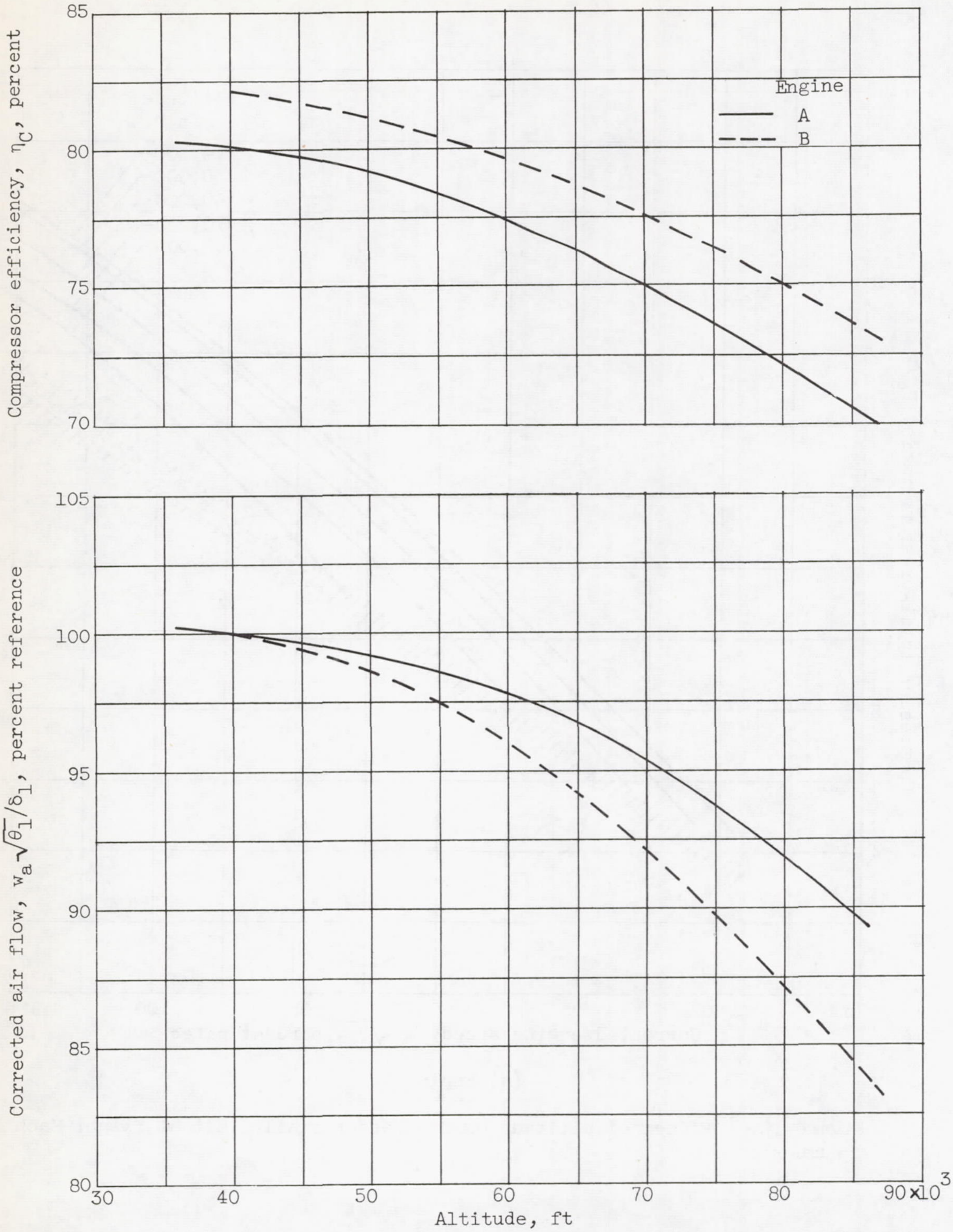

Figure 11. - Effect of altitude on compressor performance at rated corrected speed and rated compressor pressure ratio.

