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RESEARCH MEMORANDUM

HIGH-ALTITUDE PERFORMANCE INVESTIGATION OF J65-B-3 TURBOJET ENGINE WITH BOTH JP-4 AND GASEOUS HYDROGEN FUELS

By Harold R. Kaufman

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Cleveland, Ohio

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RESEARCH MEMORANDUM

HIGH-ALTITUDE PERFORMANCE INVESTIGATION OF J65-B-3 TURBOJET ENGINE WITH BOTH JP-4 AND GASEOUS-HYDROGEN FUELS

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SUMMARY

An investigation was conducted to determine the performance of the J65-B-3 turbojet engine with both JP-4 and gaseous-hydrogen fuels. With JP-4 fuel, the altitude range investigated was from about 40,000 to 75,000 feet at a flight Mach number of 0.8. The combustor was then modified slightly to permit the use of gaseous hydrogen, and an altitude range from 65,000 to over 85,000 feet was investigated at a Mach number of 0.8.

With JP-4 fuel, the maximum altitude for stable combustion was from about 60,000 to 65,000 feet, and the ultimate blowout limit was at an altitude of about 75,000 feet. In contrast, the combustion with hydrogen fuel was stable up to the facility altitude limit of 89,000 feet. At rated speed and temperature the thrust with hydrogen fuel was 2 to 4 percent higher and the specific fuel consumption 60 to 70 percent lower than with JP-4 fuel.

The steady-state performance with either fuel decreased considerably with increasing altitude. Rated speed and temperature operation with JP-4 fuel resulted in a 12-percent drop in corrected net thrust with an increase in altitude from 40,000 to 75,000 feet. The specific fuel consumption increased 33 percent for the same altitude range.

INTRODUCTION

The analysis of reference 1 indicates the advantages of hydrogen fuel for long-range aircraft at altitudes as high as 80,000 or 90,000 feet. Subsequent to the analysis of reference 1, two experimental turbojet-engine investigations were made to determine the problems associated with modifying an engine for operation with gaseous-hydrogen fuel and to evaluate the relative merits of JP-4 and gaseous-hydrogen fuels, particularly at very high altitudes. One of these two engines was the J65-B-3. Presented herein are detailed maps of the component and over-all engine performance of the J65-B-3 turbojet engine. Pumping characteristics are also included to permit calculation of engine performance over a wide range of flight conditions.

The J65-B-3 data also were used for a generalized study of hydrogen-fueled operation of turbojet engines. This study includes data from two turbojet-engine investigations, in addition to several component investigations, and is reported in reference 2.

Because reference 1 indicates applications of hydrogen-fueled aircraft at altitudes as high as 80,000 or 90,000 feet, the altitude range 4303
of this investigation was extended up to either the engine operating limits or the test-facility limits. With JP-4 fuel, the engine was operated at altitudes from about 35,000 to 75,000 feet at a flight Mach number of 0.8. The combustors were then modified slightly and, with gaseous hydrogen as the fuel, the engine was operated at altitudes from about 65,000 feet to the facility limit of about 85,000 feet, also at a Mach number of 0.8.

APPARATUS AND PROCEDURE

The J65-B-3 turbojet engine has a 13-stage axial-flow compressor, an annular prevaporizing-type combustor, and a two-stage turbine. At rated conditions the engine speed is 8300 rpm, the exhaust-gas temperature is 1626° R, and the compressor pressure ratio is about 6.9. The rated sea-level static thrust is 7220 pounds with a specific fuel consumption of 0.92 pound per hour per pound of net thrust.

In the standard JP-4 combustor configuration, the fuel is injected into the upstream end of the vaporizing tubes. There are openings around the nozzles so that air can also enter with the fuel to help the vaporizing process. When the combustor was modified for gaseous hydrogen (fig. 1), the JP-4 injectors were removed and larger open-end tubes were inserted in their place to inject the hydrogen. To prevent possible destruction of the vaporizing tubes due to internal combustion, the upstream openings were closed so that only hydrogen flowed through the vaporizing tubes.

The engine was installed in the 20-foot-diameter test section of the altitude wind tunnel at the NACA Lewis laboratory. Two features of the installation permitted simulation of altitudes considerably above those obtained in previous turbojet investigations. The usual turbojet installations in this facility are supplied with air ducted from an external source, so that the exhausters must remove engine airflow in addition to tunnel leakage. In this installation, the engine drew air from the tunnel so that only enough exhauster capacity was required to handle tunnel leakage flows. Thus, at the low exhauster flow capacity required to handle the tunnel leakage air, lower tunnel pressures were attainable than when the engine airflow came from an external source.

The second feature was an exhaust diffuser attached directly to the turbine-outlet flange. In previous investigations, exhaust nozzles were used and the tunnel pressure was regulated to provide sonic flow at the exhaust-nozzle exit. Thus, the tunnel pressure was one-half or less of the turbine-outlet total pressure. With an exhaust diffuser, however, the tunnel pressure need be only slightly less than the turbine-outlet total pressure. A more detailed description of this diffuser technique can be found in reference 2. The combined effect of these two features was to increase the altitude limit of the facility by 25,000 to 30,000 feet. A sketch of the engine and exhaust diffuser, together with an inlet throttle valve, is shown in figure 2. The inlet throttle valve and the diffuser butterfly valve were required for pressure regulation at the engine inlet and outlet.

With the diffuser installation, the thrust force due to the engine alone could not be measured directly. The total-pressure losses in the diffuser (ahead of the butterfly valve) were measured and found to agree closely with the values obtained in conventional tailpipe configurations. The diffuser losses were, therefore, subtracted from the turbine-outlet pressure to obtain a calculated exhaust-nozzle-inlet pressure. This pressure, together with the altitude static pressure and a typical convergent-nozzle effective-velocity coefficient of 0.975, was used to calculate thrust. Thus, the altitude static pressure was not experimentally simulated and appeared only in the calculation procedure.

The simulation of a flight condition was accomplished by regulating the tunnel temperature and throttling the inlet airflow so that the pressure and temperature at the engine inlet corresponded to altitude ram conditions. For each altitude, data were obtained over a range of exhaust-gas temperatures for each of several engine speeds. Fuel control was manual for all steady-state data. Fuel steps were used to determine the compressor stall limits.

The locations of the instrumentation stations are shown in figure 2. Also shown in figure 2 is a table of the temperature and pressure instrumentation at each station.

Because of the very low tailpipe pressures at the higher altitudes investigated, radiation corrections to the thermocouple readings were necessary. Descriptions of the shielded thermocouples used at station 9, together with discussions of the nature and magnitude of the corrections, can be found in references 2 and 3.

RESULTS AND DISCUSSION

Engine Performance

The altitude operating limits at a flight Mach number of 0.8 with both gaseous hydrogen and JP-4 fuels are presented in figure 3. Compressor stall, combustor blowout, exhaust-gas temperature, and facility limits are shown.

The stall limit shown in figure 3 results from the intersection of 4303
the steady-state operating line with the compressor stall limit. The speed at which they intersect increased with increasing altitude until, at the higher altitudes, only a small speed range was available for steady-state operation.

Operation with JP-4 fuel at altitudes above about 60,000 to 65,000 feet resulted in unstable and erratic combustion, so that an operational limit should be considered to exist near these altitudes rather than at the ultimate blowout limit of about 75,000 feet. With hydrogen as the fuel, the combustion was stable and the throttle could be manipulated rapidly without causing blowout. The blowout limit with hydrogen was not obtained below the facility altitude limit of 89,000 feet.

The difference between the temperature limits for fixed exhaust-nozzle operation with the two fuels resulted from the difference in exhaust-gas properties. The higher gas constant and specific heat with hydrogen permitted the engine to operate at a lower exhaust-gas temperature for the same engine speed and exhaust-nozzle area. Or, conversely, the same exhaust-gas temperature was obtained at a higher engine speed for the same exhaust-nozzle area. The reasons for this shift in engine operating point are discussed in the Turbine section.

Rated engine speed, of course, represents a structural limit. With JP-4 fuel, the exhaust nozzle was sized to obtain rated exhaust-gas temperature and rated engine speed at an altitude of 40,000 feet. This reference altitude was used because the Reynolds number effects are small at altitudes up to 40,000 feet, and the results would be confused by changes in corrected speed below the tropopause.

The performance maps at several altitudes with JP-4 and hydrogen fuels are shown in figures 4 and 5. Although engine speed and exhaust-gas temperature are given in uncorrected values, the thrust was left in corrected form to facilitate comparisons at different altitudes. The conversion constant to obtain uncorrected thrust is shown in each figure. The point of best specific fuel consumption is at or near an engine speed of 7700 rpm at all altitudes. Performance maps are not presented for the highest altitudes investigated with each fuel because extremely limited operation was obtained.

The variations of corrected net thrust, specific fuel consumption, and exhaust-nozzle area with altitude for two modes of operation at rated exhaust-gas temperature are shown in figure 6. With both fuels and both modes of operation, the corrected thrust decreased with increasing altitude.

The difference in thrust with the two fuels (2 to 4 percent at rated speed and temperature) is again a result of the differences in gas properties. Because the specific heat of the gas in the turbine is higher with hydrogen fuel than with JP-4 fuel, the turbine pressure ratio is reduced for the same work output. The engine, therefore, operated at a higher total-pressure ratio and thus higher thrust with the hydrogen fuel. There was also an effect in the expansion process through the exhaust nozzle. That is, for a given exhaust-nozzle pressure ratio, the jet velocity is proportional to the square root of the gas constant. The increase in thrust with hydrogen was, therefore, a result of both the higher engine pressure ratio and the higher gas constant at the exhaust nozzle.

The reduced thrust with fixed-area exhaust-nozzle operation compared with rated-speed operation (fig. 6) was indirectly a result of the speed reductions necessary to avoid excessive temperatures (fig. 3). The speed reductions reduced the airflow, which, in turn, reduced the thrust.

The specific fuel consumption with both fuels (fig. 6) increased with increasing altitude. The 60- to 70-percent decrease in specific fuel consumption with hydrogen fuel was primarily a result of the higher heating value of hydrogen fuel. Some of this difference in specific fuel consumption, however, was a result of the higher combustion efficiency with hydrogen fuel than with JP-4 fuel. That is, although the combustion efficiency with both fuels decreased with increasing altitude, at any given altitude the combustion efficiency with hydrogen fuel was higher than with JP-4 fuel.

The general trends of decreasing corrected thrust and increasing specific fuel consumption are, of course, altitude effects and can be traced to reductions in component performance. A breakdown of the loss contribution by each of the components is shown in figure 7 for rated speed and temperature operation with JP-4 fuel. The largest contribution to the thrust loss at high altitude is made by the compressor, while the most important component for specific-fuel-consumption increase is the combustor. The other components contributed about equally to the losses in thrust and increases in specific fuel consumption. At 75,000 feet, the thrust had decreased 12 percent and the specific fuel consumption had increased 33 percent as compared with the 40,000-foot reference values. The loss breakdown would be essentially the same with hydrogen fuel, with the exception of the effect of combustion efficiency on specific fuel consumption. The decrease of combustion efficiency with increasing altitude was smaller with hydrogen fuel than with JP-4 fuel. Hence, the increase in specific fuel consumption from this cause would be slightly less with hydrogen fuel than with JP-4 fuel.

Component Performance

Most of the engine operational limits shown in figure 3 are associated with component limits. In a similar manner, the decreases in engine performance at high altitude (fig. 6) are associated with component performance reductions, as shown in figure 7. As an aid to understanding the over-all engine performance and operational limits, the component performance and limits are presented in this section. 4303

<u>Compressor</u>. - Compressor performance maps at several flight conditions are shown in figure 8. At a given pressure ratio and corrected engine speed, the efficiency and corrected airflow decrease with decreasing Reynolds number. A cross plot from these maps at rated corrected engine speed and a compressor pressure ratio of 6.75 is shown in figure 9. The compressor efficiency drops about 10 points as altitude is increased from 36,000 to 86,000 feet. The reduction in corrected airflow was about 11 percent for the same altitude range. Although the magnitude of the Reynolds number effects at other speeds is slightly different, the general trends are the same.

The compressor stall limits are shown in figure 10 as functions of corrected engine speed. Because a fixed-area exhaust nozzle was not used, steady-state operation is not defined by a single line for each altitude. Instead, the approximate region of steady-stage operation is shown by the shaded region. The steady-state compressor pressure ratio was essentially independent of altitude and fuel type at any given corrected engine speed and engine temperature ratio. The stall limit, however, decreased with increasing altitude, with the result that the margin between the steady-state region and the stall limit was considerably less at an altitude of 80,000 feet than at 40,000 feet. At rated corrected speed (8300 rpm), the 80,000-foot margin was less than half of the 40,000-foot value. The engine speed at which the stall line intersects the steady-state region increases with increasing altitude. This effect is also shown by the stall limit in figure 3.

<u>Combustor</u>. - The combustor efficiency is shown in figure 11 as a function of the combustor parameter $w_a T_4$. The combustor parameter is derived from the basic combustor parameter PT/V. As is usually the case with combustor data taken at high altitudes, the degree of correlation is poor. That is, the erratic nature of the combustion process at high altitude and the low efficiency prevent good reproducibility.

The combustion efficiency with both fuels decreased with increasing altitude (decreasing $w_a T_4$). The combustion efficiency with hydrogen fuel, with just the simple combustor modifications, was from 2 to about 10 percent higher than with JP-4 fuel.

