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RESEARCH MEMORANDUM

ANALYSIS OF FACTORS AFFECTING SELECTION AND DESIGN OF AIR-COOLED SINGLE-STAGE TURBINES FOR TURBOJET ENGINES

IV - COOLANT-FLOW REQUIREMENTS AND PERFORMANCE OF ENGINES USING AIR-COOLED CORRUGATED-INSERT BLADES

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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SUMMARY

The estimated minimum cooling requirements and related performance of turbojet engines equipped with high-performance single-stage turbines with air-cooled corrugated-insert blades were obtained for wide ranges of turbine-inlet temperature, tip speed, and hub-tip radius ratio for a flight Mach number of 2 at 50,000 feet. The turbine was the only component considered for cooling. The effects of stress-ratio factor, flight Mach number, altitude, turbine rotor impeller efficiency, and outside heat-transfer coefficient on cooling requirements were also studied.

For operation at Mach 2 at 50,000 feet, turbine-inlet temperature could possibly be increased to about 3000° and 3500° R for turbine tip speeds of 1700 and 1100 feet per second, respectively. For nonafterburning and afterburning engines with a turbine hub-tip ratio of 0.75, increasing turbine-inlet temperatures and tip speeds still seems beneficial in spite of the cooling requirements. The cooling problem in general becomes more severe as either flight Mach number or altitude increases, particularly for the higher turbine-inlet temperatures and tip speeds. These results suggest that the limits of application of the corrugated-insert blade will decrease with increasing flight Mach number and altitude. However, for the immediate future, the corrugated-insert blade is promising for flight speeds and altitudes higher than 2 and 50,000 feet, respectively. The turbine rotor stress-ratio factor, turbine rotor cooling-air-impeller efficiency, cooling-air-supply ducting losses, and variations in turbine rotor and stator outside heat-transfer coefficients may have quite serious effects on the cooling requirements. These effects are important considerations in design and research.

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INTRODUCTION

Turbojet-engine performance may be improved by using engine designs that require turbine cooling. In order to evaluate the effects of cooling on engine performance, the cooling requirements, as well as the effects of various factors on these requirements, must be determined. Such results are needed to establish emphasis in research and as a guide for design. Therefore, analyses were made at the NACA Lewis laboratory (1) to obtain the estimated minimum cooling requirements for high-performance air-cooled turbojet engines equipped with corrugated-insert turbine blades, (2) to reflect these cooling requirements in the performance of the air-cooled turbojet engine, and (3) to illustrate the factors affecting cooling requirements. These last factors were estimated, since accurate design values are presently unknown. This investigation was conducted for design-point operation of turbojet engines equipped with single-stage turbines.

Evaluations of both the uncooled-turbojet-engine performance and the general effects of cooling on performance are published in the literature. The changes in engine performance resulting from turbine cooling with compressor air bleed are analyzed in reference 1, and the performance potentials of single-stage turbojet engines utilizing turbine cooling are presented in reference 2. Wide ranges in engine design specifications are considered in both references for flight at a Mach number of 2.0 in the stratosphere. However, rather than considering the cooling requirements for a particular type of air-cooled turbine blade, a range of coolant flows was investigated.

As a result of recent developments (refs. 3 and 4), the air-cooled corrugated-insert blade is amenable to analytical treatment for determining cooling requirements. In addition, considerable research has been devoted toward developing fabrication techniques for this blade and testing experimental blades. An experimental corrugated-insert blade that has been operated in a standard turbojet engine is shown in figure 1(a). Experimental investigations conducted in modified turbojet engines to determine the cooling effectiveness of air-cooled corrugated-insert blades are reported in references 5 and 6. The corrugated-insert blade is a promising design because (1) the corrugated surfaces have high heat-transfer effectiveness, (2) there is freedom in selecting a corrugation geometry, (3) corrugated surfaces can serve as structural elements, and (4) an island insert permits corrugations of uniform height and thus a more uniform peripheral distribution of the cooling air. To some extent, the application of the corrugated-insert blade is compared with other types of air-cooled blades in reference 7. The cooling requirements of the corrugated-insert blade and the effects of various factors on cooling are computed herein for the engine designs presented in reference 2. The analytical procedures used to evaluate the cooling requirements of air-cooled turbines with corrugated-insert blades are presented in detail in references 3 and 4.