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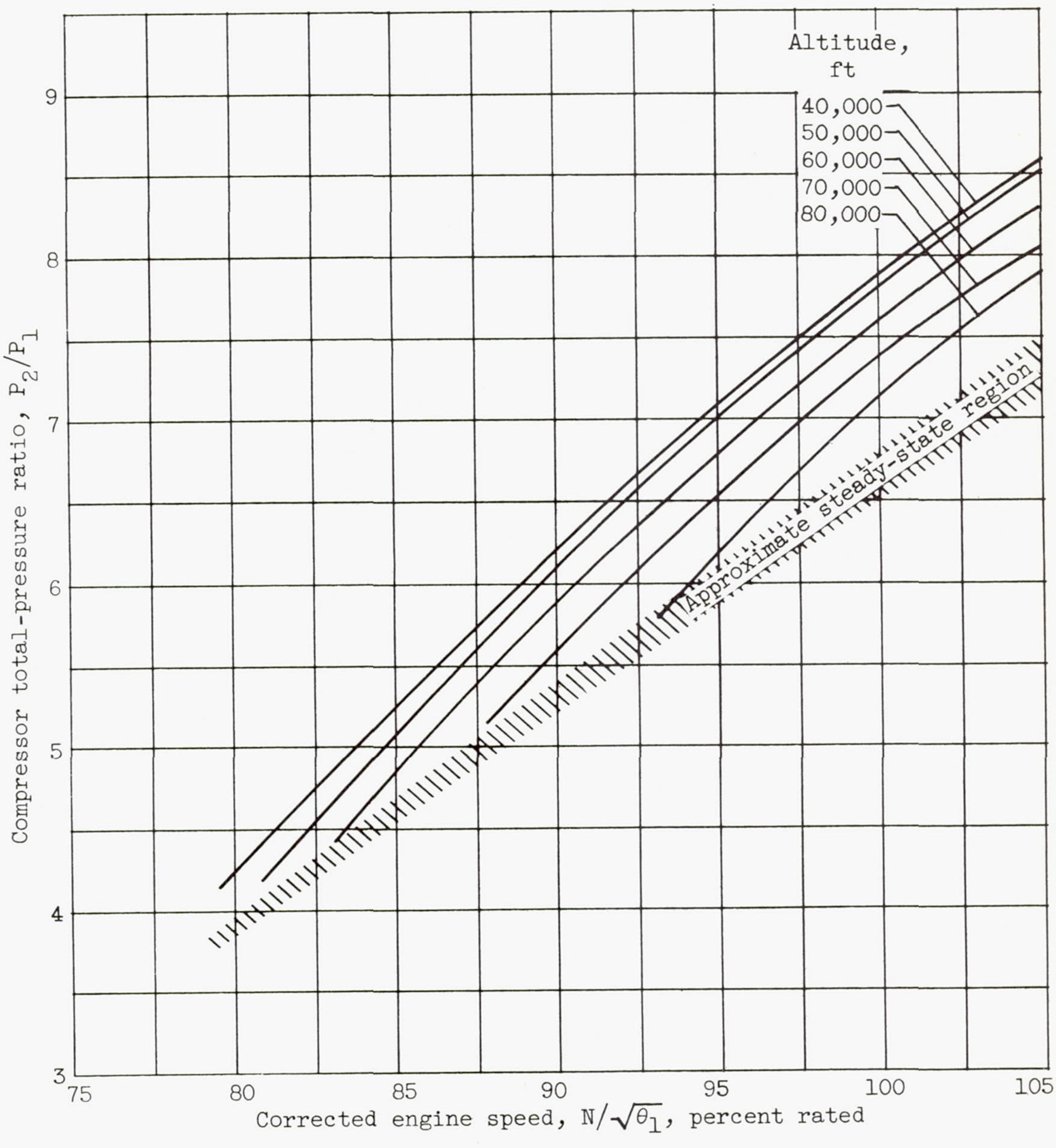

(a) Engine A.

Figure 12. - Effect of altitude on compressor stall limit at flight Mach number of 0.8.

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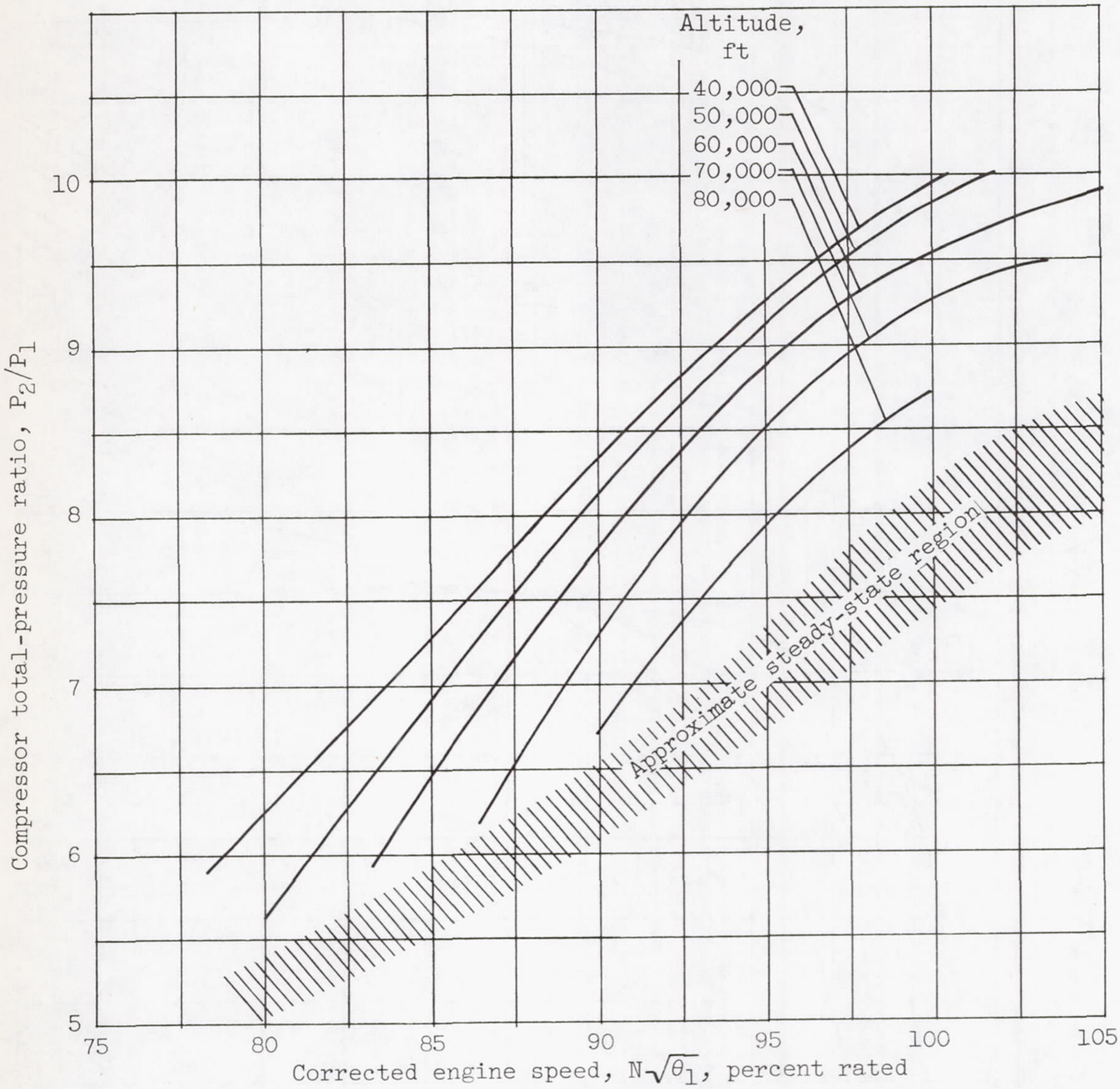

