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RESEARCH MEMORANDUM

CALCULATED EFFECTS OF TURBINE ROTOR-BLADE COOLING-AIR FLOW, ALTITUDE, AND COMPRESSOR BLEED POINT ON PERFORMANCE OF A TURBOJET ENGINE

By Vernon L. Arne and Alfred J. Nachtigall

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Cleveland, Ohio

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RESEARCH MEMORANDUM

CALCULATED EFFECTS OF TURBINE ROTOR-BLADE COOLING-AIR FLOW, ALTITUDE, AND COMPRESSOR BLEED POINT ON PERFORMANCE OF A TURBOJET ENGINE

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SUMMARY

The effects of air-cooling the turbine rotor blades on the performance of an axial-flow turbojet engine were calculated for a range of altitude from sea level to 40,000 feet. The measured wind-tunnel performance of a commercial turbojet engine was used as the basis of comparison with the calculated performance for a turbine inlet temperature of 1960° R and for a range of cooling-air flows up to 3 percent of the compressor air flow. A flight Mach number of 0.788 was assumed for all altitudes.

The percentage change in thrust and specific fuel consumption compared with the performance of the uncooled engine is plotted for two cases: (1) with the cooling air bled from the compressor at the pressure required for each coolant flow; (2) for cooling air bled at the compressor discharge and throttled to the pressure required for each coolant flow for the entire range of conditions.

When the coolant was bled at the required coolant pressure, the sea-level percentage thrust reduction was approximately twice the percentage coolant flow. The percentage increase in specific fuel consumption relative to the uncooled engine at sea level was approximately equal to the percentage coolant flow. The percentage thrust reduction and percentage increase in specific fuel consumption decreased with an increase in altitude at a constant value of coolant-flow ratio. Bleeding the coolant at the compressor discharge resulted in an additional 1-percent loss in thrust and increase in specific fuel consumption at sea level with a smaller increase in loss at higher altitudes.

Additional calculations were made at sea level. Changing from a tapered blade cooling passage to a constant passage area equal to the area at the mean section of the tapered passage resulted in an improvement in engine performance. Increasing the compressor and turbine efficiencies from 0.80 and 0.79, respectively, to 0.88 resulted in

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considerably less performance loss due to cooling. Reducing the temperature of the cooling air at the inlet to the tapered blade-coolant passage from 300° to 100° F resulted in a reduction in static-pressure ratio from root to tip.

INTRODUCTION

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The application of cooled turbines is a promising means of improving the performance of aircraft gas-turbine engines by permitting the use of higher turbine-inlet temperatures. Another objective of cooled-turbine research is to reduce the critical-material content of the turbine blades and disks of current commercial turbojet engines. Both air- and liquid-cooled hollow blades offer possibilities of achieving these objectives, but air-cooling systems are considered a more immediate solution of the problem of substituting noncritical materials at current turbine inlet-gas temperature. The present status of the analytical and experimental cooled-turbine research and the general principles of turbine-cooling theory are reviewed in reference 1. Analytical methods of determining cooled-blade temperatures and required coolant flows are presented in references 2 and 3. These methods can be used in conjunction with experimental data to provide design values of required coolant flow.

Reference 2 discusses some of the problems in the design of air-cooled turbines and presents the effects of only the required cooling-air flow on turbojet engine performance for two particular blade configurations, various blade materials, and a particular stress-rupture life for a range of altitudes and turbine-inlet temperatures. The present report presents the calculated effects of coolant flow, altitude, and coolant bleed pressure on a turbojet engine operating at 1960° R (1500° F) turbine-inlet temperature and using the finned blade of reference 2. The results herein are independent of blade material and blade life. A range of rotor-blade coolant flows up to 3 percent of the compressor air flow is covered. The amount of coolant flow is discussed in terms of coolant-flow ratio, defined as the ratio of cooling-air flow to compressor-air flow. The results given in reference 2 indicate that coolant-flow ratios from 0 to 0.03 are of most interest for cases in which nonstrategic materials are substituted in turbine blades for current gas temperatures. This range of coolant flows necessitates the use of blades with high cooling effectiveness if satisfactory blade life is to be obtained. Altitudes from sea level to 40,000 feet were used for a constant flight Mach number of 0.788. The two conditions of compressor coolant bleed considered were: (1) The coolant is bled from the compressor at the pressure required to provide the specified coolant pressure determined from a pressure-drop calculation through the blade passage for a given cooled-blade design; (2) the coolant is bled at the compressor discharge and throttled to the

pressure required for all coolant flows and altitudes considered. For the second case, the performance is calculated independently of the blade configuration and heat-transfer characteristics, and the results are applicable to any blade configuration for which the compressor discharge pressure is adequate to supply the range of coolant flows desired.

Additional calculations were made at sea level to determine the effect of changing certain design variables that may, for other air-cooled turbine applications, have values different from those used here for the whole range of calculations. The effect on coolant-passage pressure drop and engine performance of changing from a tapered coolant passage to one having a constant passage area equal to the area of the tapered passage at the mean section was calculated for a coolant-flow ratio of 0.02 at sea level. Another calculation of engine performance was made at sea level for turbine and compressor efficiencies substantially higher than those used for the whole range of calculations. The effect of reducing the inlet cooling-air temperature at the blade root from 300° to 100° F on the coolant-passage pressure drop was also calculated for a coolant-flow ratio of 0.02 at sea level.

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METHODS

The method used in calculating engine performance is essentially the same as that in reference 2. The measured wind-tunnel performance of a commercial uncooled turbojet engine (references 4, 5, and 6) is used as a basis of comparison with the calculated performance of the same engine in which a cooled turbine is substituted. The results obtained are supplementary to those presented in reference 2 for a turbojet engine operating at current gas temperatures with nonstrategic material substituted in the turbine blades.

For both the cooled and the uncooled engine, the bulk turbine-inlet temperature was assumed to remain constant for all altitudes at 1960° R at rated engine speed of 7600 rpm. Coolant-flow ratios of 0.01, 0.02, and 0.03 were used at each altitude of sea level, 10,000, 20,000, 30,000, and 40,000 feet. A flight Mach number of 0.788 was used for all altitudes.

The general modifications to the turbine required for air cooling are shown in figure 1, which is identical to figure 1 of reference 2 except that no provision is made for cooling the stator blades. The rotor-blade cooling air, bled from the compressor, is introduced through a supply pipe to the stationary diffusing section at the turbine hub and passes into a vaned shroud on the face of the rotor. The air passes through the shrouded passage to the blade base, through the blade coolant passages, and is discharged at the tip to mix with the main flow of working gas.

A midspan cross section of the finned blade adapted to the turbine is shown in figure 2. Values of blade geometry were taken from this section to determine the average heat-transfer characteristics of the blade. The cooling-air pressure requirements for the range of conditions considered were computed for a passage that had a 3:1 taper ratio in cross-sectional area from the blade root to the blade tip. The cooling-air Mach number in the tip of the coolant passage and the pressure drop through the blade coolant passage were calculated by the method given in reference 7, which requires an initial assumption of cooling-air temperature at the blade root. An average value of 300° F was assumed for these calculations. The ducting losses between the compressor bleed and the turbine hub were neglected, and the effects of bleed on compressor performance and efficiency were not considered. The design of a turbojet engine or the modification of an existing engine for air cooling involves matching of the compressor and turbine. For the calculations reported herein, it was assumed that the compressor and turbine would still be matched when the air-cooled turbine was substituted for the uncooled turbine and when the turbine-blade cooling air was bled from the compressor.

The performance of the engine was calculated for the entire range of operating conditions for two cases of compressor cooling-air bleed. For the first case, the required cooling-air pressure was determined by the method of reference 7; the coolant pumping work in the main compressor up to the point of required bleed pressure (defined as external pumping work) and the coolant pumping work in the turbine rotor itself (defined as internal pumping work) were calculated. For the second case, where the cooling air was bled at the compressor discharge, it was unnecessary to make a separate calculation of external pumping work because the enthalpy rise of the cooling air is the same as that of the working fluid in the main compressor and could therefore be included in the total compressor work. The effects of internal and external pumping work were included in all calculations of turbine pressure ratio, jet-nozzle inlet temperature, jet-nozzle pressure ratio, and the percentage change in thrust and specific fuel consumption. For the case of coolant bleed at the compressor discharge, only the percentage change in thrust and specific fuel consumption are presented. The results for bleed at the compressor discharge apply to any coolant-passage configuration requiring a coolant supply pressure less than or equal to the compressor discharge pressure.