The purpose of this report, which concludes a series, is to present the estimated minimum cooling requirements and the resulting changes in performance of engines with single-stage turbines with air-cooled corrugated-insert blades. The possible limitations of this blade are also presented. A range of turbine-inlet temperatures from 2000° to 3500° R, turbine blade tip speeds of 1100 and 1700 feet per second (actual tip speeds as used throughout), and turbine hub-tip radius ratios of 0.60 and 0.75 were considered for a flight Mach number of 2.0 at a 50,000-foot altitude. In addition, typical variations in the corrugation geometry and coolant flows with flight Mach number, altitude, stress-ratio factor, pressure losses in cooling-air ducting and turbine rotor disk, and outside heat-transfer coefficients are presented.

ANALYTICAL PROCEDURES

General Considerations

The analytical procedures used to evaluate the cooling requirements of air-cooled turbines with corrugated-insert blades are presented in detail in references 3 and 4. Figures 1(b) and (c) show a cross-sectional view of a corrugated-insert blade and a typical corrugated section. The principal dimensions of the corrugation that may be varied in the selection of corrugation geometry are indicated in figure 1(c). These dimensions are the corrugation amplitude Y, the corrugation spacing m, and the corrugation thickness τ. References 2 and 8 present a general discussion of the procedures for computing the performance of turbojet engines equipped with single-stage air-cooled turbines and the turbine operating conditions required in evaluating the cooling requirements. Detailed procedures are given in reference 9. In addition, references 2 and 8 tabulate the engine performance and many of the operating conditions for the complete range of turbine design specifications considered in this analysis.

Schematic drawings of the turbojet engine and the turbine rotor cooling-air impeller are shown on figure 2. In some cases, three compressor bleed points were considered for the rotor blade cooling-air supply and two compressor bleed points for the stator blades. The relative positions of these bleed points are shown on figure 2(a). The exact locations actually vary for each engine design specification. Only one compressor bleed point was considered in the analysis for the engine performance results of reference 2. However, as shown in reference 1, the effect of cooling-air bleed point on engine performance is small. For this reason, the engine performance results of reference 2 were used herein, even though the compressor bleed point did differ somewhat for many of the cases studied.

The assumptions and constants used in the present analysis are presented in appendix A. Reference 2 gives the assumptions employed in the engine performance analysis. The present study and that of reference 2 are directed toward exploring the outer limits for single-stage turbines using turbine blade cooling. For this reason, the engine performance results are based on turbines with high turbine aerodynamic limits and high turbine-exit whirl in the interest of obtaining the highest turbine specific work and weight flow commensurate with the efficiencies assigned. Also, the present study was aimed toward determining the approximate minimum cooling requirements anticipated for the corrugated-insert blade without cooling the cooling air. In most cases, these calculated cooling requirements are lower than will be possible until improved design techniques are evolved. Of principal importance in this connection are the design stress-ratio factor, the cooling-air-supply ducting losses, the rotor cooling-air-impeller efficiency, and the blade metal taper from root to tip. Each of these variables is chosen so as to minimize the cooling requirements for the corrugated-insert blade. Their effects on cooling requirements are finally illustrated and discussed in subsequent sections, except for the blade metal taper ratio. The effect of the blade metal taper ratio from blade root to tip is small. A value of 0.50 was used for this ratio, since it represents the approximate lower limit for corrugated-insert blades that have been fabricated.

The stress-ratio factor, which is the ratio of the average allowable blade stress-rupture strength to the blade average centrifugal stress (ref. 10), was assumed to be 1.5. The magnitude of the stress-ratio factor is meant to include the effects of other stresses, such as bending, vibration, and thermal stresses, as well as to provide some margin of safety for the effects that construction may have on the blade material strength. Experimental data (ref. 10) show that a stress-ratio factor of about 2.3 is required for a particular blade constructed of carbon steel. However, limited unpublished data with higher temperature alloys indicate that smaller values of stress-ratio factor are possible through proper choice of materials and fabrication techniques.

The cooling-air-supply ducting losses were neglected in the present analysis. In addition, the turbine rotor cooling-air-impeller efficiency (defined as the ratio of the actual relative enthalpy difference to the ideal relative enthalpy difference between the rotor impeller inlet and the rotor blade inlet) was assumed to be the same as that for the main compressor, 0.83. Certainly, before these assumptions can be practically realized, good design techniques must be evolved through development work. The greater the cooling requirements, the more difficult the problem of reducing pressure losses. The efficiencies for a single cooling-air-impeller design operating over a range of conditions have been determined from test results but not reported. These unpublished data show impeller efficiencies as high as 0.50 for this early design.