Turbine. - The turbine maps are presented in figure 12 for several flight conditions. A range of compressor pressure ratios (turbine-inlet pressures) and turbine-inlet temperatures was obtained at each flight condition so that each map represents a range of turbine-inlet Reynolds number indices.

4305

The corrected gas flow was approximately constant at each flight condition and decreased 6 percent with an increase in altitude from 36,000 to 81,000 feet. The maximum efficiency dropped 9 points over this same altitude range. It should be noted, however, that for a given corrected turbine speed the range of enthalpy drop and turbine pressure ratio shifted to higher values at higher altitudes. Thus, a direct comparison of high- and low-altitude performances at the same corrected turbine speed and pressure ratio cannot be made. The shift of turbine pressure ratio and enthalpy drop with altitude is a result of the decrease in compressor performance at high altitudes, so that the corrected turbine work requirements increase with increasing altitude.

This shift in turbine operating point is shown in figure 13. The work required from the turbine to drive the compressor increases with increasing altitude, while the limiting-loading work decreases. If the blowout limit of the combustor with JP-4 fuel had not prevented operation above an altitude of about 75,000 feet, the limiting-loading line would have been reached with JP-4 fuel at an altitude of about 80,000 feet. Reduced engine speed would then have been necessary to avoid excessive temperatures, similar to a fixed-area exhaust-nozzle limit.

With hydrogen as the fuel, the work required to drive the compressor is about the same as with JP-4 fuel at any given altitude. The higher gas constant, however, results in a lower corrected enthalpy drop with hydrogen fuel than with JP-4 fuel. Hence, the limiting-loading line with hydrogen fuel would be reached at an altitude 5000 to 10,000 feet higher than with JP-4 fuel.

The variation of corrected enthalpy drop with altitude is reflected in the turbine pressure ratio and the turbine-outlet Mach number (fig. 13). The operation at a lower corrected enthalpy drop with hydrogen fuel than with JP-4 fuel resulted in a lower turbine-outlet Mach number at the same altitude. The high turbine-outlet Mach numbers obtained as the turbine approached limiting loading (fig. 13) caused the tailpipe total-pressure losses to become more than double the 40,000-foot value.

Engine Pumping Characteristics

The engine pumping characteristics, which are useful for the calculation of engine performance over a wide range of flight conditions, are presented in figures 14 and 15 for JP-4 and hydrogen fuels, respectively. The method of presentation is to provide pumping characteristics for a reference Reynolds number index (part (a)) together with correction curves for other Reynolds number indices (parts (b) and (c)). The highest Reynolds number index that was common to operation with both JP-4 and hydrogen fuels, 0.105, was selected as the reference condition. Actually, the corrected-speed and temperature-ratio operating range shown in the reference plot with JP-4 fuel (fig. 14(a)) is larger than was experimentally obtained at a Reynolds number index of 0.105. If only the operating range at a Reynolds number index of 0.105 had been shown, the range of performance calculation of higher Reynolds number indices would have been severely restricted. To permit a greater range of calculation at higher Reynolds number indices, figure 14(a) was extended with the aid of figures 14(b) and (c) and low-altitude data. Thus, some of the speed-temperature operating range shown in figure 14(a) corresponds to actual operating points only at high Reynolds number indices and may be in a compressor stall or turbine limiting-loading condition at a Reynolds number index of 0.105. The reference curve for hydrogen fuel was at the highest Reynolds number index investigated with hydrogen fuel; so a similar extension of the operating range was not necessary with figure 15(a). Of course, operating limits such as compressor stall, combustor blowout, and turbine limiting loading must be considered at Reynolds number indices other than 0.105.

The correction factors K_P and K_a are used as follows:

$$(P_4/P_1)_{\delta/\varphi\sqrt{\theta}} = (P_4/P_1)_{0.105} K_{P,\delta/\varphi\sqrt{\theta}}$$

$$(w_a\sqrt{\theta}/\delta)_{\delta/\varphi\sqrt{\theta}} = (w_a\sqrt{\theta}/\delta)_{0.105} K_{a,\delta/\varphi\sqrt{\theta}}$$

where $\delta/\varphi\sqrt{\theta}$ is the desired Reynolds number index. (Symbols are defined in the appendix.)

It should be noted that the airflow correction factor K_a of figure 15(c) is the same as that of figure 14(c) in the range where the two overlap. This similarity results from the fact that the compressor performance and operating point are essentially independent of the fuel (JP-4 or hydrogen). The pressure-ratio correction factor, however, is not the same for JP-4 fuel (fig. 14(b)) as it is for hydrogen fuel (fig. 15(b)). This difference might be expected from the difference in turbine operating point for the two fuels.

The compressor, combustor, and turbine limits must be considered to determine the operational limits within which performance can be predicted with the pumping characteristics of figures 14 and 15. The compressor stall limit is shown in figure 16. For a given Reynolds number index, performance calculations should be attempted only at corrected engine speeds above the stall value. For combustor blowout with JP-4 fuel, of course, the lowest values of w_aT_4 on figure 11 should be used. The stable combustion altitude with JP-4 fuel was considerably lower than the ultimate blowout limit. For this stable combustion lim-
4303 it, a value of 15,000 should be used for w_aT_4. The blowout limit with hydrogen fuel was never reached. The lowest values of w_aT_4 for hydrogen fuel in figure 3 represent the facility limit instead.

For a turbine limit, the gas-flow parameter at the turbine outlet $w_g\sqrt{T_4}/P_4$ should not exceed 11.3. This limit is not quite limiting loading. That is, at higher values of the turbine-outlet gas-flow parameter than 11.3, the turbine work still increases with increasing pressure ratio, but the efficiency decreases rapidly. This rapid change of turbine efficiency with turbine operating point in this range pre-
CJ-2 vents accurate prediction of pumping characteristics by the method shown.

The combustor performance from the Component Performance section (fig. 11) can be used in calculating fuel flow. The combustion parameter w_aT_4 is calculated from the turbine-outlet temperature and the airflow. The airflow, of course, can be obtained from figure 14 or 15.

The tailpipe pressure losses are presented in figure 17 to permit calculation of thrust. The turbine limit of 11.3 is also shown in figure 17. The tailpipe losses must, of course, be subtracted from the turbine-outlet total pressure to obtain the exhaust-nozzle-inlet total pressure. As was mentioned in the APPARATUS AND PROCEDURE section, an effective velocity coefficient of 0.975 was used for jet thrust. This value should be satisfactory for any conventional convergent nozzle.

CONCLUDING REMARKS

The use of gaseous hydrogen as a fuel greatly increased the maximum operating altitude of the J65-B-3 turbojet engine. With JP-4 fuel the maximum altitude for stable combustion was from about 60,000 to 65,000 feet, and the ultimate blowout limit was at an altitude of about 75,000 feet. With hydrogen as the fuel, the combustion was stable up to the facility limit of 89,000 feet. The use of hydrogen also reduced the turbine-outlet Mach number, so that a larger range of operation was possible without encountering limiting loading. This difference in turbine-outlet Mach number was most important at high altitudes where the turbine operating point at rated speed and exhaust-gas temperature was close to limiting loading.

The steady-state performance was also improved by the use of gaseous hydrogen as a fuel. The thrust at rated speed and exhaust-gas temperature was 2 to 4 percent higher with hydrogen fuel than with JP-4 fuel. This difference is attributed to the change in turbine operating point (resulting in a higher engine pressure ratio) and the change in gas constant at the exhaust nozzle. The specific fuel consumption was 60 to 70 percent less with hydrogen fuel than with JP-4 fuel. The majority of this decrease was, of course, due to the higher heating value of hydrogen.

The steady-state performance with either fuel decreased considerably with increasing altitude. At rated speed and exhaust-gas temperature, the thrust with JP-4 fuel decreased about 12 percent with an increase in altitude from 40,000 to 75,000 feet. The specific fuel consumption increased 33 percent for the same altitude range. The decrease in compressor performance with increasing altitude caused over half the thrust reduction, while the decrease in combustion efficiency caused the majority of the increase in specific fuel consumption. 4303

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, January 17, 1957

APPENDIX - SYMBOLS

A cross-sectional area, sq ft

F_n net thrust, lb

H total enthalpy, Btu/lb

h altitude, ft

K_a airflow correction factor

K_p pressure-ratio correction factor

N engine speed, rpm

P total pressure, lb/sq ft abs

T total temperature, $^{\circ}R$

V velocity, ft/sec

w_a airflow, lb/sec

w_f fuel flow, lb/hr

w_g gas flow, lb/sec

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

θ_{cr} squared ratio of critical velocity to critical velocity at NACA standard sea-level conditions

φ ratio of absolute viscosity of air or gas to absolute viscosity of air or gas at NACA standard sea-level conditions

Subscripts:

B combustor

C compressor

N exhaust nozzle

T turbine

1 compressor inlet

2 compressor outlet

3 turbine inlet

4 turbine outlet

9 exhaust-nozzle inlet

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2. Fleming, W. A., Kaufman, H. R., Harp, J. L., Jr., and Chelko, L. J.: Turbojet Performance and Operation at High Altitudes with Hydrogen and JP-4 Fuels. NACA RM E56E14, 1956.

3. Glawe, George E., Simmons, Frederick S., and Stickney, Truman M.: Radiation and Recovery Corrections and Time Constants of Several Chromel-Alumel Thermocouple Probes in High-Temperature, High-Velocity Gas Streams. NACA TN 3766, 1956.

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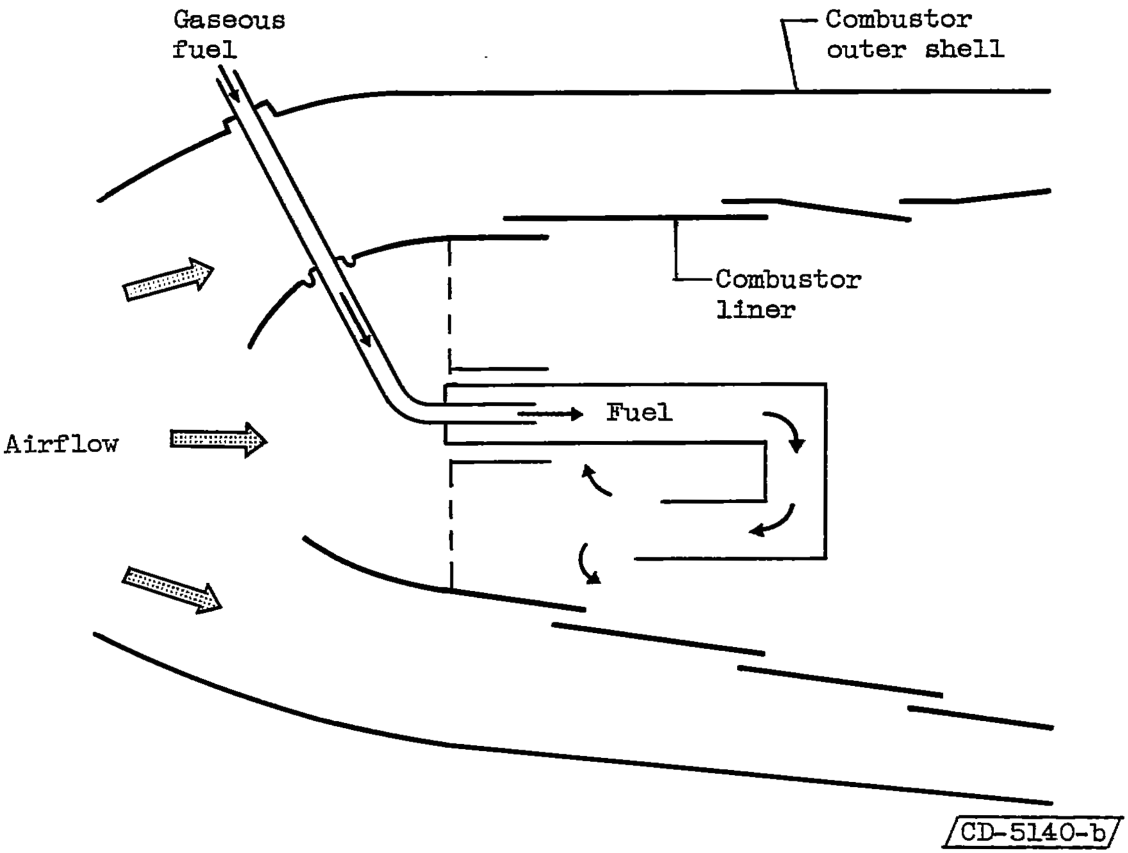
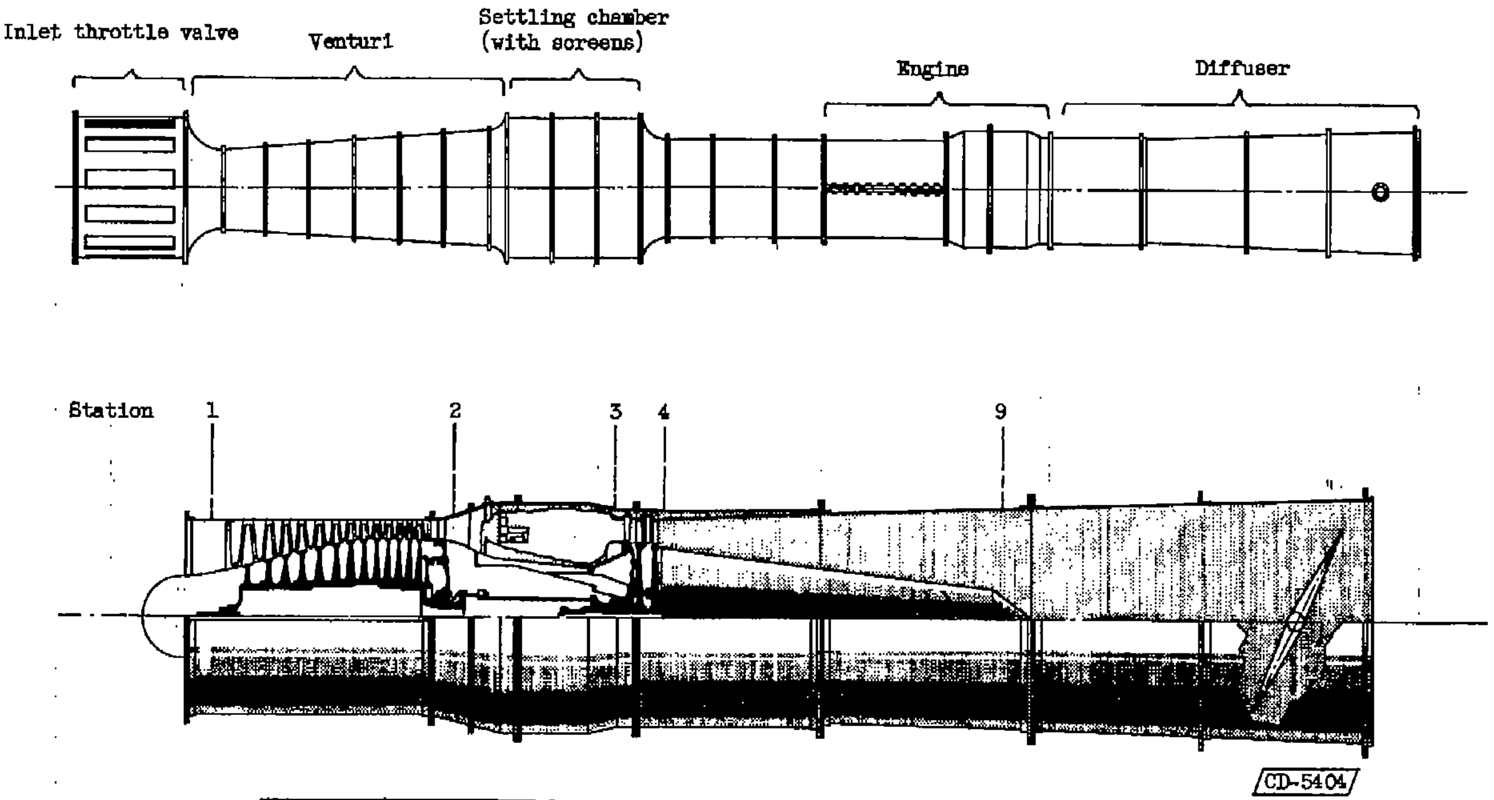