The pressure drop through a turbine-blade coolant passage depends to a large extent on the relative Mach number of the cooling air. For a given coolant-passage area at the root of the blade and a cooling-air weight flow, a cooling passage that converges in the direction of flow will have a rapidly increasing cooling-air Mach number toward the blade tip and will have a larger pressure drop than a blade coolant passage of constant flow area. The blade used for the entire range of calculations had a taper ratio of 3:1 from root to tip. In order to study

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the effect of changing to a constant-area coolant passage on coolant-passage pressure drop and engine performance, additional calculations were made for a constant-area coolant passage for a coolant-flow ratio of 0.02 at sea level. The area of the constant-area passage was equal to the mean area of the tapered passage so that the average heat-transfer characteristics of the blade remained unchanged.

The compressor and turbine adiabatic efficiencies used for most of the calculations were based on the measured engine performance. At sea level, the turbine efficiency was 0.79 and the compressor efficiency was 0.80. In order to determine the effect of higher component efficiency, the sea-level performance was recalculated using 0.88 adiabatic efficiency for both compressor and turbine.

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As previously mentioned, the cooling-air temperature at the blade root was assumed to be 300° F. For effective blade cooling, it is desirable to supply the coolant at as low a temperature as possible. A coolant-passage pressure-drop calculation was made assuming 100° F blade-root cooling-air temperature for 0.02 coolant-flow ratio at sea level so that a comparison could be made with the 300° F inlet-air calculation.

RESULTS AND DISCUSSION

The effect of cooling-air bleed on axial-flow turbojet engine performance was calculated for a range of coolant flows up to 3 percent of the compressor air flow for altitudes ranging from sea level to 40,000 feet. Performance of the cooled engine was compared with the measured altitude-wind-tunnel performance of the commercial uncooled engine.

Cooling-Air-Pressure Requirements

Cooling-air Mach number in tip of passage. - The Mach number of the cooling air in the tip of the passage is needed to initiate the pressure-drop calculation by the method of reference 7. Results of calculations have shown that the tip Mach number itself is an approximate indication of the cooling-air pressure drop, the higher Mach numbers resulting in higher static-pressure losses.

The variation of cooling-air Mach number in the tip of the coolant passage is shown in figure 3 for altitudes of sea level and 40,000 feet for two cross-sectional tip coolant-passage areas. At a given coolant-flow ratio, the Mach number is slightly higher at 40,000 feet than at sea level for either tip area. For a tip area of 0.028 square inch, corresponding to a root-to-tip taper ratio of 3:1 for a blade with a mean cross section as shown in figure 2, the Mach number increases with

cooling-air flow and is between 0.9 and 1.0 at a coolant-flow ratio of 0.03. When the tip area is increased by assuming a constant-area coolant passage with an area equal to the mean area shown in figure 2, the tip Mach number as shown by the dashed lines has a value of about 0.5 at a coolant-flow ratio of 0.03. The coolant pressure drop and the engine performance for the constant-area blade were calculated for comparison with the tapered blade at one coolant flow at sea level and the computed values are given in table I.

Coolant pressure ratio. - The variation in static pressure from rotor hub to blade tip is shown in figure 4 for the tapered blade and the constant-area blade for two values of rotor compression efficiency at sea level with a coolant-flow ratio of 0.02. The pressure distribution in the blade was calculated by the method of reference 7, which assumes one-dimensional compressible flow. For the tapered blade, there is a static-pressure ratio of 1.29 from blade root to blade tip. The pressure builds up near the root in this blade because of compressor action but rapidly decreases as the effects of momentum pressure loss and friction predominate. The constant-area blade has a slight pressure rise from root to tip.

The pressure drop in the tapered blade was also calculated for a coolant-flow ratio of 0.02 and a coolant-inlet temperature of 100° F. The pressure ratio from root to tip was reduced from 1.29 for a coolant-inlet temperature of 300° F to 1.16 for a coolant-inlet temperature of 100° F. This result does not show the entire advantage to be gained with lower coolant temperatures because both calculations were done for one value of coolant-flow ratio. For a given blade, the lower coolant temperature would give reduced blade temperature, thus making possible a reduction in the amount of coolant flow required to maintain a given allowable blade temperature. The lower coolant flow would result in a further reduction in coolant pressure drop in the blade. The effects on engine performance of the lower blade coolant pressure drop for a coolant-inlet temperature of 100° F were not calculated inasmuch as the effects of the resultant lower required coolant bleed pressure are similar to the effects on engine performance observed for the 300° F coolant-inlet temperature for the two cases of compressor bleed.

The static-pressure variation in the rotor was calculated using polytropic compression efficiencies of 0.50 and 1.00. For an efficiency of 0.50, very little static-pressure rise is obtained in the rotor. The variation obtained using 1.00 compression efficiency indicates ideal rotor performance and shows that there is room for substantial reduction in coolant-pressure requirements. Experimental evidence indicates that an efficiency of 0.50 is a reasonable value to be expected in early air-cooled rotor designs and, consequently, this value was used in the calculations of engine performance.

The ratio of required total pressure at the rotor hub to the static pressure in the blade tip is given in figure 5 for a range of altitude and coolant-flow ratio. The required cooling-air pressure ratio increases considerably with coolant flow and only slightly with altitude.

At a coolant-flow ratio of 0.02 at sea level, the required pressure ratio from hub to tip is 1.29. Table I shows that use of a blade with constant cross-sectional passage area reduces the required pressure ratio to 1.04, a 20-percent reduction.

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The information of figure 5 was expressed in terms of required coolant pressure ratio assuming no loss in pressure between the compressor bleed point and the turbine hub. Figure 6 compares these values with compressor pressure ratio for sea level and 40,000-foot altitude. The curve shows that for this blade configuration the compressor discharge pressure ratio is required for coolant-flow ratios of approximately 0.040 and 0.035 at sea level and 40,000 feet, respectively. The determination of the blade pressure drop thus may be of considerable importance in air-cooled engine design because the required coolant pressure should not exceed the available coolant pressure.

Coolant Pumping Work and Specific Turbine Work

The work done on the cooling air in the main compressor is called the external work and is plotted in figure 7 for the range of coolant-flows and altitudes considered. Inasmuch as the values are dependent on the bleed point in the compressor given by figure 6, the magnitude of external pumping work in figure 7 is not applicable for all blade configurations. The figure does give a qualitative picture of the pumping work for this range of coolant flows as compared with the total turbine work.

The internal pumping work of the turbine rotor in Btu per pound of working gas for coolant-flow ratios up to 0.03 is shown in figure 8. Because this work is a function of only the coolant weight flow and the blade tip speed, it can be calculated easily for any turbine design. The values of figure 8 are applicable for tip speeds around 1125 feet per second.

The specific turbine work, obtained by adding the external and the internal pumping work to the compressor work, is shown in figure 9. The reduction in working fluid because of coolant bleed is the main effect of cooling on the specific turbine work. The variation given by figure 9 is therefore typical for any engine using this coolant flow range having a compressor pressure ratio and turbine-inlet temperature approximately that of the engine for which these calculations were made.

Figure 9 indicates that for a coolant flow ratio of 0.03 the specific turbine work shows an increase over that of the uncooled engine of about 4.4 percent at sea level and about 4.8 percent at 40,000 feet. At a given coolant-flow ratio, the specific turbine work increases slightly with altitude owing to the increase in the required cooling-air pressure ratio at higher altitudes. It is desirable to have a maximum amount of energy available for thrust; the energy drop across the turbine or the specific turbine work should therefore be kept as low as possible. Figure 9 also shows that a more rapid increase in specific turbine work occurs as the coolant flow is increased. This suggests the importance of minimizing the amount of required coolant flow by using blades of high cooling effectiveness.

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Jet-Nozzle Inlet Conditions

The effect of cooling on the state of the working fluid entering the jet nozzle can be observed from the effect on turbine pressure ratio and jet-nozzle total temperature as shown in figures 10 and 11. Because of higher specific turbine work, the turbine pressure ratio increases with coolant-flow ratio from 2.15 for the uncooled engine to about 2.24 for a 0.03 coolant-flow ratio. This variation is almost the same over the range of altitudes considered. The jet-nozzle total temperature at sea level decreases from 1695° R in the uncooled engine to about 1661° R at a coolant-flow ratio of 0.03. Again the effect of altitude is slight. The absence of appreciable altitude effect on turbine pressure ratio and jet-nozzle total temperature is an engine characteristic and has little direct connection with the cooling characteristics.