(b) Engine B.

Figure 12. - Concluded. Effect of altitude on compressor stall limit at flight Mach number of 0.8.

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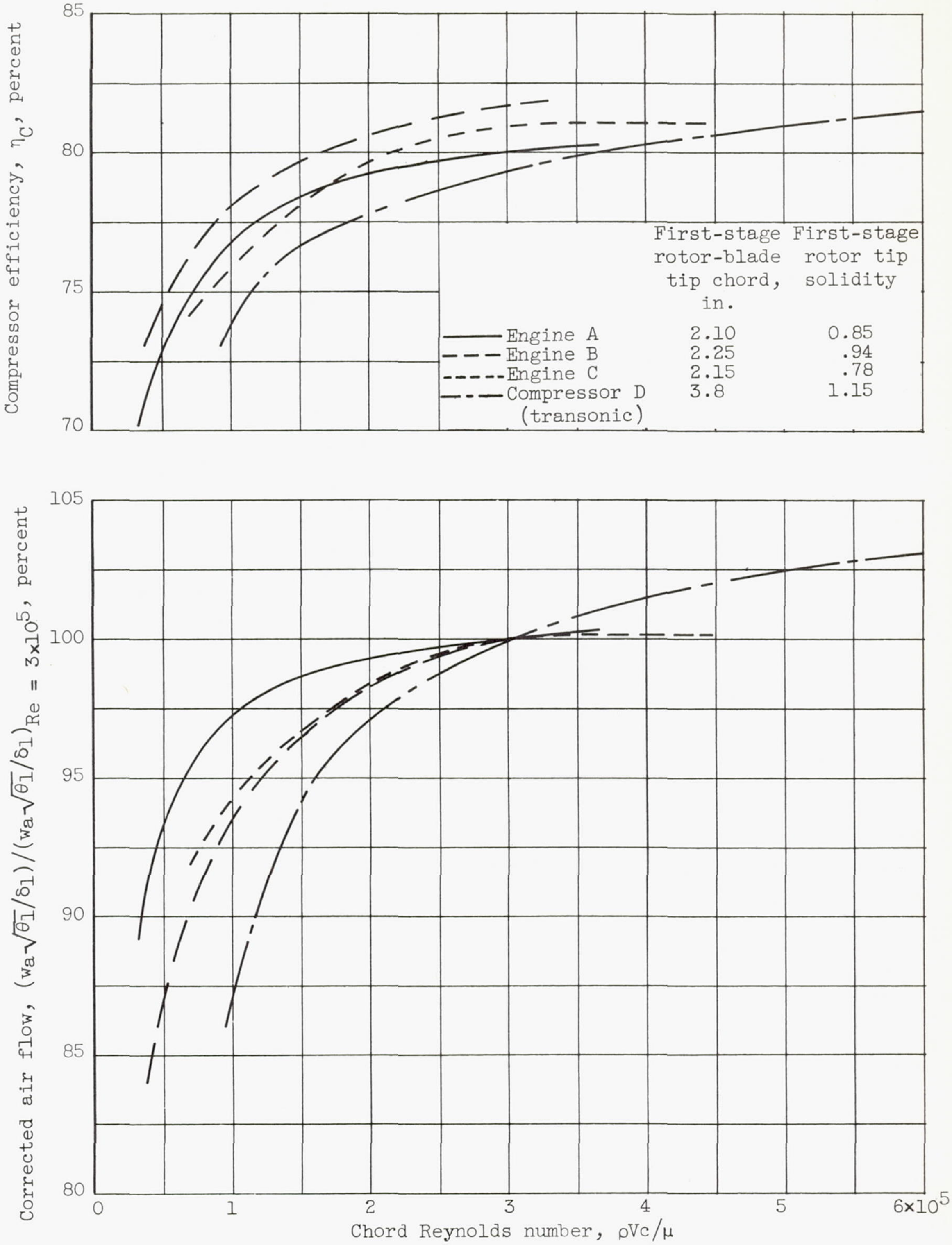