Figure 1. - Sketch of J65-B-3 combustor modified for gaseous hydrogen.



Station	Location	Total pressure	Static pressure	Temperature
1	Compressor inlet	28	16	16
2	Compressor outlet	20	-	16
3	Turbine inlet	24	-	14
4	Turbine outlet	32	-	18
9	Diffuser (equivalent to exhaust-nozzle inlet)	32	-	27

Figure 2. - Sketch of installation and cross section of J65-B-3 turbojet engine showing instrumentation stations.

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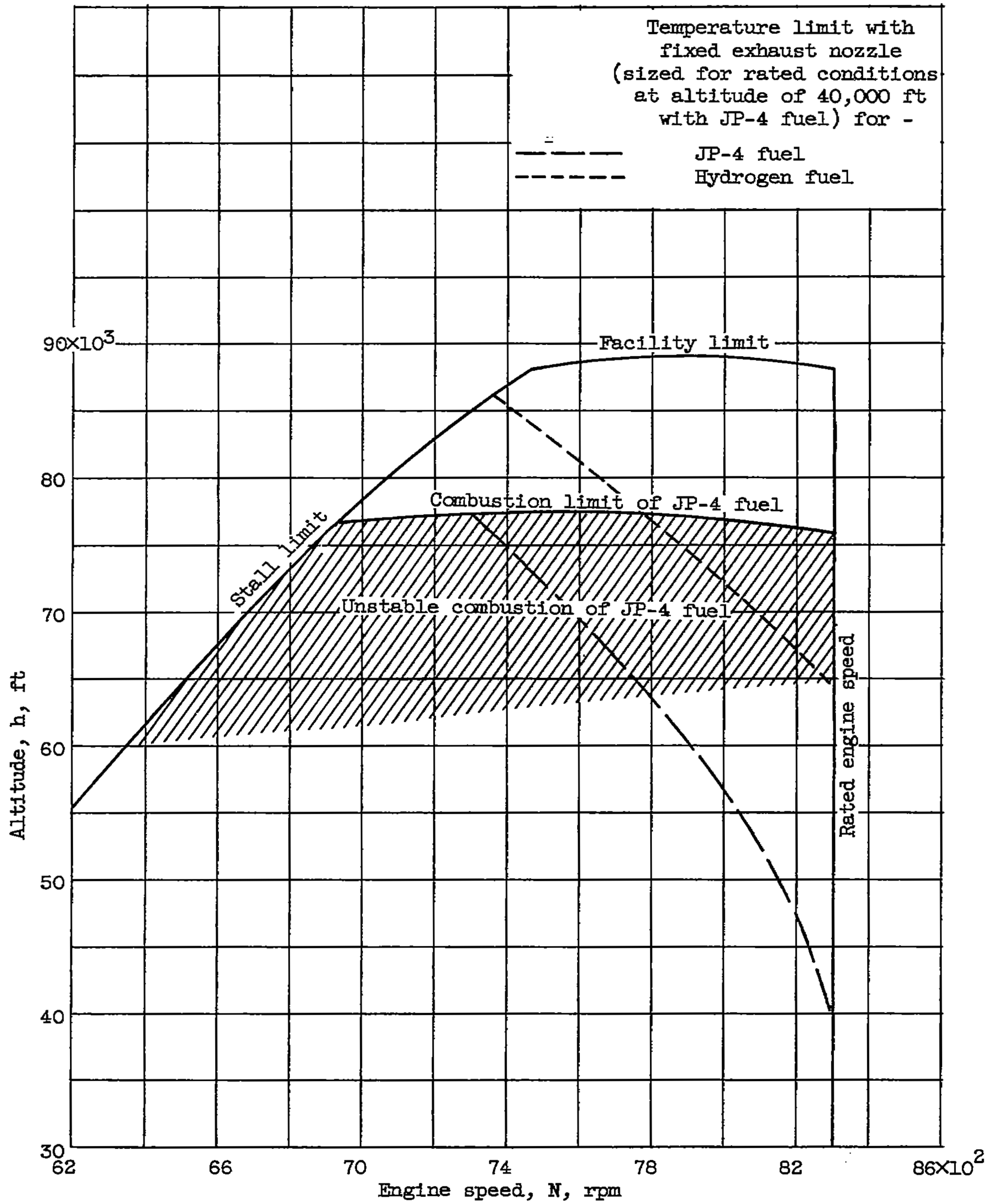

Figure 3. - Operating limits of J65-B-3 turbojet engine at flight Mach number of 0.8.

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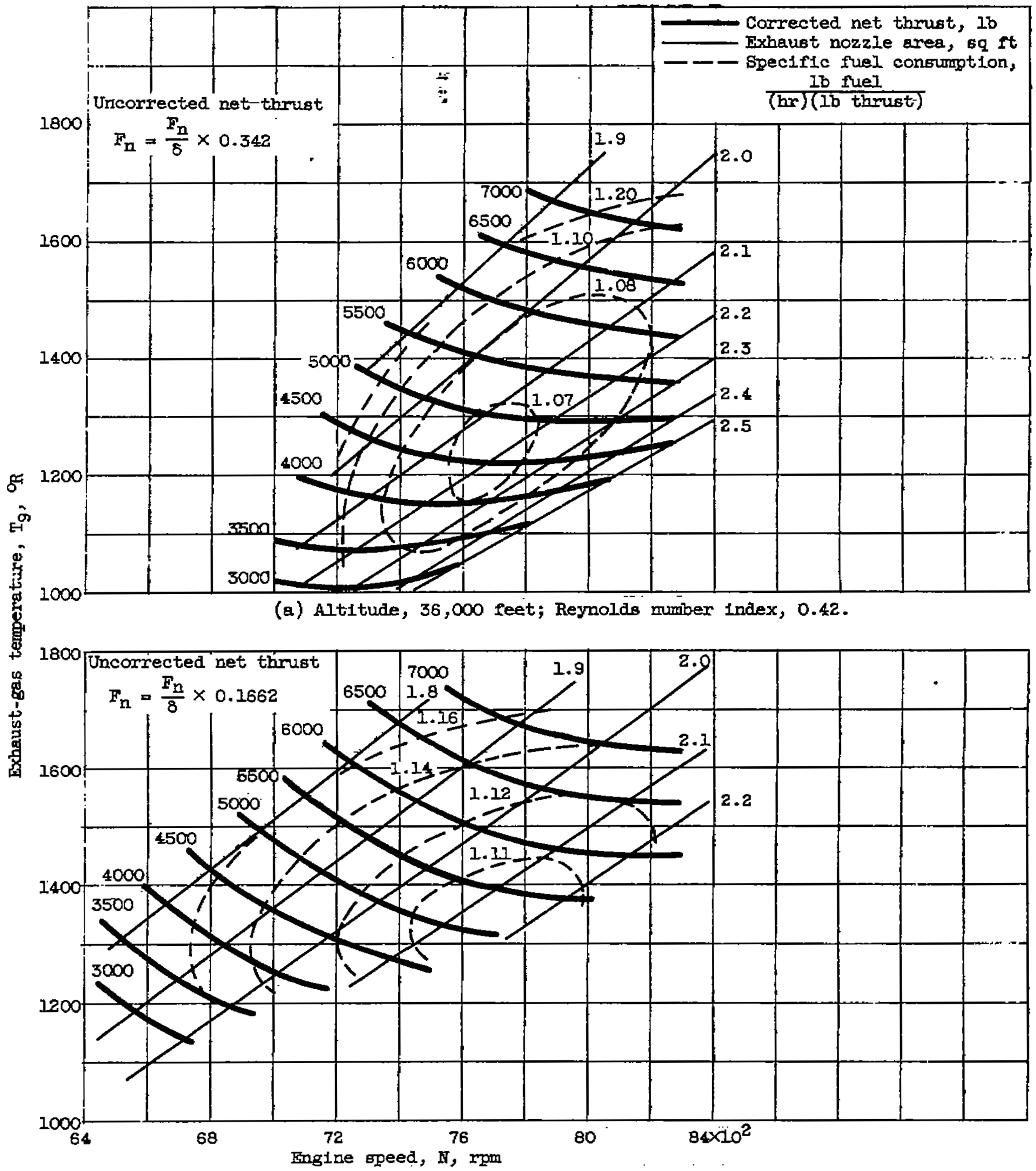

(a) Altitude, 36,000 feet; Reynolds number index, 0.42.

(b) Altitude, 51,000 feet; Reynolds number index, 0.21.

Figure 4. - Over-all engine performance maps obtained with JP-4 fuel. Flight Mach number, 0.8.