Inasmuch as compressor pressure ratio increases quite appreciably with altitude and turbine pressure ratio is not affected by altitude; the increase in compressor pressure ratio is reflected in the jet-nozzle pressure ratio, which increases with altitude as shown in figure 12. The jet-nozzle pressure ratio decreases with coolant-flow ratio at all altitudes because of the additional turbine pressure ratio requirements that cooling introduces.

Engine Performance

The ultimate objective of the analysis is to compute the changes in engine performance that occur with cooling. Results are presented for the case of cooling-air bleed at the calculated required pressure and the case of cooling-air bleed at the compressor discharge.

Cooling air bleed at calculated required pressure. - The percentage changes in thrust and specific fuel consumption are plotted

against coolant-flow ratio in figures 13(a) and 13(b), respectively. The sea-level percentage decrease in thrust is approximately twice the percentage coolant flow, whereas the sea-level percentage increase in specific fuel consumption is approximately equal to the percentage coolant flow. Both the thrust reduction and specific fuel consumption increase become smaller as altitude increases at a constant value of coolant-flow ratio. The thrust reduction is larger than the increase in specific fuel consumption because the thrust is affected by the reduction in turbine working fluid due to compressor bleed as well as by the coolant pumping losses and the changes in propulsive efficiency. The coolant pumping losses can be controlled to some extent by careful coolant-passage design, but the loss in thrust due to bleeding from the engine compressor weight flow and by-passing the combustion chambers is a loss that is inherent in this type of air-cooled engine configuration and can be decreased only by decreasing the amount of cooling air used. The possibility of restoring the thrust by increasing the turbine-inlet temperature above its former value is discussed to some extent in reference 2.

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The smaller loss in performance with increase in altitude is probably due to the compressor characteristics of the engine considered. The compressor pressure ratio increases with altitude and thus improves the thermal efficiency of the cycle. The cooling losses for a given coolant-flow ratio then become a smaller part of the available thermal energy.

Cooling-air bleed at compressor discharge. - If the point of compressor bleed is known or can be assumed, the engine performance can be calculated independent of the blade configuration or heat-transfer requirements because any heat transferred to the coolant is reintroduced into the jet nozzle during mixing after the turbine, making the energy drop across the turbine a function of only the specific turbine work. This fact is of importance to the designer because it allows him to make an approximate analysis of the effect of cooling on performance without an accurate determination of the blade pressure drop and cooling characteristics, which are difficult to determine quickly.

A calculation was made to determine the effect of bleeding the cooling air at the compressor discharge for the range of coolant flows and altitudes considered. The results are shown in figures 14(a) and 14(b). Comparison with figures 13(a) and 13(b) shows that the additional loss is about 1 percent at sea level, with even smaller differences at higher altitudes. This results from the additional pumping work required for bleed at a higher compressor pressure ratio, but for a fixed coolant temperature it appears that the amount of compression done on the cooling air has a smaller effect on engine performance than does the amount of coolant bleed. On the other hand, unless a

heat exchanger is used to control coolant temperatures, higher coolant-bleed pressure ratios result in higher bleed temperatures, which in turn give less blade-temperature reduction for a given coolant flow.

Effect on engine performance of changing to constant-area blade-coolant passage. - Table I shows the effect on the performance loss of making the coolant passage of constant cross section rather than tapered. The flow area of the constant-area blade is equal to the mean flow area of the tapered blade. Required coolant pressure was calculated for the two cases for a coolant-flow ratio of 0.02 at sea level. The thrust reduction changes from 3.9 percent for the tapered blade to 3.5 percent for the constant-area blade; the specific fuel consumption increase changes from 1.9 percent to 1.6 percent. From the standpoint of performance, it appears worthwhile to reduce the pressure drop of the cooling air in the blade. It must be remembered, however, that the heat-transfer effectiveness and, therefore, the life of a blade depend on the relative cooling-air velocities, and there must be a design compromise between cooling effectiveness and blade pressure drop. The average cooling effectiveness of the blade considered herein was not affected by changing from a tapered flow area to a constant-flow area because the average heat-transfer characteristic is based on the mean area, which was unchanged.

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Effect on engine performance of increased engine component efficiencies. - The compressor and turbine adiabatic efficiencies used in the calculations were measured in the operating engine. Because these efficiencies were rather low compared with the performance of present-day axial-flow engines, a calculation was made at sea level with improved values of efficiencies to determine the effect on performance losses. The turbine efficiency was increased from 0.79 to 0.88 and the compressor efficiency was increased from 0.80 to 0.88. A comparison of significant performance factors with the different values of efficiency is shown in table II. The loss in thrust at a coolant-flow ratio of 0.03 was reduced from 6.2 to 4.7 percent and the fuel consumption increase was reduced from 3.4 to 1.8 percent. Here again, as with the decrease in performance loss with altitude, the more efficient engine can absorb the losses due to cooling more easily. The losses become a smaller part of the total energy available.

SUMMARY OF RESULTS

The effect of air-cooling the turbine blades on the performance of an axial-flow turbojet engine was investigated. The state of the working gas and the cooling air was determined at various points in the cycle. The following results were obtained:

1. When the coolant was bled at the required coolant pressure, the sea-level percentage reduction in thrust was approximately twice the percentage coolant flow and the increase in specific fuel consumption was approximately equal to the percentage coolant flow.

2. As altitude increased, both the percentage reduction in thrust and percentage increase in specific fuel consumption became less for any fixed value of coolant-flow ratio.

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3. Bleeding the coolant at the compressor discharge resulted in an additional 1-percent loss in performance at sea level and in smaller increase in loss of performance at higher altitudes.

4. For a blade coolant passage with a 3:1 taper ratio from root to tip, the required cooling-air pressure ratio increased with coolant-flow ratio, reaching compressor discharge pressure at a coolant-flow ratio of about 0.035 at 40,000 feet and about 0.040 at sea level.

5. For a blade of constant coolant-passage area equal to the area of the mean section of the tapering passage, the Mach number of the cooling air at the blade tip was reduced from 0.670 to 0.345 and the cooling-air pressure ratio required was reduced from 2.10 to 1.68 at sea level for a coolant-flow ratio of 0.02. This resulted in a change in percentage thrust reduction from 3.9 percent for the tapered blade to 3.5 percent for the untapered blade. The specific fuel consumption increase changed from 1.9 percent for the tapered blade to 1.6 percent for the untapered blade.

6. Increase in the compressor and turbine efficiencies from 0.80 and 0.79, respectively, to 0.88 resulted in less performance loss due to cooling. At sea level, for a coolant-flow ratio of 0.03, the loss in thrust was reduced from 6.2 to 4.7 percent and the specific fuel consumption increase was reduced from 3.4 to 1.8 percent when the higher component efficiencies were used.

7. Reducing the temperature of the cooling air at the inlet to the tapered blade-coolant passage from 300° to 100° F resulted in a reduction in static-pressure ratio from root to tip from 1.29 for a coolant-inlet temperature of 300° F to 1.156 for a coolant-inlet temperature of 100° F.