Evaluation of Principal Heat-Transfer Parameters

Before the procedures of references 3 and 4 can be used, certain independent variables must be known or calculated. The majority of these variables are determined from the operating conditions and the design specifications of the turbojet engines being considered. The effective gas temperature $T_{g,e}$ can be evaluated from reference 11. The average Nusselt number from which the average outside heat-transfer coefficient
3644 $H_{o,av}$ is obtained is determined by use of the following correlation equation (ref. 12):

$$\frac{Nu_{av}}{Pr^{1/3}} = \overline{F}\,(Re_{av})^{z} \tag{1}$$

where perimeter l_o/π is the reference length in Nu_{av} and Re_{av}, and $\overline{F}$ and z are functions of the transition ratio and Euler number of the blade. (Symbols are defined in appendix B.) When an actual blade profile is available, the values of transition ratio and Euler number may be determined from the velocity distribution around the blade profile (ref. 12). In the present study, however, the actual blade profiles are unknown for the various engine designs reported in references 2 and 8, but values of turbine-inlet and -exit pressures and velocities are known.

For channel flow at high subsonic Mach numbers, the product of the static pressure and relative velocity is approximately constant. Therefore, since the turbine designs of references 2 and 8 have high subsonic Mach numbers corresponding to high blade loading, the product of the integrated average static pressure and relative velocity can be closely approximated by the product of the averages of the inlet and exit static pressures and velocities. Thus, the Reynolds number becomes

$$Re_{av} = \frac{p_{av} W_{av} \dfrac{l_o}{\pi}}{\mu_b g R T_b}$$

where p_{av} and W_{av} are the averages of the inlet and exit pressures and velocities.

The values of $\overline{F}$ and z (eq. (1)) were obtained from reference 13, wherein ten blade profiles, including both impulse and reaction blades, were analyzed and average values of $\overline{F}$ and z were evaluated. These values are $\overline{F} = 0.092$ and $z = 0.70$, and equation (1) becomes

$$Nu_{av} = 0.092\,(Re_{av})^{0.70}\,Pr^{1/3} \tag{2}$$

Equation (2) was used to calculate the value of $H_{o,av}$ for both turbine rotor and stator blades. The values of $H_{o,av}$ computed from equation (2) may be somewhat high for the reaction blading of the turbine stators, since the favorable pressure gradient may permit laminar flow over a large portion of the blade surface. This results in a lower value of z and a higher value of $\bar{F}$.

The gas properties in equation (2) were evaluated on the basis of the allowable blade temperature at the midspan of the rotor blades and of an average blade temperature for the stator blades. The maximum allowable blade temperature distributions for the rotor blades were evaluated from the 100-hour stress-rupture properties of a promising alloy A-286. Reference 3 discusses the procedures used for determining the maximum allowable blade temperature. The limiting allowable blade temperature of the stator blades was taken as 2000° R at the stator blade tip. This value is chosen as a conservative value for present uncooled stator blades.

Selection of Corrugation Geometry

In references 3 and 4, analytical procedures and charts are presented that permit the selection of a corrugation geometry which gives, for a specific application, the lowest possible coolant flow within limitations of the pressure levels through the engine and the blade pressure drop. The selection of a suitable corrugation geometry, however, also depends on the blade profile and the ease of fabricating the corrugation configuration. In general, small values of corrugation height or amplitude Y are required for air-cooled blades in order to provide trailing-edge cooling with minimum trailing-edge thickness. In addition, changes in Y have a larger effect on coolant-flow and pressure requirements than changes in m and τ. As a result, Y is the most important of the corrugation-geometry parameters, and the values of corrugation thickness τ and corrugation spacing m are secondary in the final selection of a corrugation geometry for an air-cooled blade. Consequently, following the suggestion of reference 3, pressure drops were calculated for various values of Y with assigned values of m and τ. In this way, the value of Y corresponding to the condition of calculated pressure drop equal to available pressure drop could be obtained for each combination of m and τ. Also, it was assumed that the required static pressure of the coolant at the rotor blade outlet was equal to the average gas static pressure across the rotor tip. The required coolant static pressure at the stator blade outlet was assumed equal to the gas static pressure downstream of the turbine rotor tip (see fig. 2(a)).