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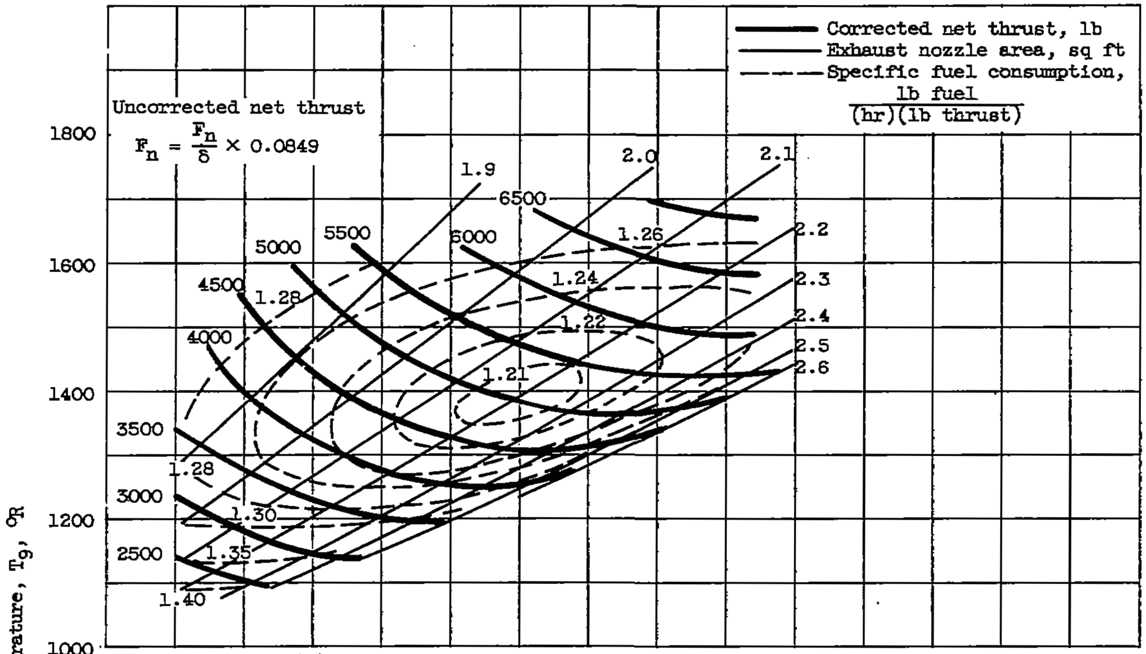
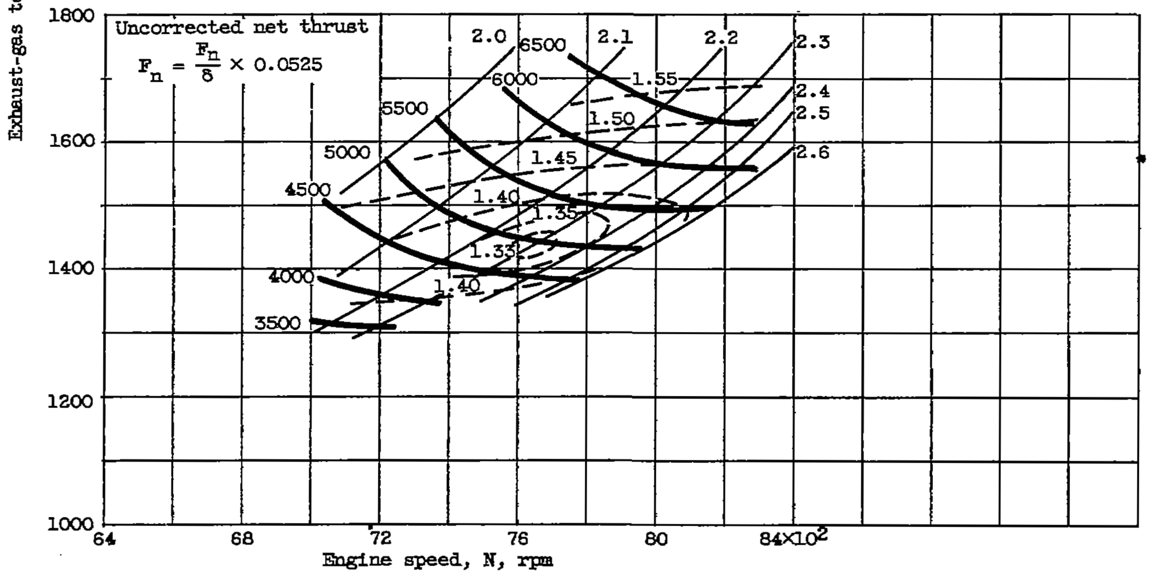


(c) Altitude, 65,000 feet; Reynolds number index, 0.105.



(d) Altitude, 75,000 feet; Reynolds number index, 0.065.

Figure 4. - Concluded. Over-all engine performance maps obtained with JP-4 fuel. Flight Mach number, 0.8.

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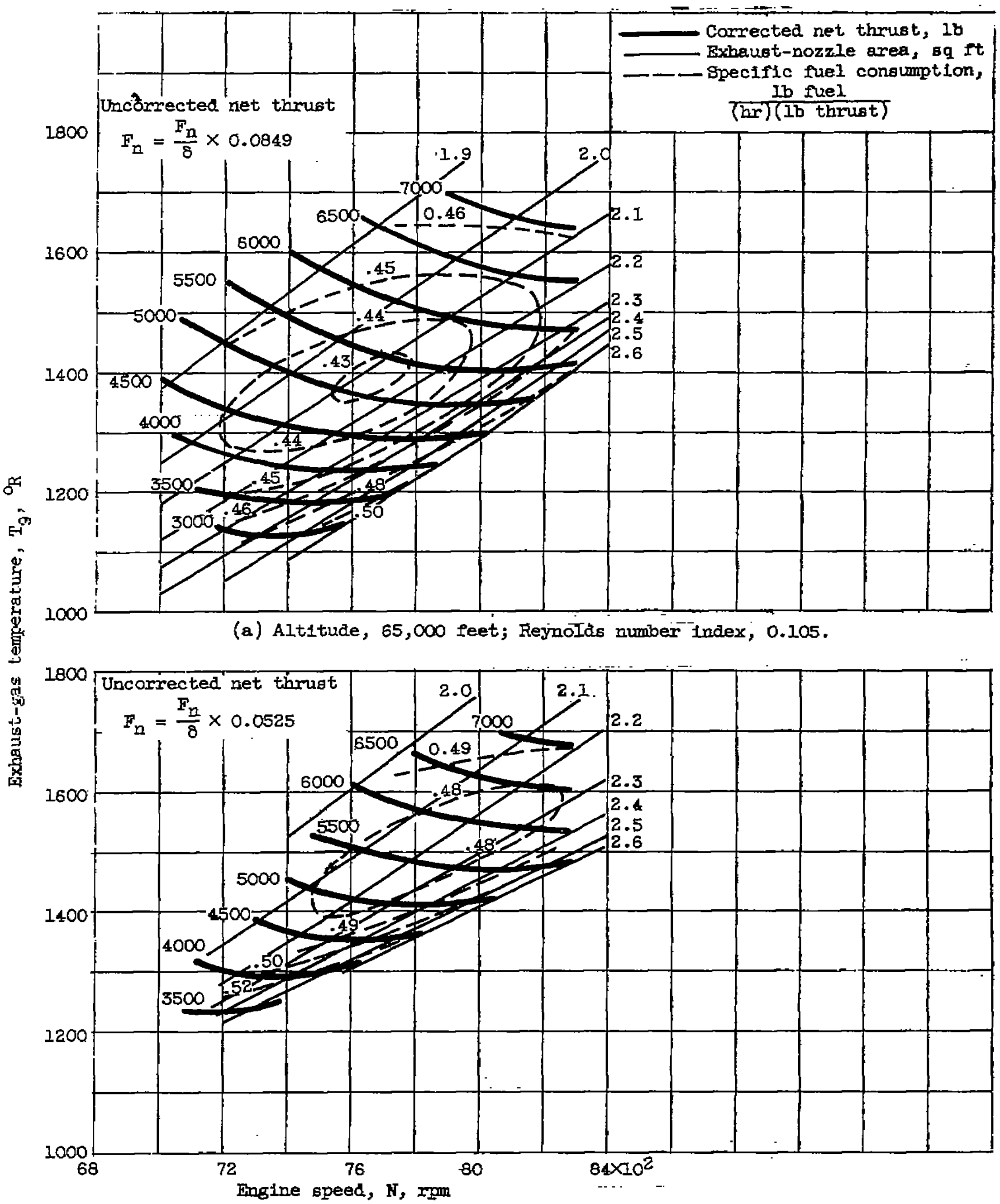

(a) Altitude, 65,000 feet; Reynolds number index, 0.105.

(b) Altitude, 75,000 feet; Reynolds number index, 0.065.

Figure 5. - Over-all engine performance maps obtained with gaseous-hydrogen fuel. Flight Mach number, 0.8.

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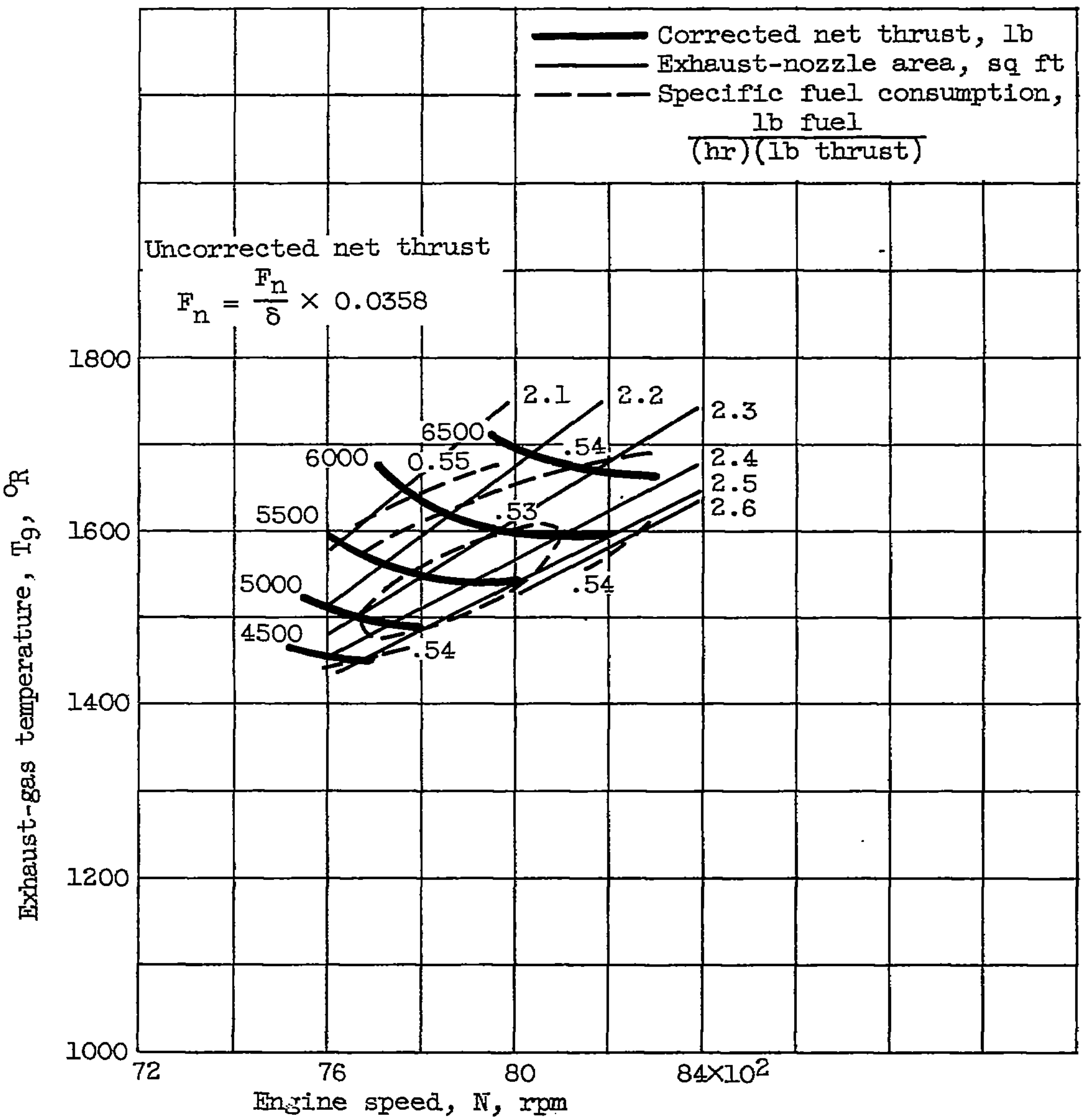

(c) Altitude, 83,000 feet; Reynolds number index, 0.044.

Figure 5. - Concluded. Over-all engine performance maps obtained with gaseous-hydrogen fuel. Flight Mach number, 0.8.

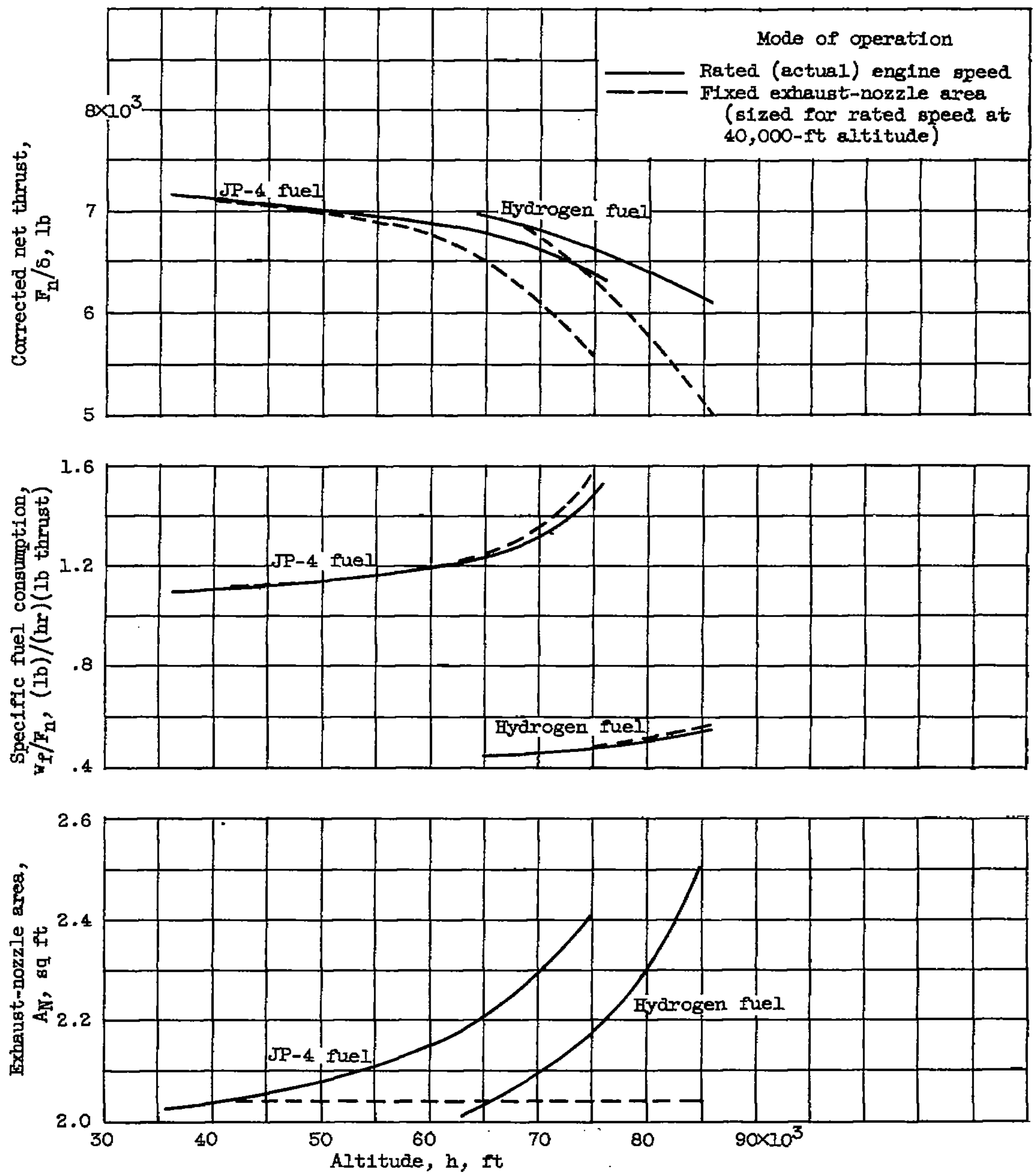

Figure 6. - Effect of altitude for two modes of operation at rated exhaust-gas temperature. Flight Mach number, 0.8.