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TABLE I - COMPARISON OF SIGNIFICANT ENGINE-PERFORMANCE FACTORS FOR TAPERED AND CONSTANT-AREA BLADE COOLANT PASSAGE

[Altitude, sea level; coolant-flow ratio, 0.02; flight Mach number, 0.788; turbine-inlet temperature, 1960° R]

Type of cooling passage	Tapered	Constant area
Cooling-air passage area at blade tip, sq in.	0.028	0.056
Cooling-air Mach number in passage tip	0.670	0.345
Pressure ratio from hub to blade tip	1.29	1.04
Required cooling-air pressure ratio	2.10	1.68
External coolant pumping work, Btu/lb working gas	0.83	0.57
Specific turbine work, Btu/lb working gas	76.2	75.9
Turbine pressure ratio	2.20	2.19
Jet-nozzle total temperature, °R	1674	1676
Jet-nozzle pressure ratio	2.165	2.175
Thrust reduction, percent	3.9	3.5
Specific fuel consumption increase, percent	1.9	1.6

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TABLE II - COMPARISON OF SIGNIFICANT ENGINE-PERFORMANCE FACTORS

FOR DIFFERENT VALUES OF COMPONENT EFFICIENCY

[Altitude, sea level; coolant-flow ratio, 0.03; tapered blade; flight Mach number, 0.788; turbine-inlet temperature, 1960° R]

Compressor efficiency	0.80	0.88
Turbine efficiency	0.79	0.88
Compressor pressure ratio	3.51	4.44
External coolant pumping work, Btu/lb working gas	1.80	1.64
Specific turbine work, Btu/lb working gas	77.7	86.0
Turbine pressure ratio	2.24	2.25
Jet-nozzle total temperature, °R	1661	1633
Jet-nozzle pressure ratio	2.13	2.68
Thrust reduction, percent	6.2	4.7
Specific fuel consumption increase, percent	3.4	1.8

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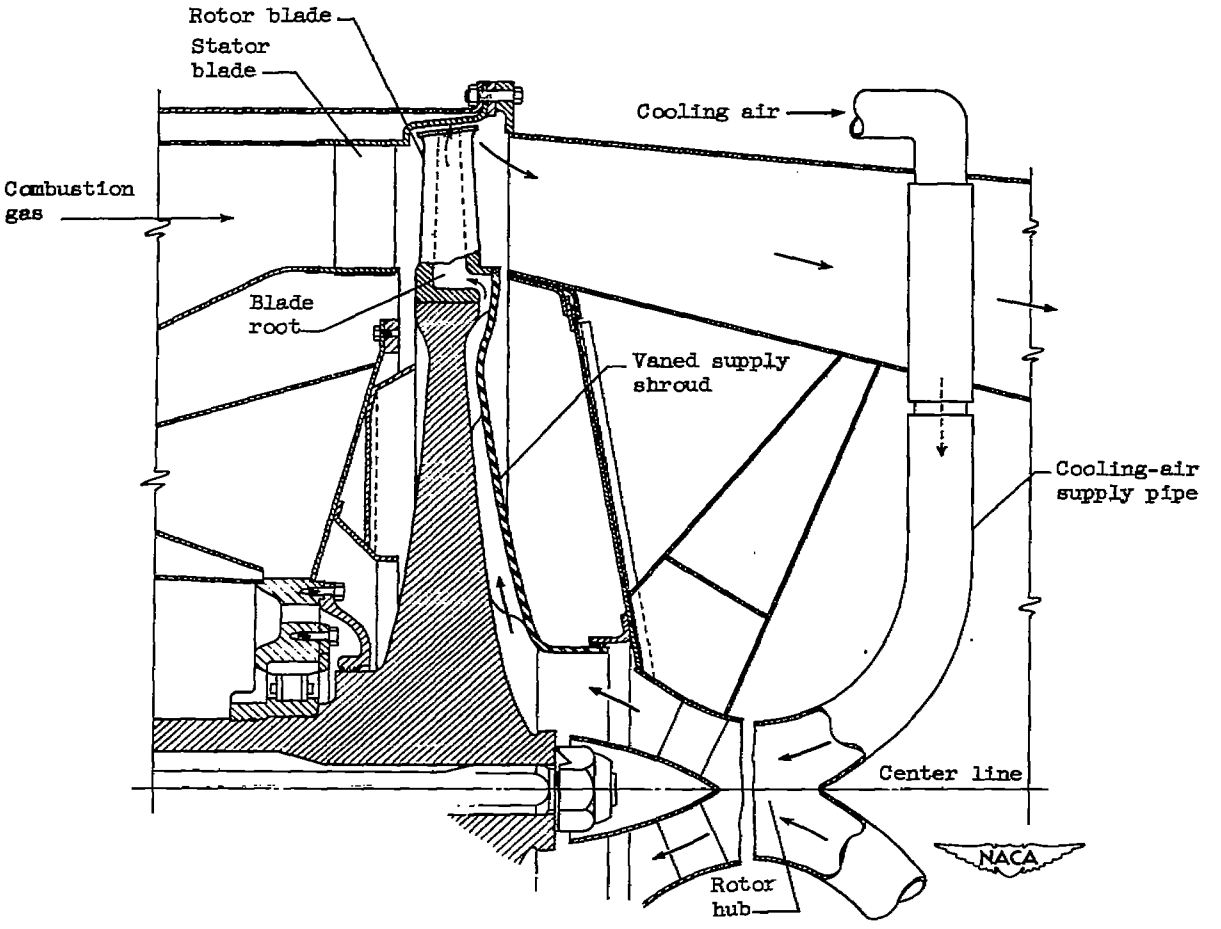

Figure 1. - Section of turbine rotor showing proposed method of introducing cooling air to rotor blades.

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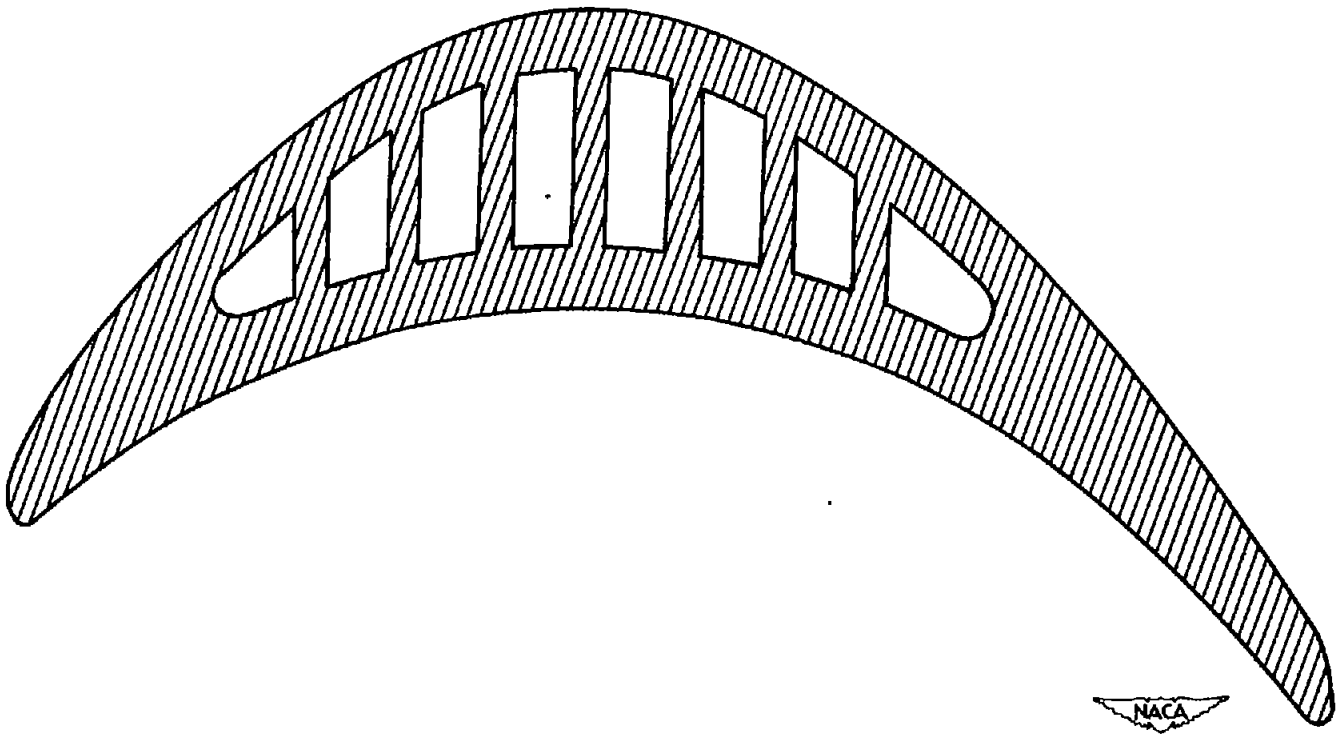

Figure 2. - Mean cross section of finned air-cooled turbine blade.