Figure 13. - Variation of compressor performance with chord Reynolds number at rated corrected speed and rated compressor pressure ratio.

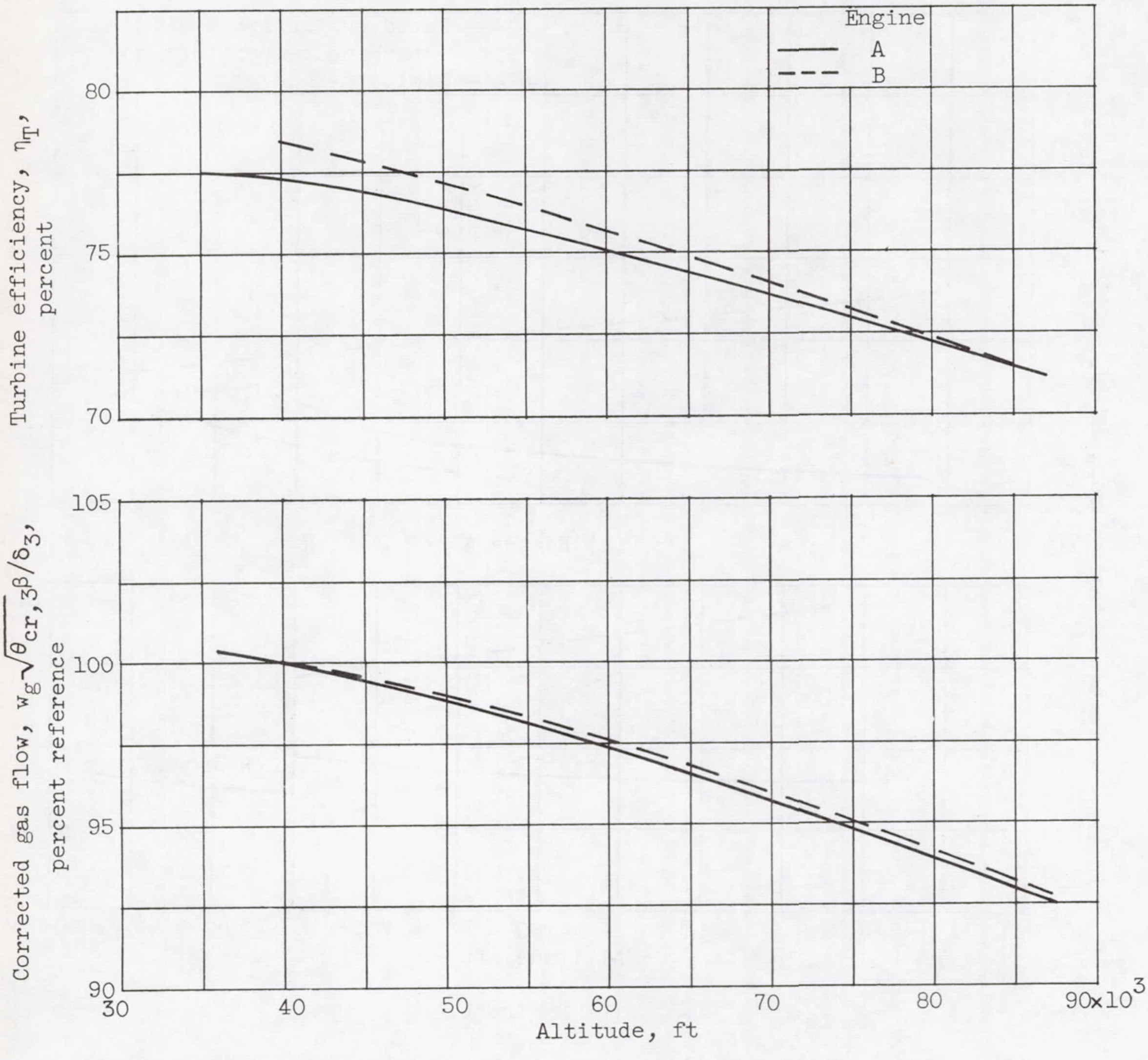

Figure 14. - Effect of altitude on turbine performance at constant corrected turbine speed (corresponding to rated engine speed and limiting turbine temperature).