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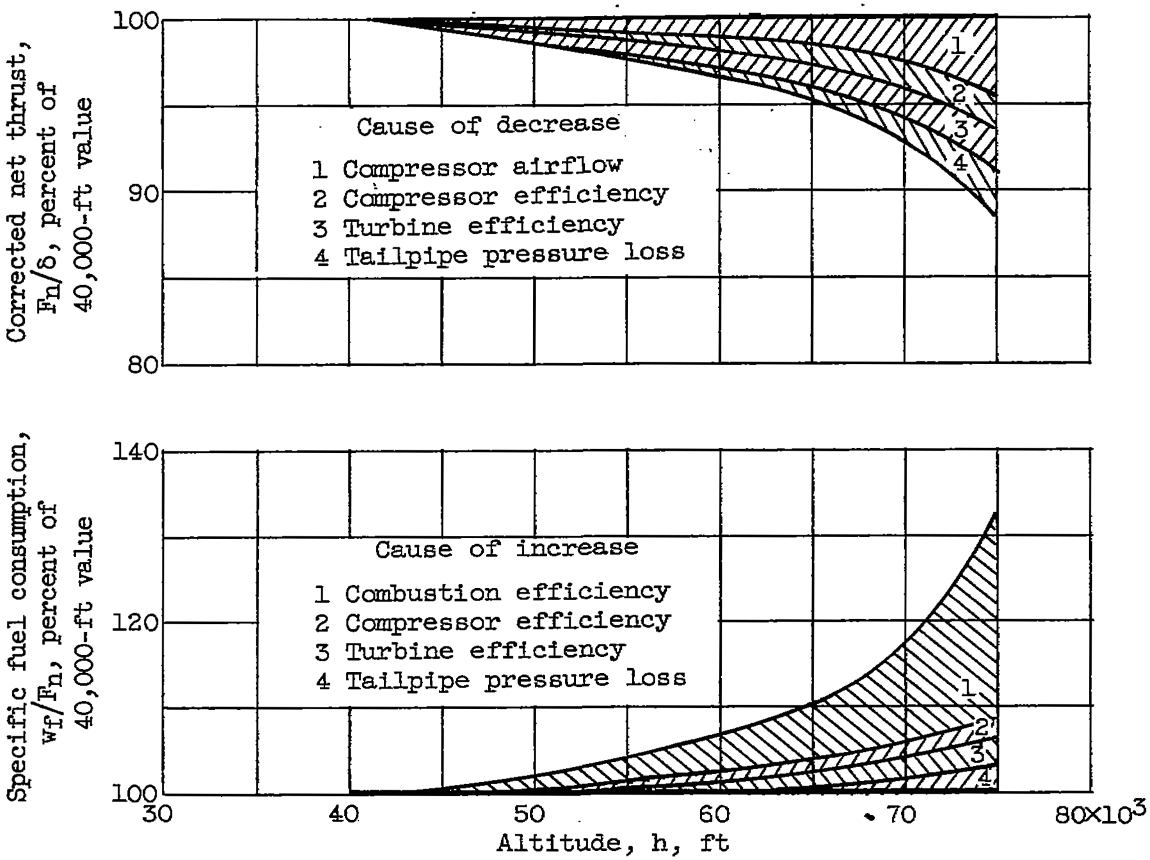

Figure 7. - Contribution of components to altitude performance losses at rated engine speed, rated turbine-outlet temperature, and flight Mach number of 0.8 with JP-4 fuel.

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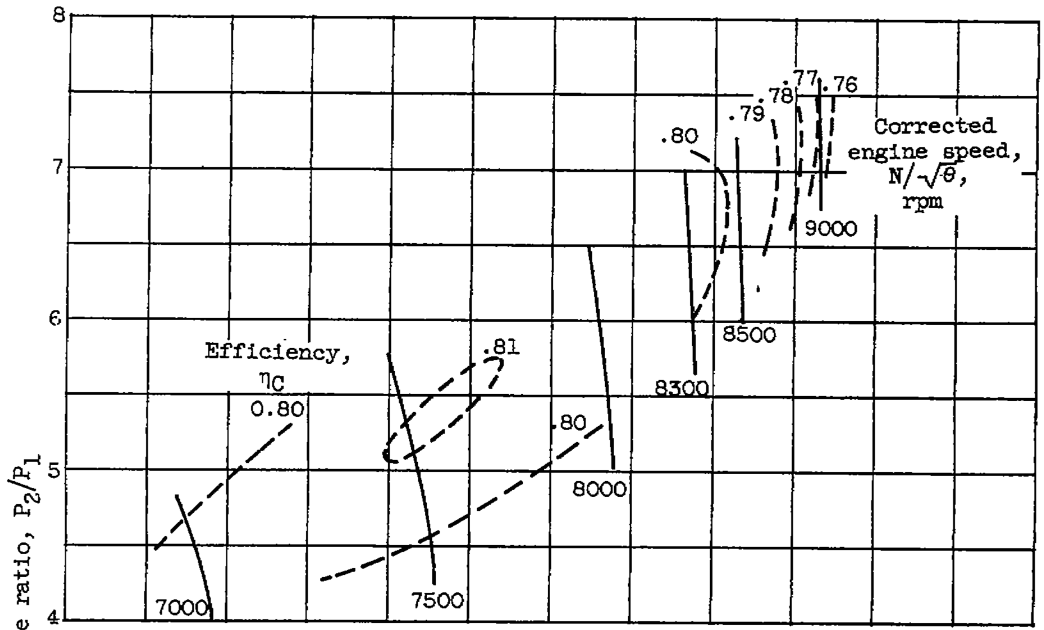
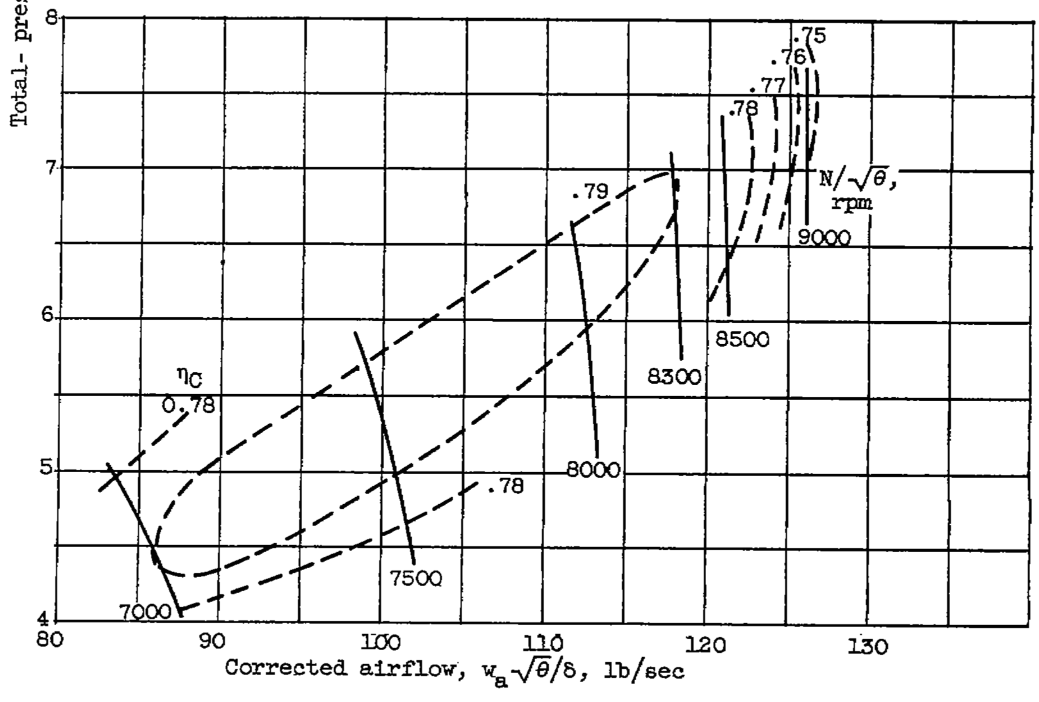


(a) Reynolds number index, 0.4.



(b) Reynolds number index, 0.2.

Figure 8. - Compressor performance maps.

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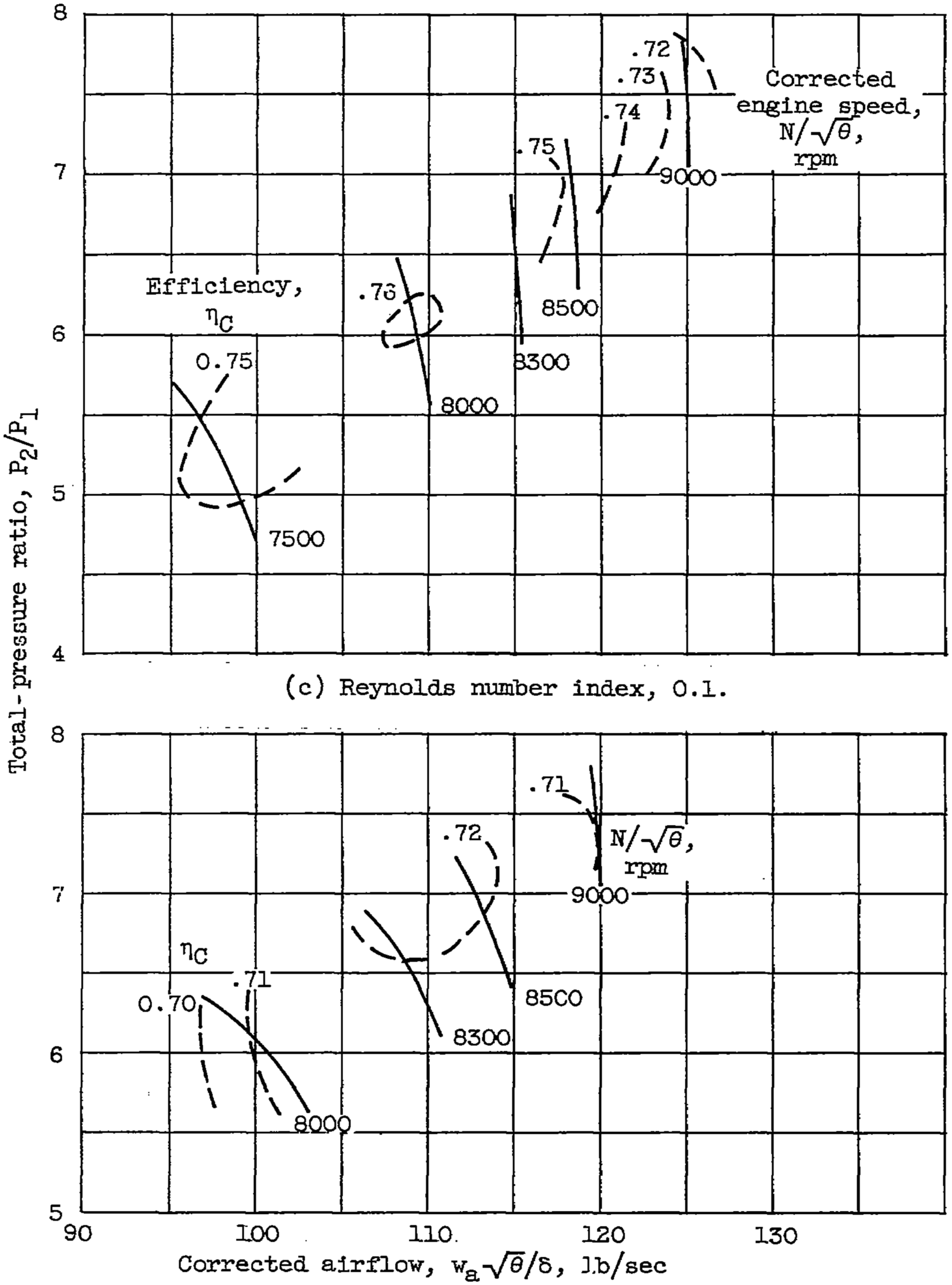

(c) Reynolds number index, 0.1.