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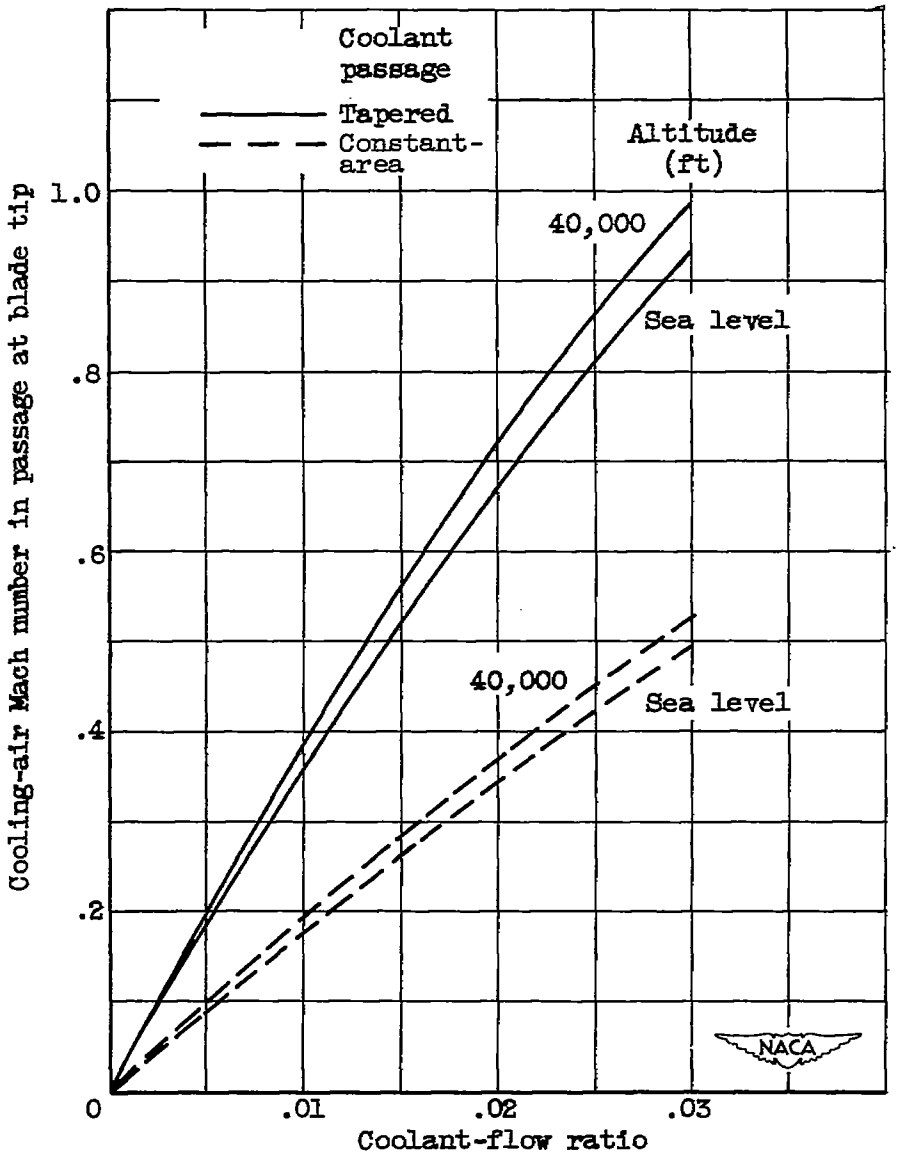


Figure 3. - Cooling-air Mach number in passage at blade tip plotted against coolant-flow ratio for blade coolant passage with seven fins. Flight Mach number, 0.788; engine speed, 7600 rpm.

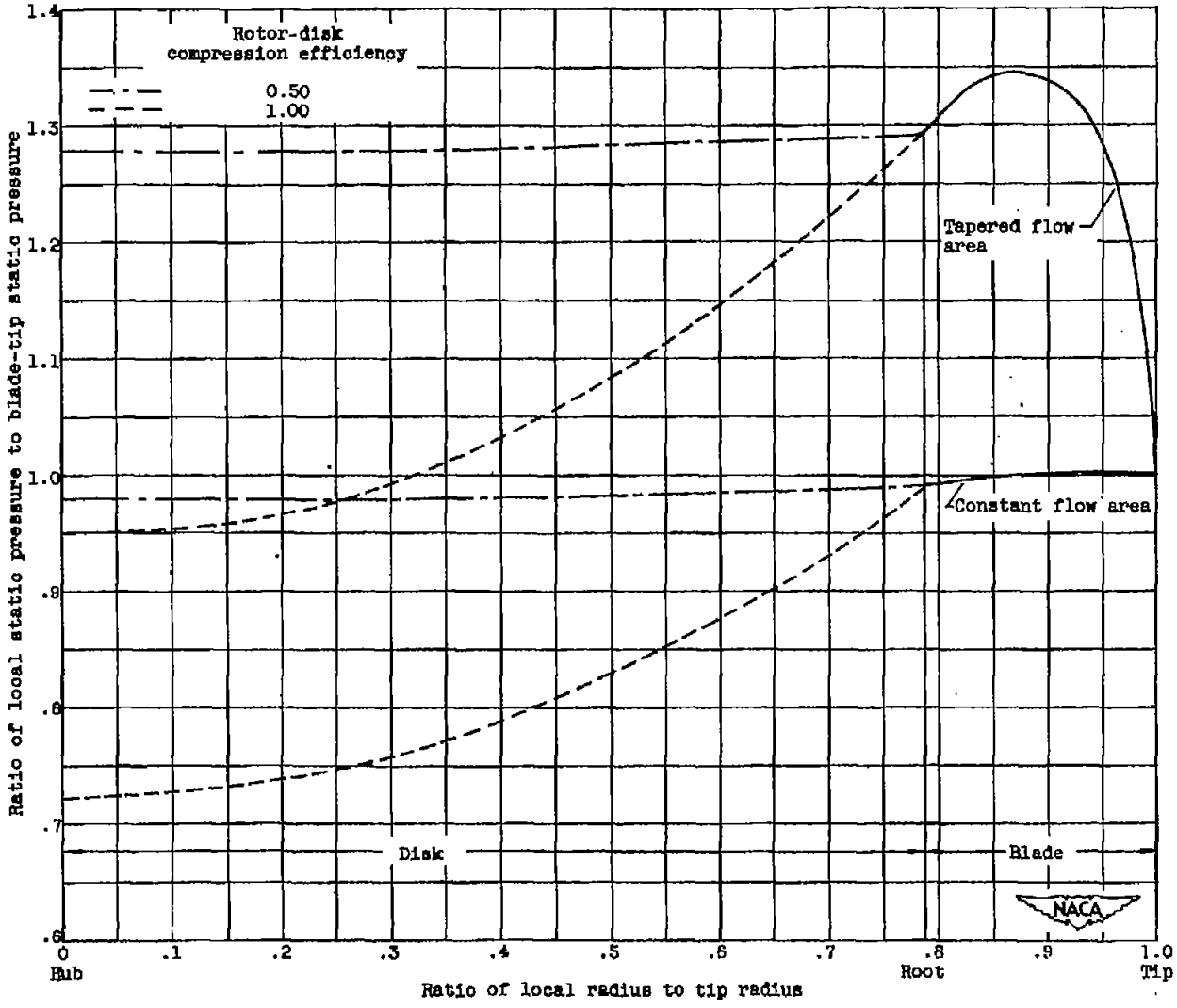


Figure 4. - Variation of ratio of local static pressure to static pressure at blade tip for tapered and constant-flow-area blade and for two values of polytropic rotor compression efficiency. Coolant-flow ratio, 0.02; flight Mach number, 0.788 at sea level; engine speed, 7600 rpm.

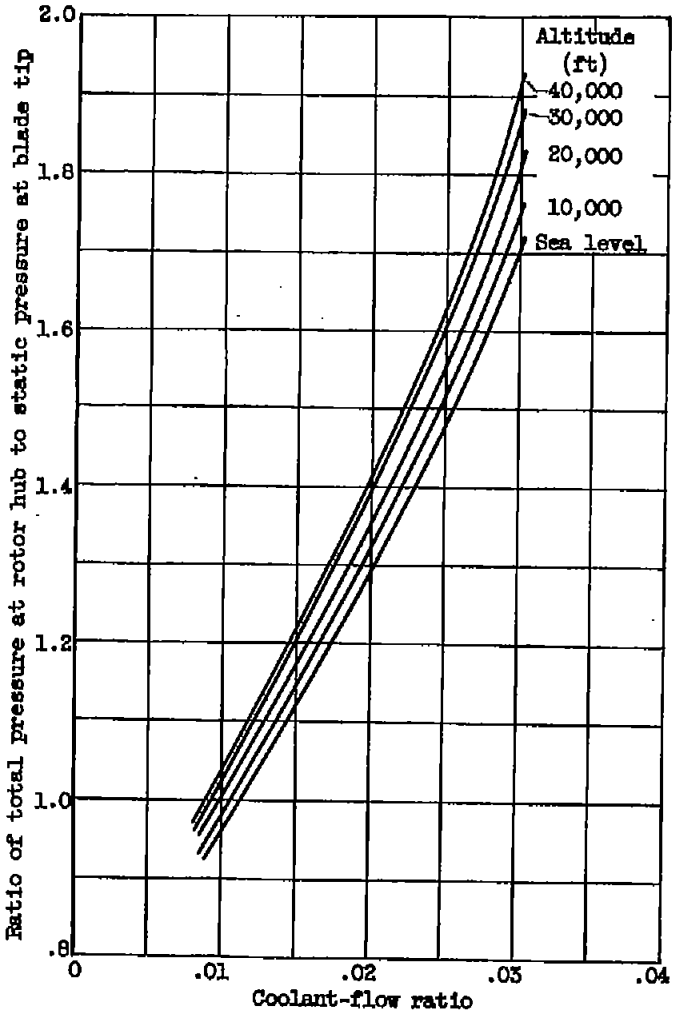


Figure 5. - Variation of ratio of total pressure at rotor hub to static pressure at blade tip with coolant-flow ratio for blade coolant passage with seven fins. Flight Mach number, 0.788; engine speed, 7600 rpm.