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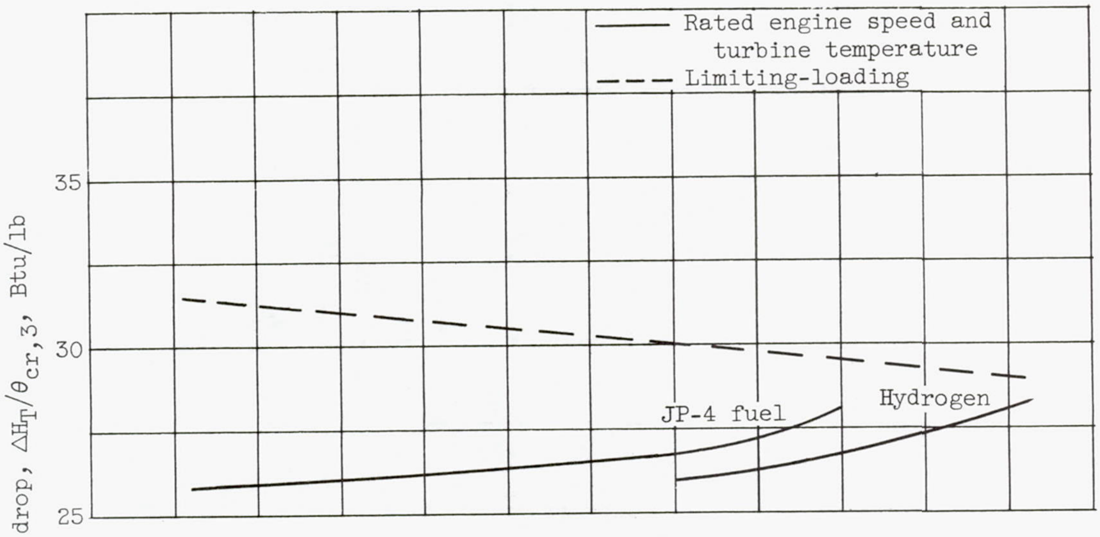
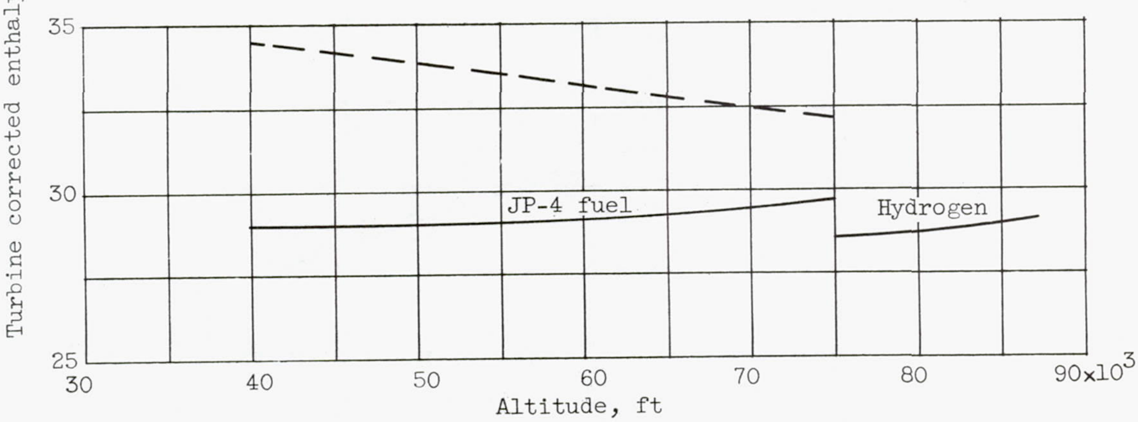


(a) Engine A.



(b) Engine B.

Figure 15. - Effect of altitude on turbine work at rated engine speed and flight Mach number of 0.8.