(d) Reynolds number index, 0.05.

Figure 8. - Concluded. Compressor performance maps.

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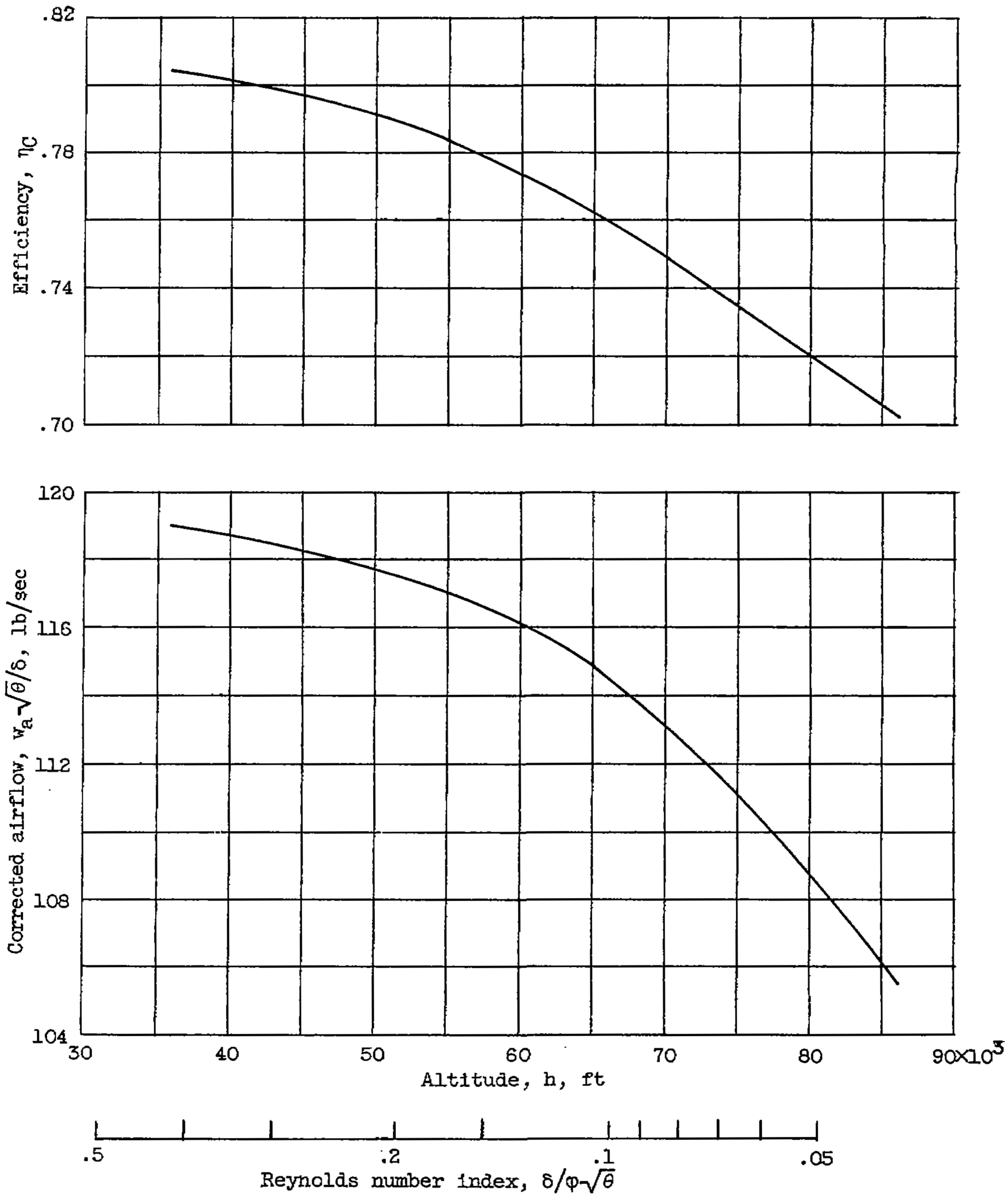


Figure 9. - Effect of flight condition on compressor performance at corrected engine speed of 8300 rpm, pressure ratio of 6.75, and flight Mach number of 0.8.

[redacted]

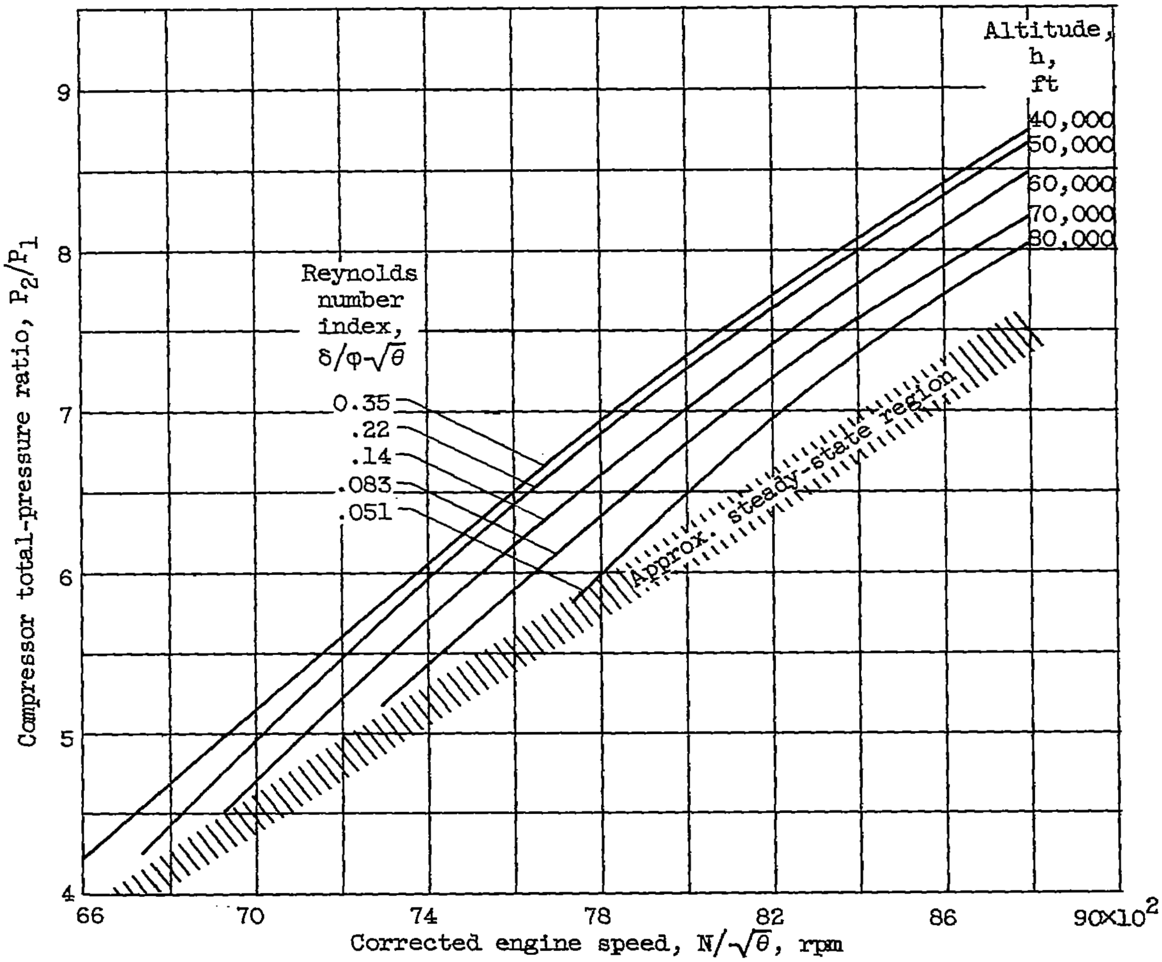


Figure 10. - Effect of flight condition on compressor stall limit. Flight Mach number, 0.8.

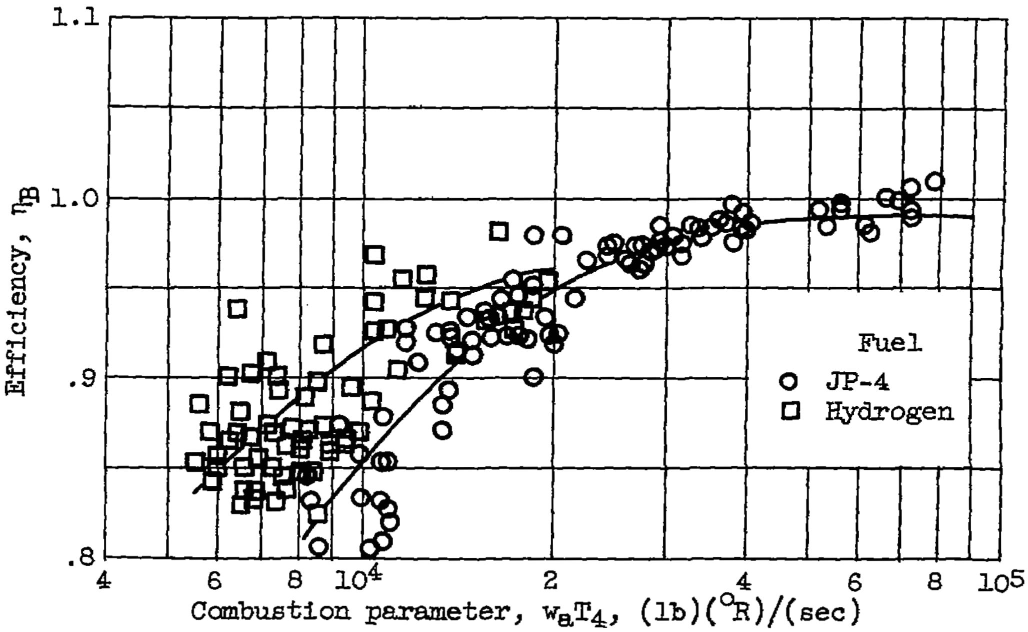
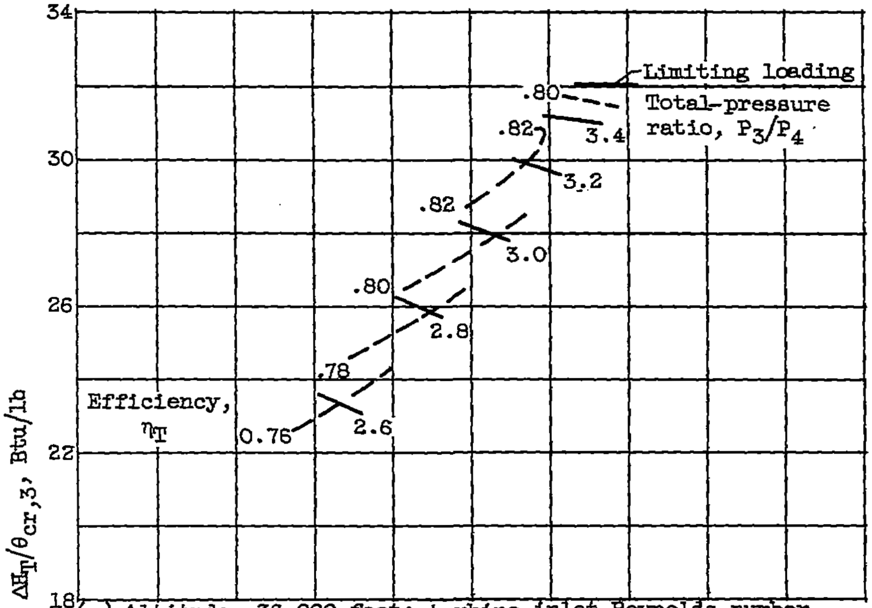
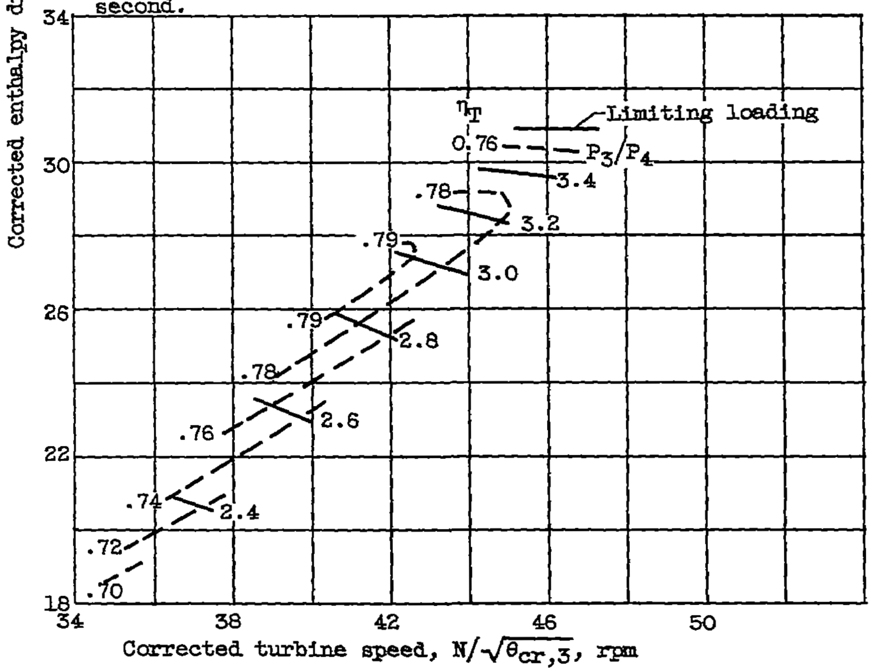


Figure 11. - Combustion efficiency.