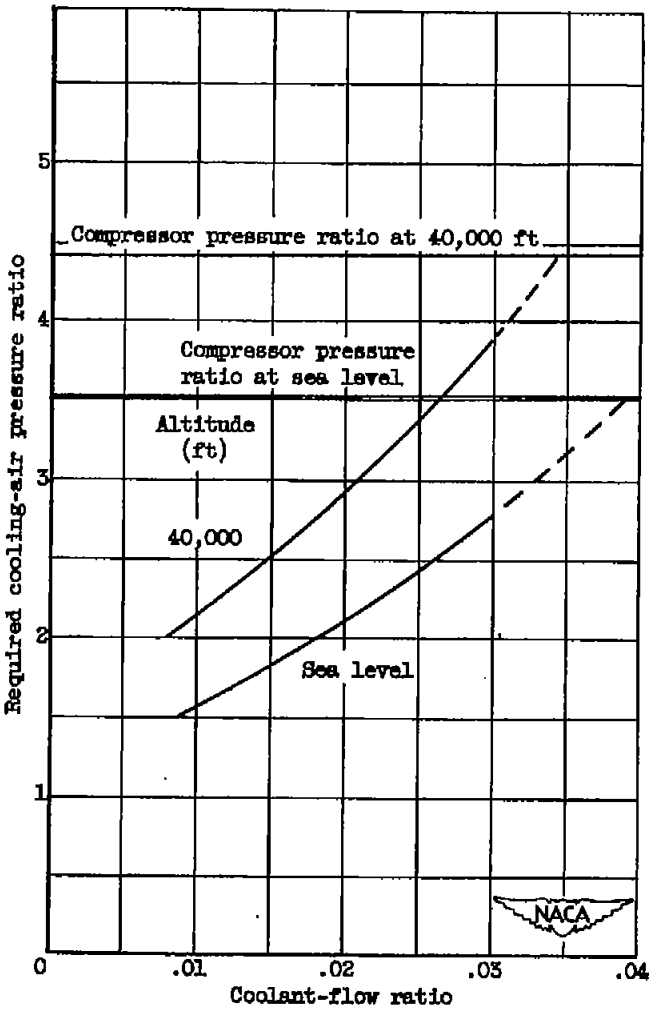


Figure 6. - Compressor pressure ratio required for cooling air for a seven-finned blade. Flight Mach number, 0.788; engine speed, 7600 rpm.

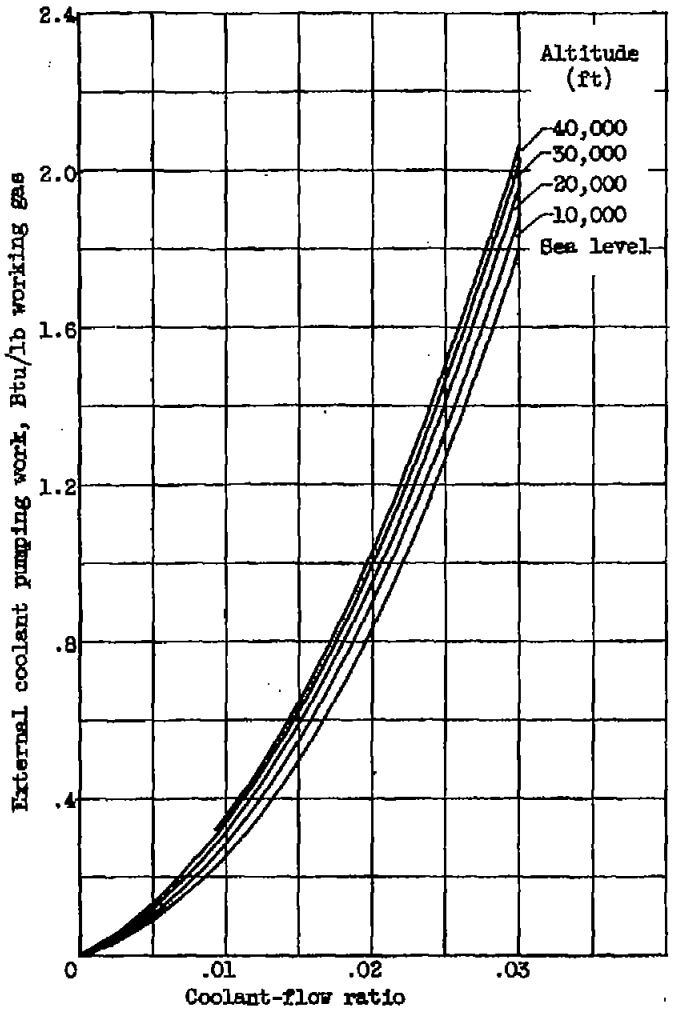


Figure 7. - Variation of external coolant pumping work per pound of working gas with coolant-flow ratio for air-cooled turbine with finned blades. Flight Mach number, 0.788; engine speed, 7600 rpm.

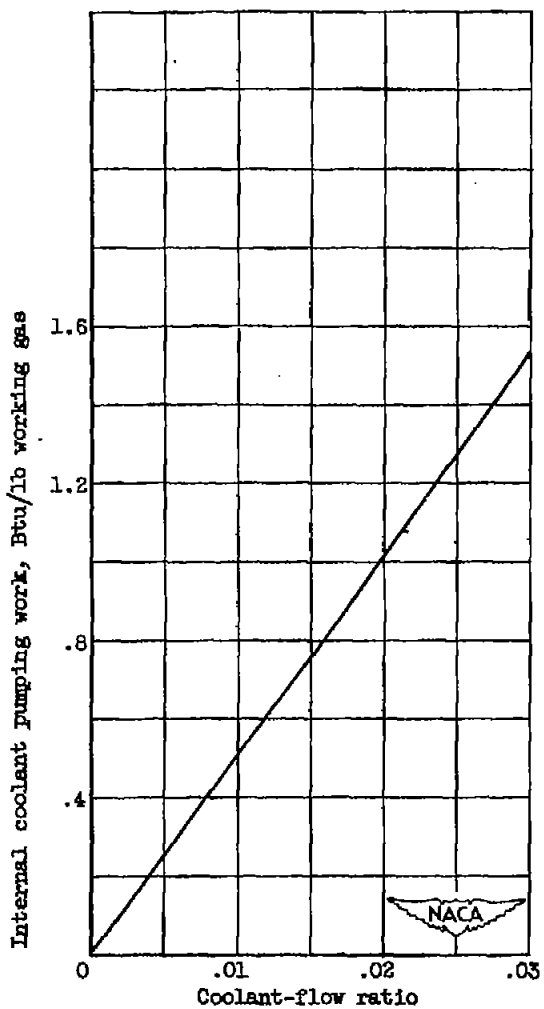


Figure 8. - Variation of internal coolant pumping work with coolant-flow ratio for air-cooled turbine rotor operating at 7600 rpm. Radius to blade tip, 1.416 feet.