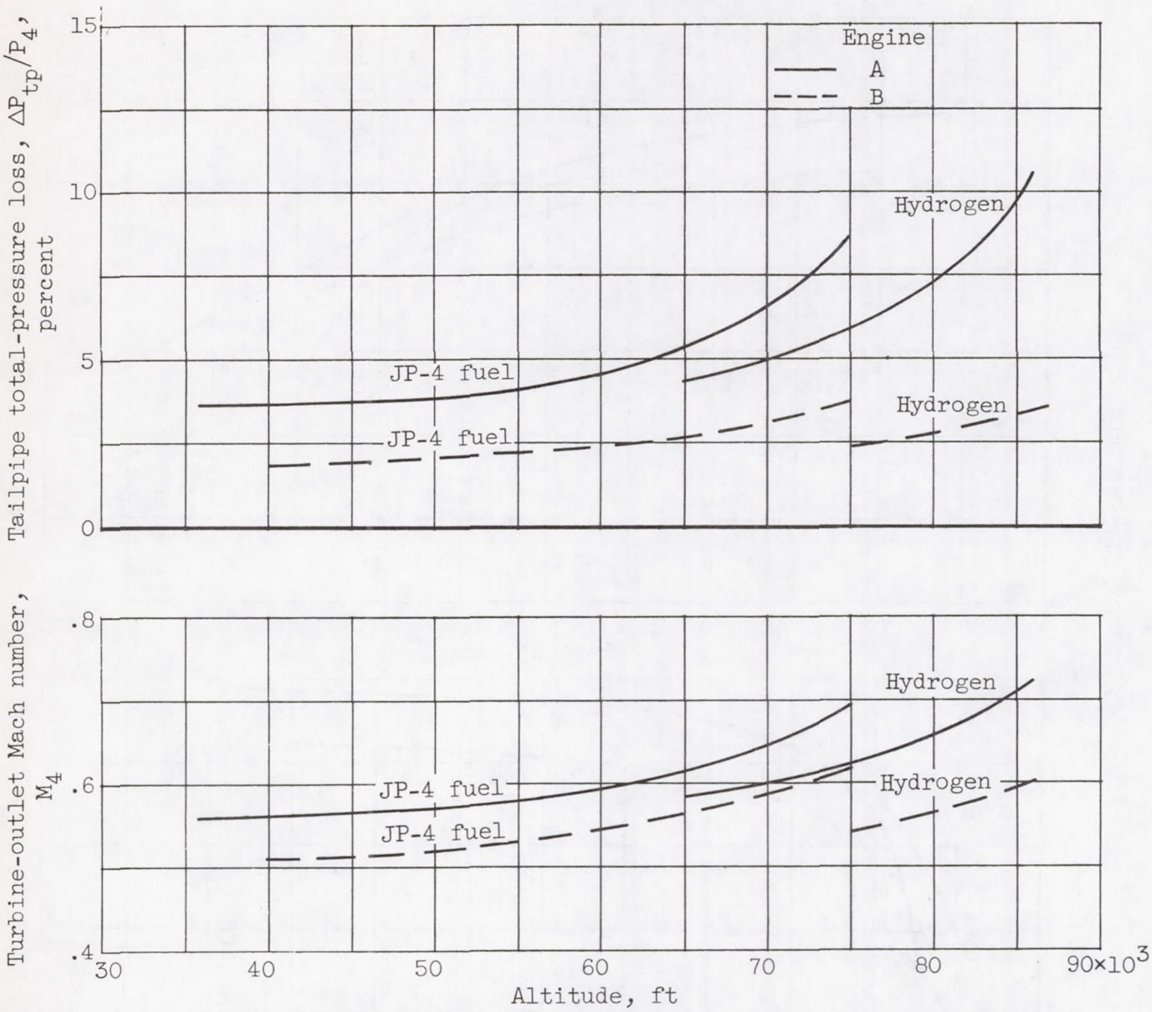


Figure 16. - Effect of altitude on tailpipe losses and turbine-outlet Mach number at rated engine speed and exhaust-gas temperature. Flight Mach number, 0.8.