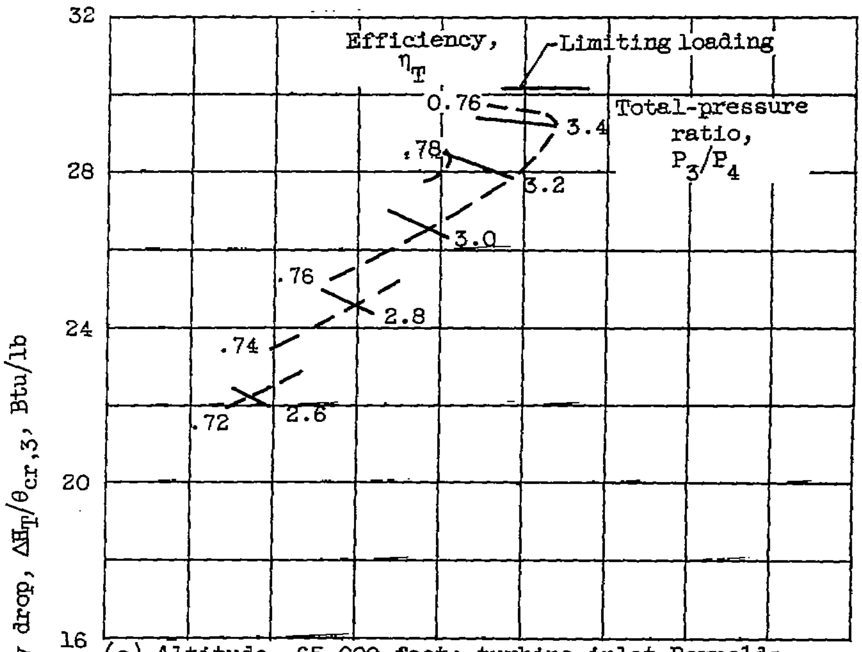


(a) Altitude, 36,000 feet; turbine-inlet Reynolds number index, 0.4 to 0.6; corrected gas flow, 38.0 pounds per second.

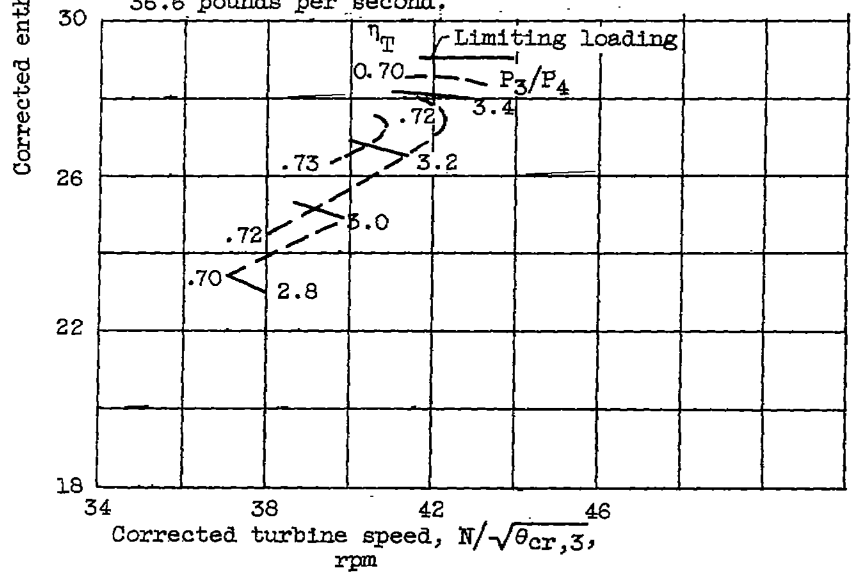


(b) Altitude, 51,000 feet; turbine-inlet Reynolds number index, 0.2 to 0.3; corrected gas flow, 37.3 pounds per second

Figure 12. - Turbine performance maps. Flight Mach number, 0.8.



(c) Altitude, 65,000 feet; turbine-inlet Reynolds number index, 0.10 to 0.15; corrected gas flow, 36.6 pounds per second.



(d) Altitude, 81,000 feet; turbine-inlet Reynolds number index, 0.05 to 0.07; corrected gas flow, 35.7 pounds per second.

Figure 12. - Concluded. Turbine performance maps. Flight Mach number, 0.8.

4303

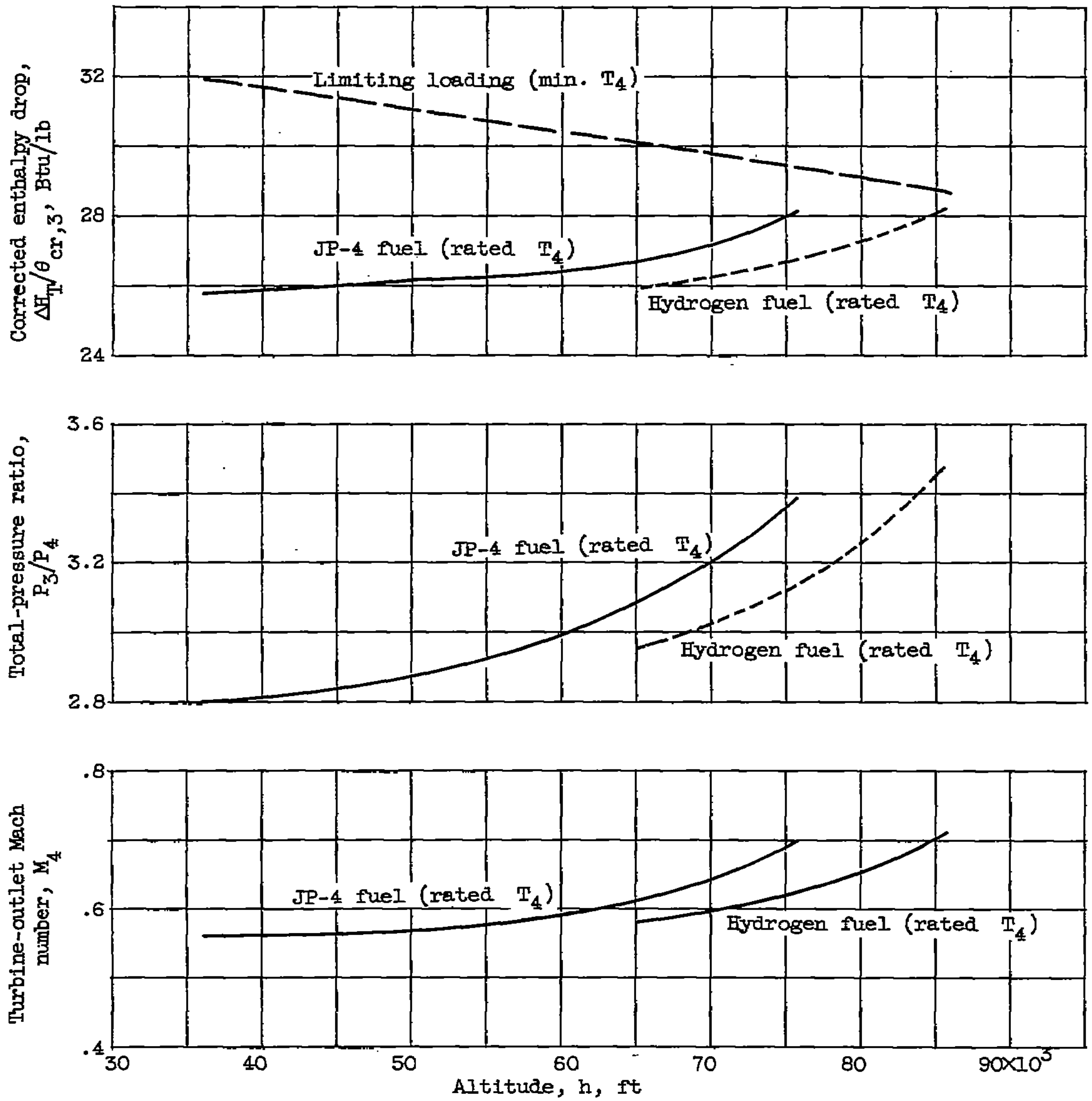
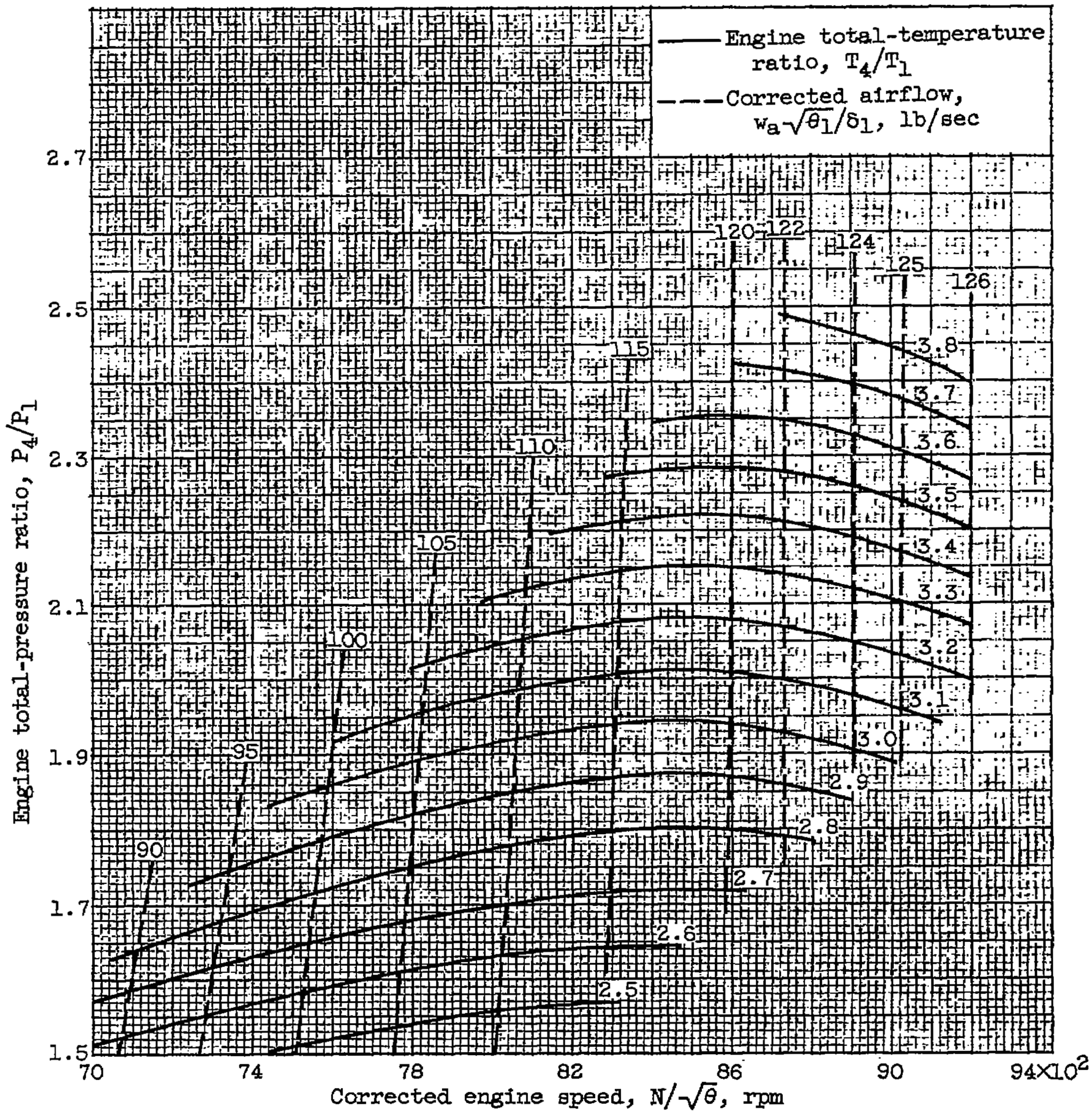


Figure 13. - Effect of altitude on turbine operation at rated engine speed and flight Mach number of 0.8.

4303



(a) Reference Reynolds number index of 0.105.

Figure 14. - Pumping characteristics with JP-4-fuel.