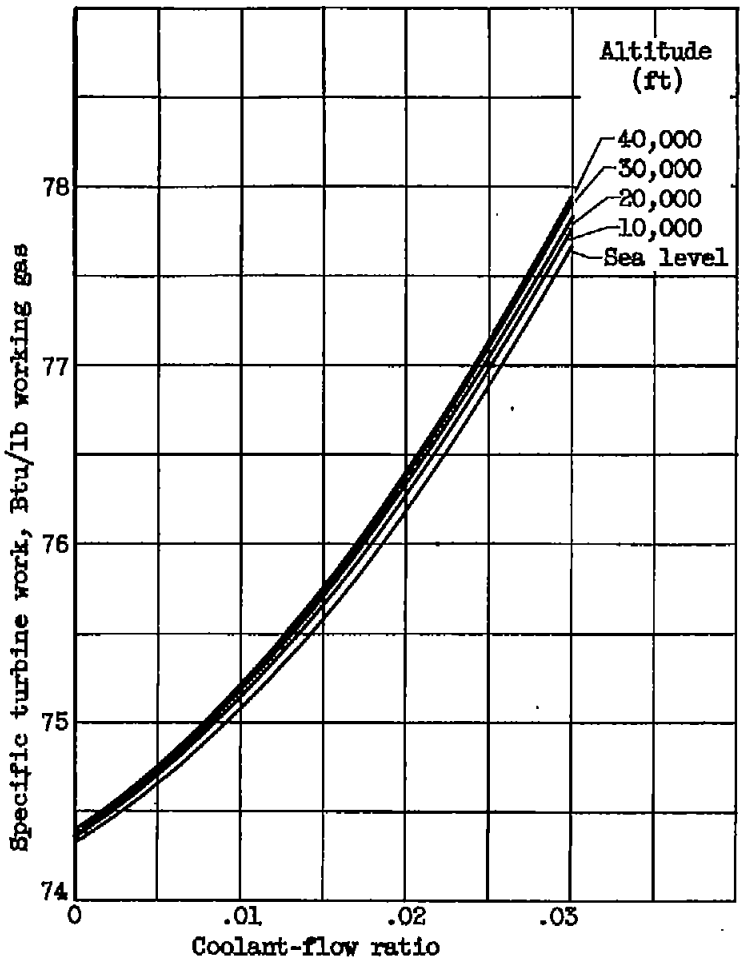


Figure 9. - Variation of specific turbine work with coolant-flow ratio for air-cooled turbine with finned blades. Flight Mach number, 0.788; engine speed, 7600 rpm.

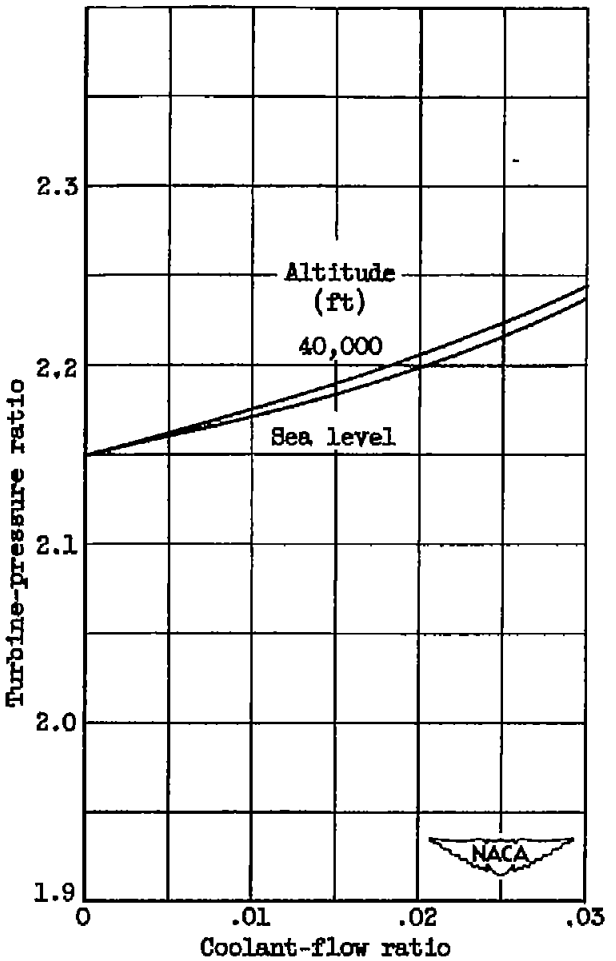


Figure 10. - Variation of turbine pressure ratio with coolant-flow ratio for air-cooled turbine with finned blades. Flight Mach number, 0.788; engine speed, 7600 rpm; turbine-inlet temperature, 1960° R.

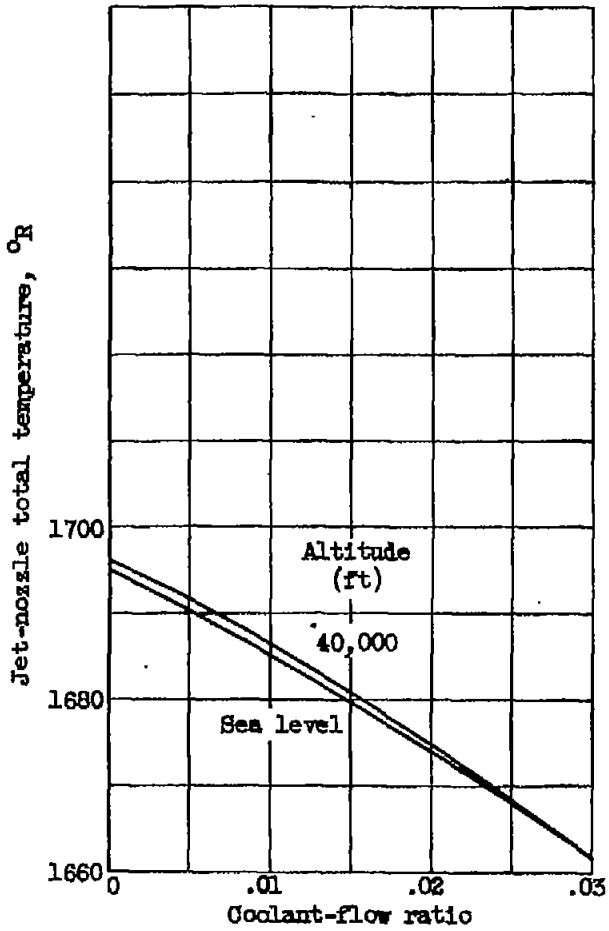


Figure 11. - Variation of jet-nozzle total temperature with coolant-flow ratio for turbojet engine with air-cooled finned turbine blades. Flight Mach number, 0.788; engine speed, 7600 rpm; turbine-inlet temperature, 1960° R.