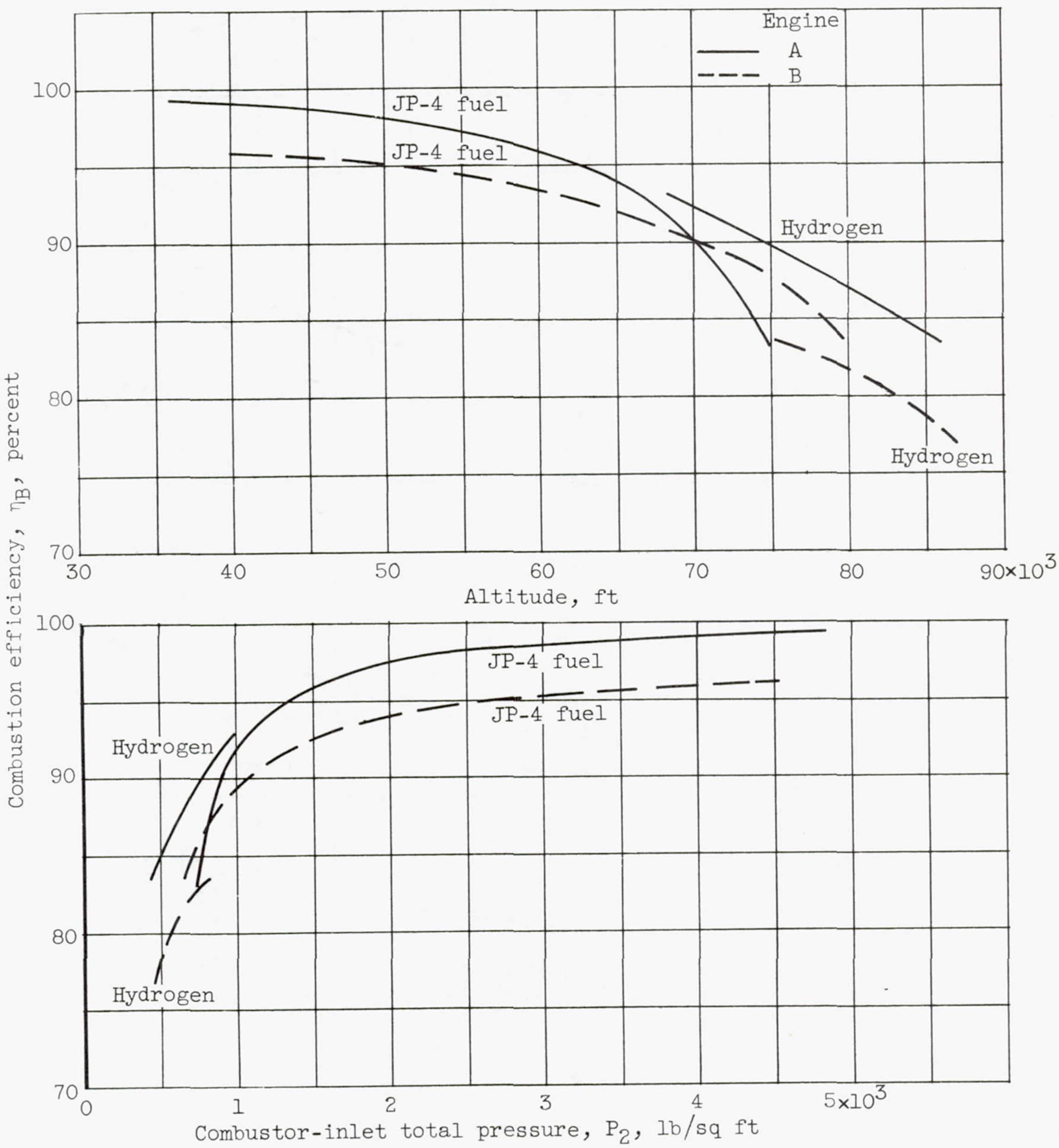


Figure 17. - Variation of combustion efficiency with altitude and combustor-inlet pressure.