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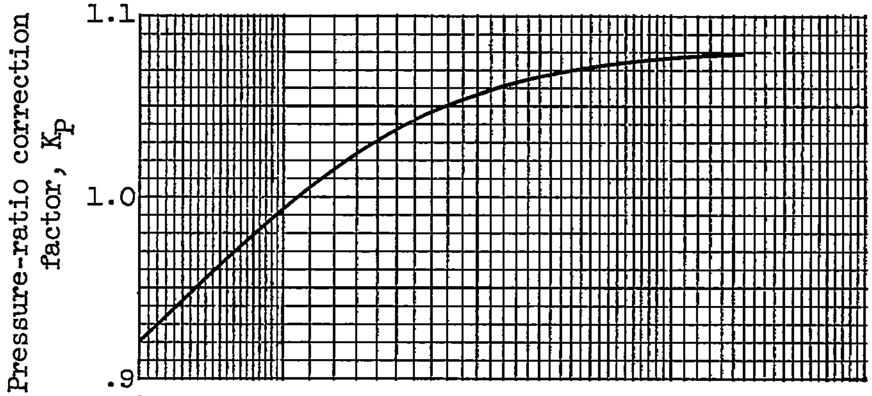
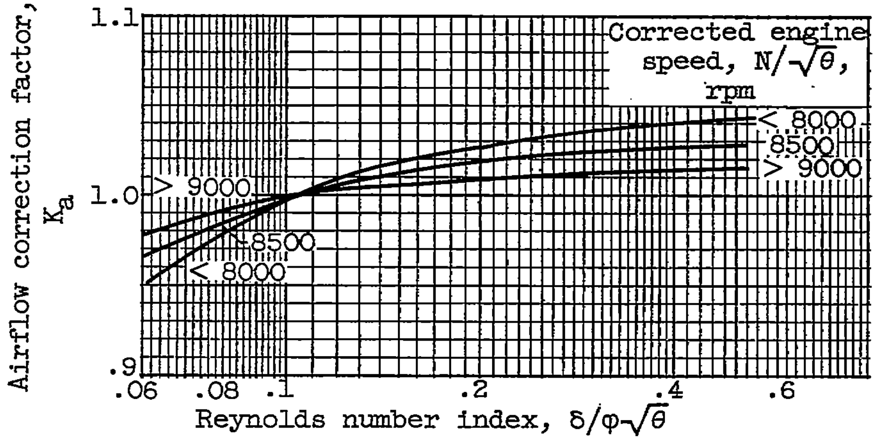


(b) Engine pressure-ratio correction for Reynolds number index.



(c) Engine airflow correction for Reynolds number index.

Figure 14. - Concluded. Pumping characteristics with JP-4 fuel.

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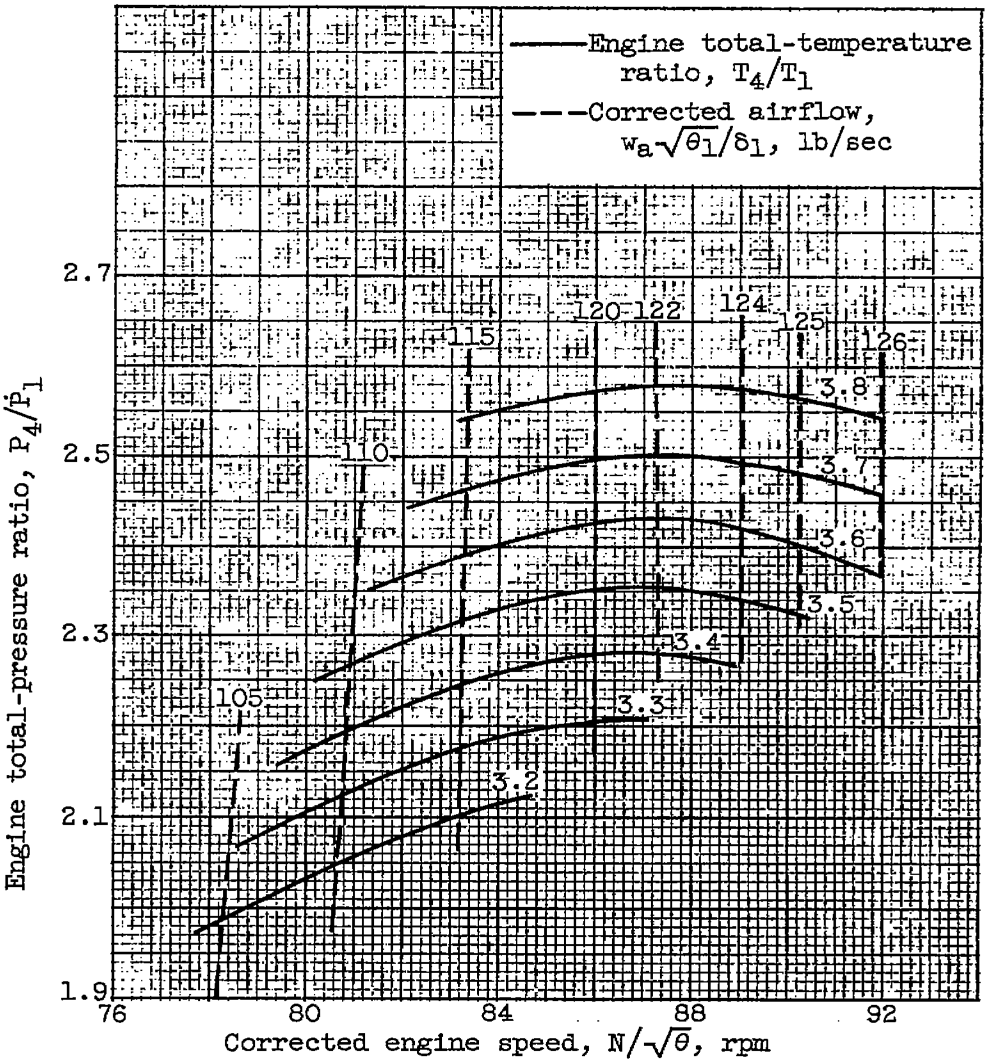
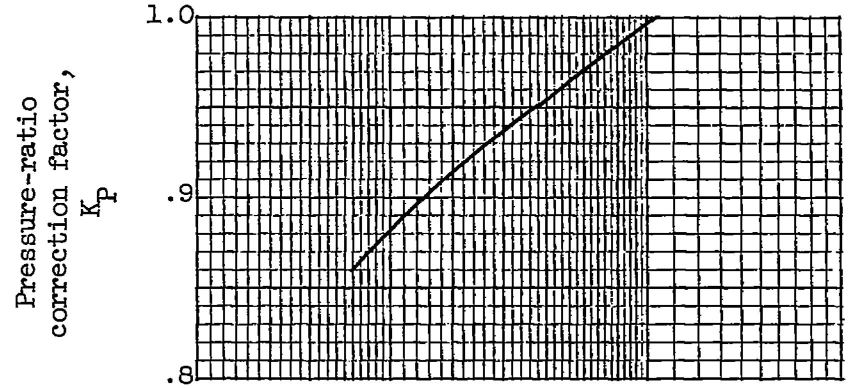


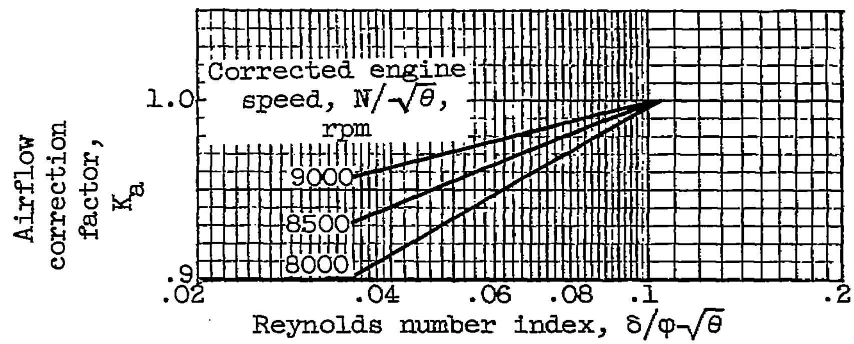
(a) Reference Reynolds number index of 0.105.

Figure 15. - Pumping characteristics with gaseous-hydrogen fuel.

[redacted]



(b) Engine pressure-ratio correction for Reynolds number index.



(c) Engine airflow correction for Reynolds number index.

Figure 15. - Concluded. Pumping characteristics with gaseous-hydrogen fuel.

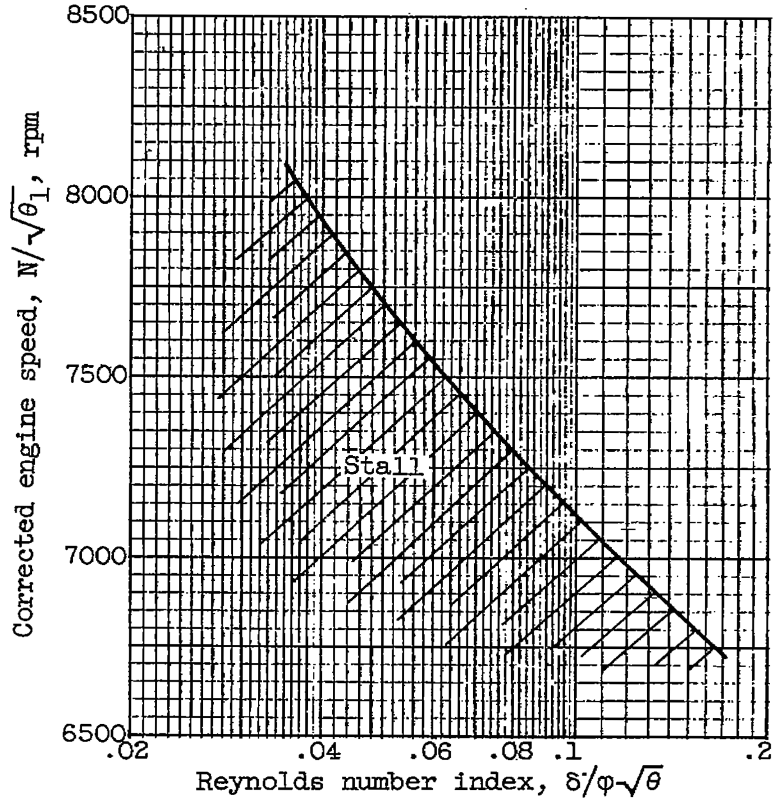


Figure 16. - Steady-state compressor stall limit.

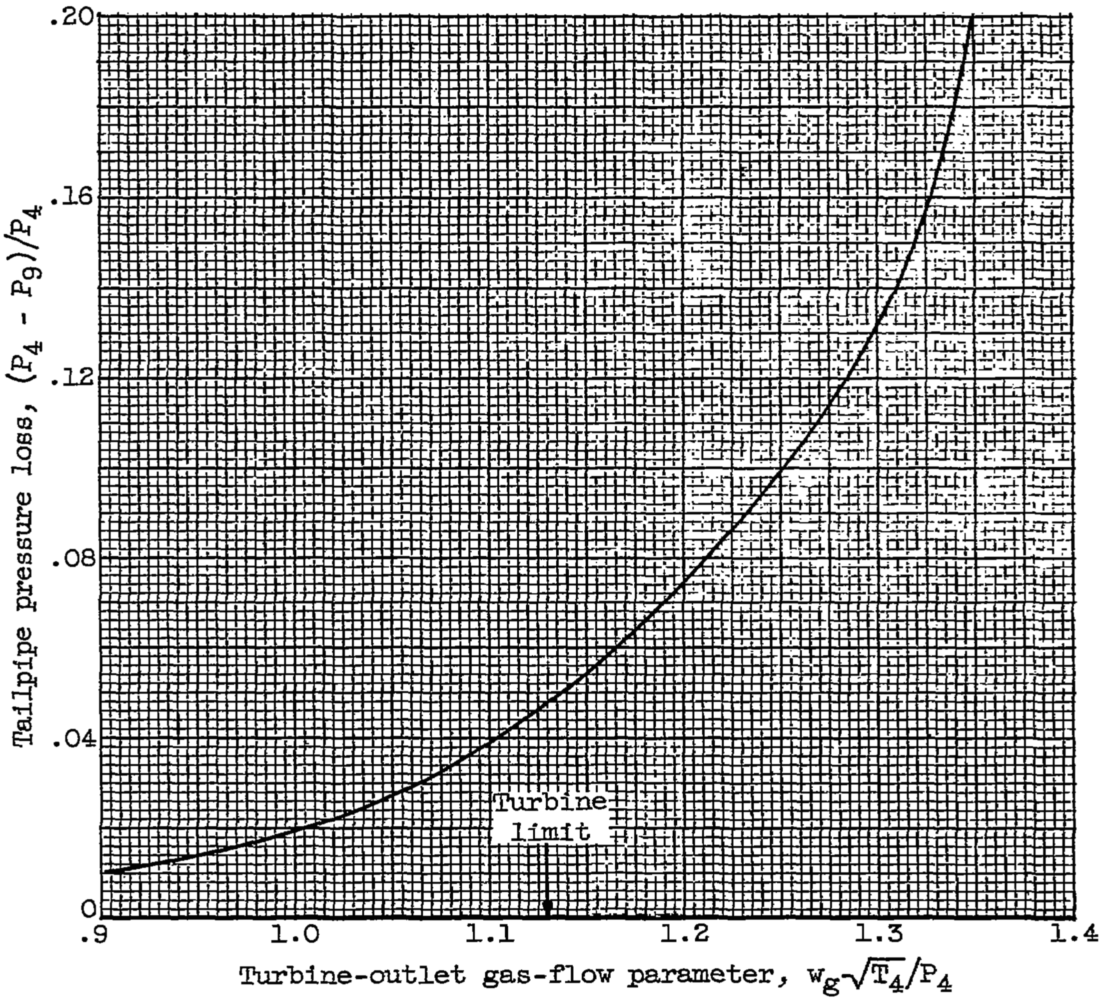


Figure 17. - Tailpipe pressure loss.