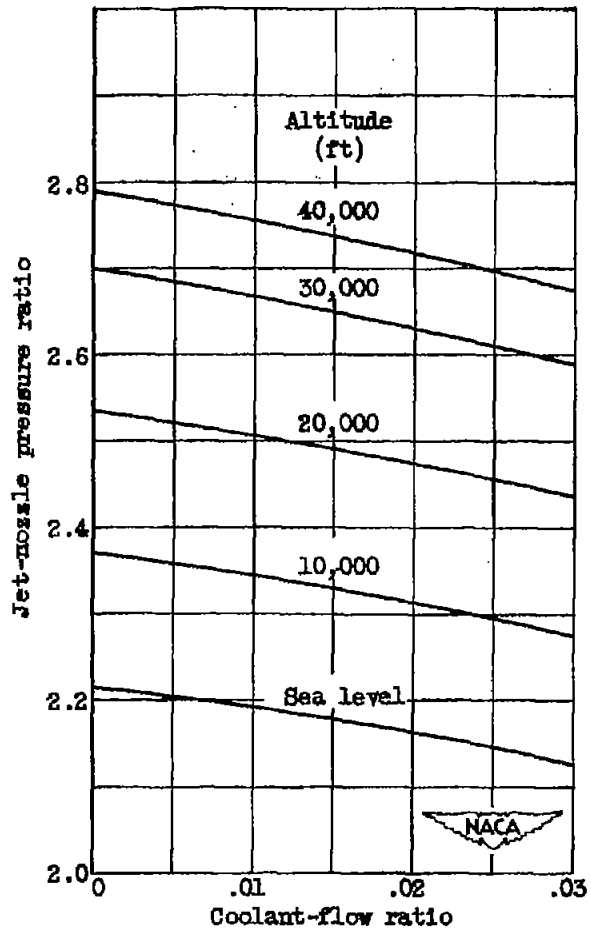
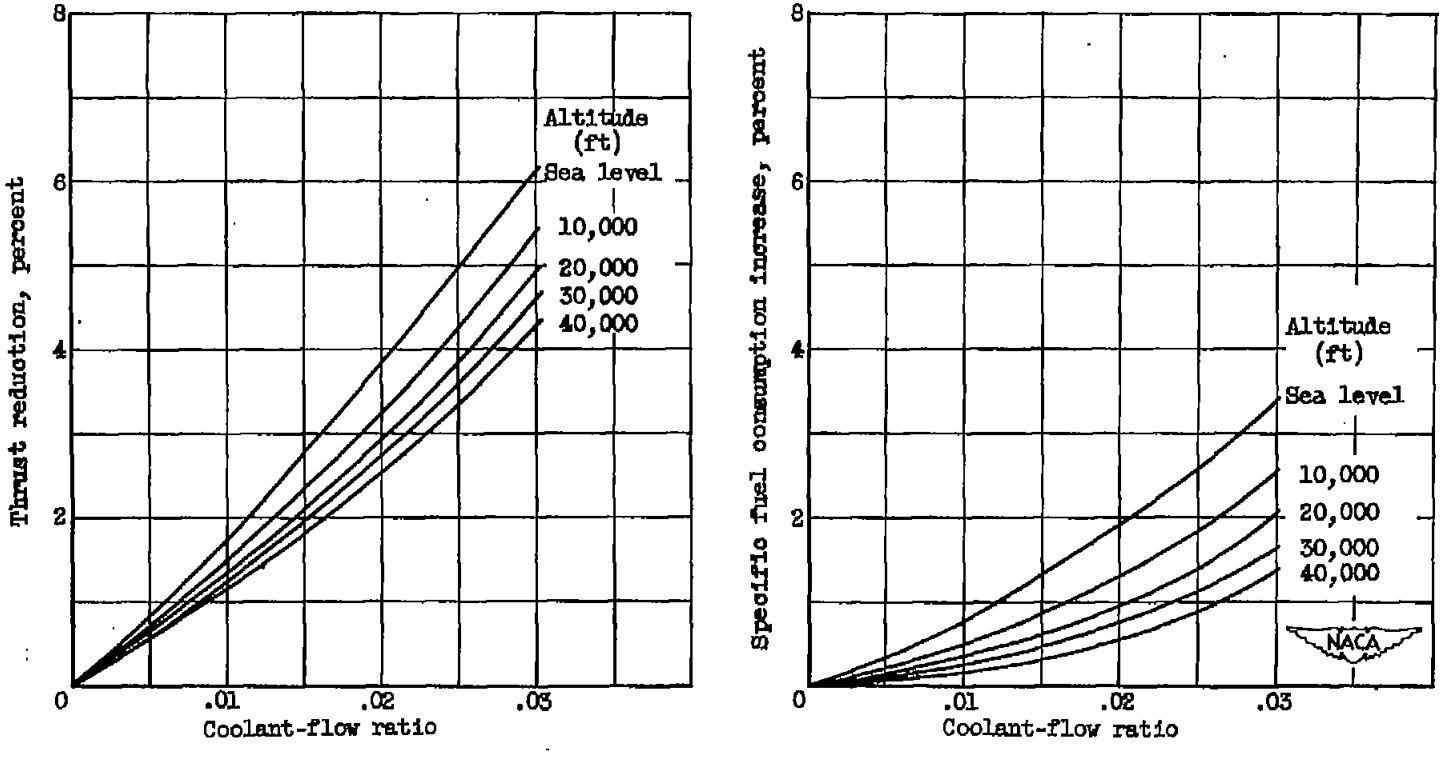


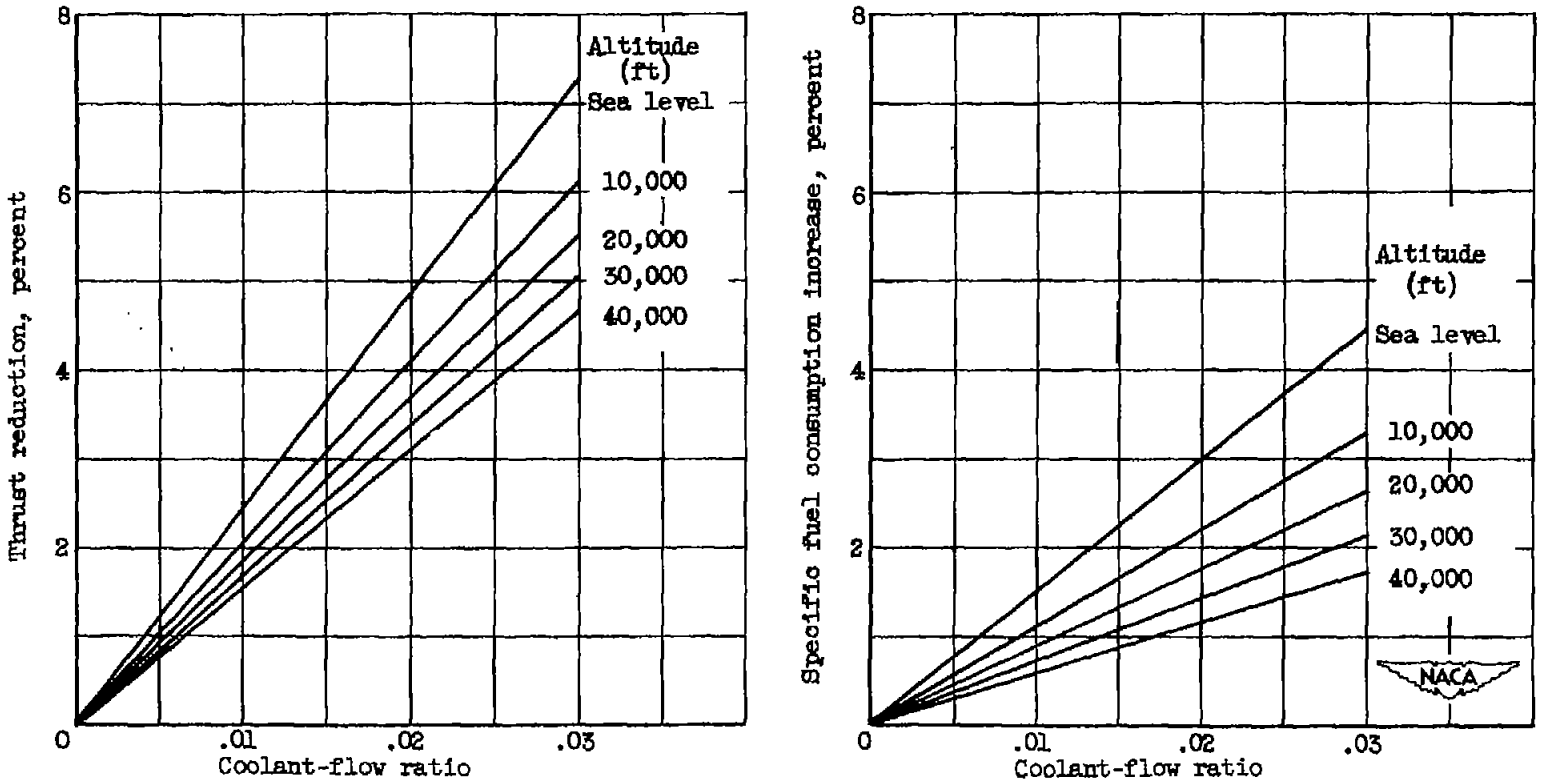
Figure 12. - Variation of jet-nozzle pressure ratio with coolant-flow ratio for turbojet engine with air-cooled finned turbine blades. Flight Mach number, 0.788; engine speed, 7600 rpm; turbine-inlet temperature, 1960° R.



(a) Percentage decrease in thrust.

(b) Percentage increase in specific fuel consumption.

Figure 13. - Percentage decrease in thrust and increase in specific fuel consumption with coolant-flow ratio for turbojet engine modified for air-cooled turbine rotor blades. Flight Mach number, 0.788; engine speed, 7600 rpm; turbine-inlet temperature, 1960° R; cooling air bled at required coolant pressure.



(a) Percentage decrease in thrust.

(b) Percentage increase in specific fuel consumption.

Figure 14. - Percentage decrease in thrust and increase in specific fuel consumption with coolant-flow ratio for turbojet engine modified for air-cooled turbine rotor blades. Flight Mach number, 0.788; engine speed, 7600 rpm; turbine-inlet temperature, 1960° R; cooling air bled from compressor discharge.

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