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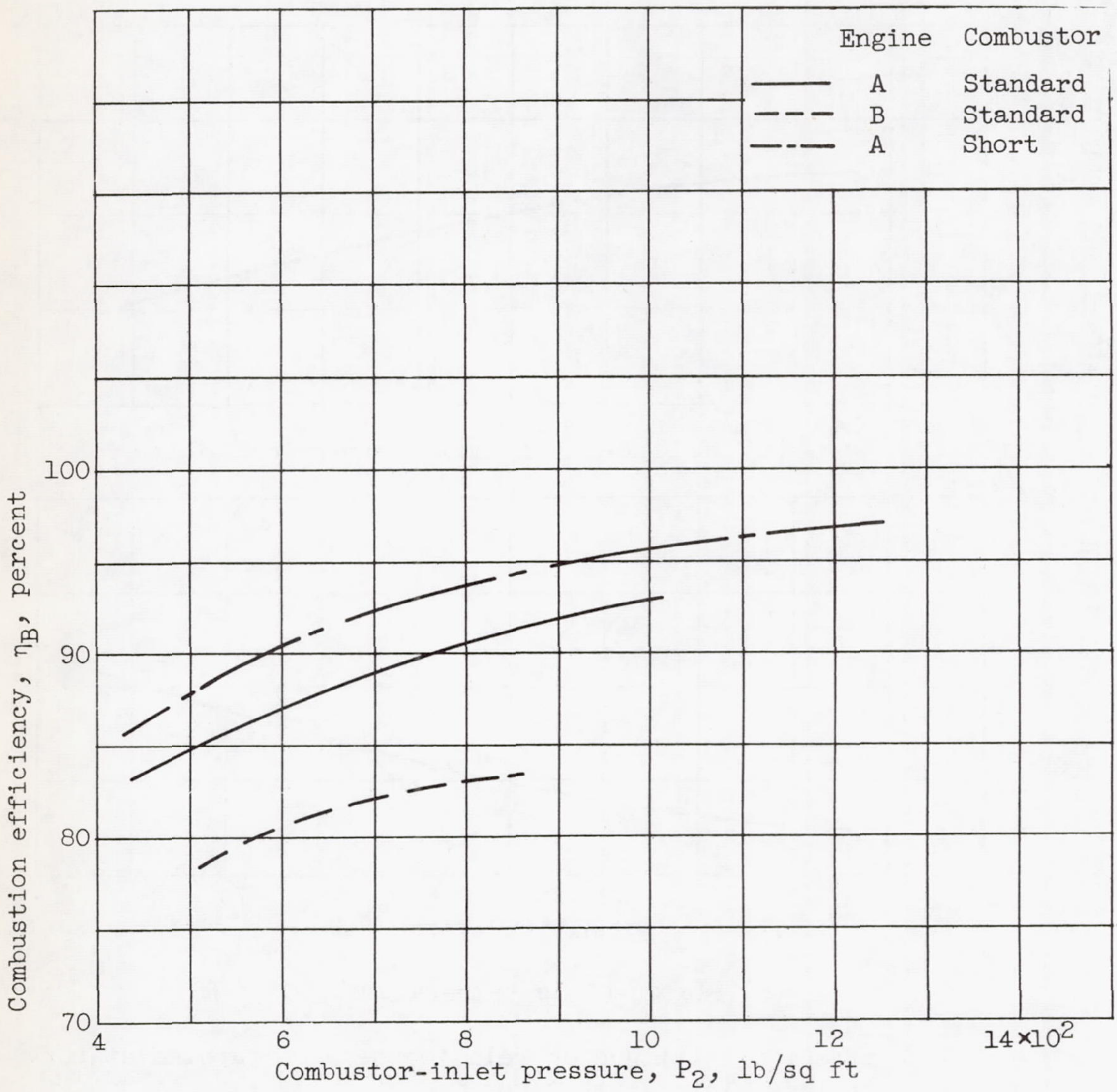
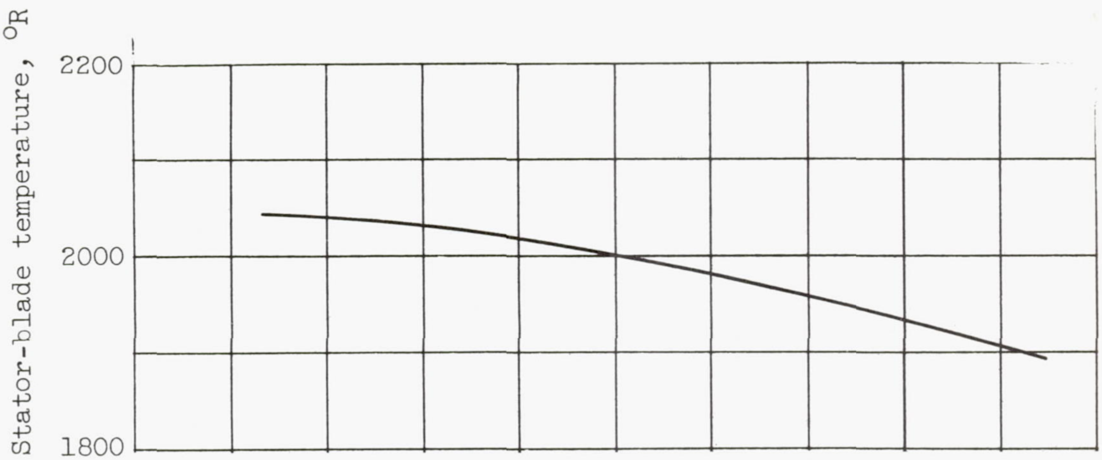
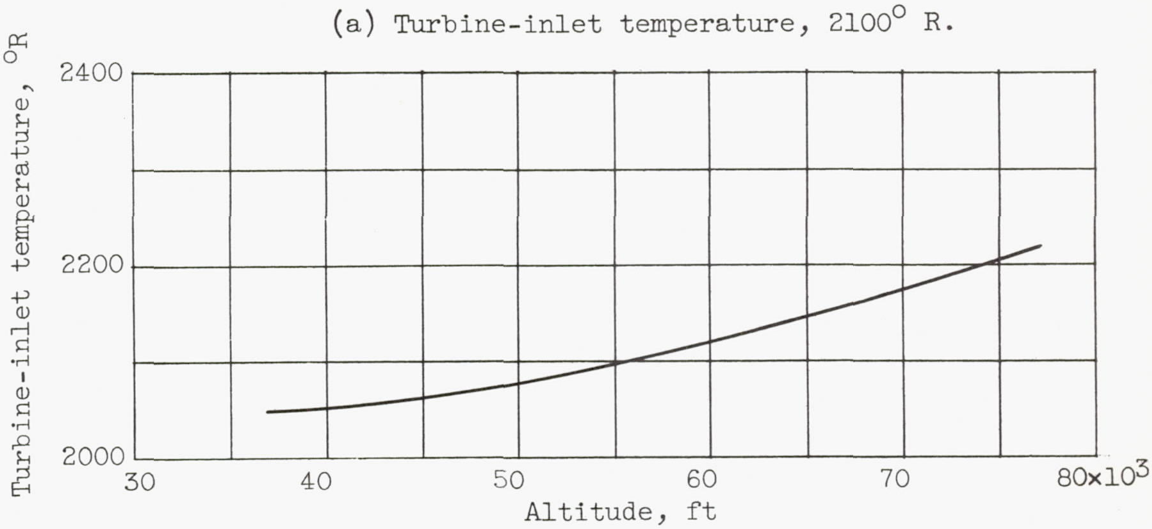


Figure 18. - Comparison of combustion efficiency with gaseous hydrogen in a short combustor designed for hydrogen and in two combustors designed for JP-4 fuel.



(a) Turbine-inlet temperature, 2100° R.



(b) Stator-blade temperature, 2000° R.

Figure 19. - Effect of altitude on relation between turbine stator-blade and turbine-inlet temperatures.