Rather than obtain coolant-flow requirements and the corresponding corrugation geometries for each of the single-stage turbojet-engine designs presented in reference 2, only the extreme design conditions were considered - that is, turbine-inlet temperatures of 2500°, 3000°, and 3500° R; actual turbine blade tip speeds of 1100 and 1700 feet per second; and turbine hub-tip radius ratios of 0.60 and 0.75. The coolant requirements are either zero or negligible at 2000° R.

RESULTS AND DISCUSSION

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For each value of m and τ, the calculated pressure drop equals the available pressure drop for only one value of Y. This condition results in either the minimum or approximately the minimum coolant flow for this combination of m and τ. However, changes in m and τ, in general, result in different solutions for Y and, therefore, different coolant-flow requirements. For this reason, the effects of m and τ must be considered for some designs, particularly for design conditions where it is desirable to reduce the coolant flow or the required value of Y. Therefore, for the present analysis, a number of combinations of m and τ are considered.

In order to aid in understanding the roles of m, τ, and Y, this section first illustrates their effects on the selection of a corrugation-geometry design. Then, the computed minimum coolant flows and the corresponding geometries for the corrugated-insert blade are summarized for the engine designs considered. From this summarization, the rotor and stator blade coolant flows and their geometry configurations are selected for each engine design and incorporated in the performance results. In this way, the possible limitations of the corrugated-insert blade used in single-stage air-cooled turbines operating at a flight Mach number of 2.0 at 50,000 feet are illustrated. Finally, the changes in cooling requirements for changes in stress-ratio factor, flight Mach number, altitude, impeller efficiency, and outside heat-transfer coefficient are shown for typical cases to illustrate their effects and importance in design and future research.

Influence of Corrugation Geometry on Coolant-Flow Requirements

Effect of corrugation amplitude. - Variations of the coolant flow and the blade-outlet coolant Mach number and static pressure with corrugation amplitude Y are shown in figure 3 for a representative turbine rotor blade with fixed values of corrugation thickness and spacing. The values of turbine-inlet temperature, tip speed, hub-tip radius ratio, and blade-inlet coolant pressure and temperature are also fixed.

As shown by figure 3, a decrease in Y results in a slight decrease in the coolant flow required to maintain maximum allowable blade metal temperatures. Variations in the required coolant flow may be considerably different from that shown on figure 3, depending upon the type of coolant flow (laminar, turbulent, or transition) and the amount of cooling required. The changes in the coolant flow and Y are reflected in the blade-outlet Mach number and coolant static pressure in the lower sections of figure 3. A decrease in Y results in a direct decrease in the coolant-flow area, which, in turn, results in an increase in the coolant Mach numbers. The higher Mach numbers and the higher pressure drops accompanying them combine to reduce the blade-outlet static pressure. Although the trends shown in figure 3 are representative of those for all conditions studied, the rate of change in the exit Mach number and static pressure may be somewhat different for other conditions because of a different variation in the required coolant flow. The minimum Y obtainable 3644 corresponds to the condition of the calculated coolant static pressure at the coolant passage outlet equal to the average gas static pressure across the rotor tip, as indicated by the intersection of the two curves in the bottom section of figure 3.

The trends shown by the curves in figure 3 are typical for both turbine rotor and stator blades, regardless of the engine design specifications considered. Because of the choking condition illustrated by the middle curve, extreme care must be used in some cases when selecting a corrugation geometry in order to provide a factor of safety.

For turbulent flow, the coolant-flow ratio will always decrease with decreasing Y. For laminar flow, however, the coolant-flow ratio in general increases with decreasing Y. Therefore, a reversal occurs for flow in the transition region. In general, then, the minimum value of Y will not correspond to the minimum value of the coolant-flow ratio for laminar flow. This condition may also exist for flow in the transition region. Consequently, these trends must be considered in the final selection of a corrugation geometry for a particular application. However, for the results presented herein, the value of minimum Y corresponding to the intersection point in figure 3 was determined. In all cases studied, the resulting values of the coolant-flow ratio were either the minimum or near the minimum values.

<u>Effect of corrugation spacing and thickness</u>. - Figure 4 shows for a representative turbine rotor design the effects of corrugation spacing m and thickness τ on the minimum allowable corrugation amplitude Y and the required rotor coolant flow C_R. These values of Y and C_R were obtained in a manner identical to that for the bottom part of figure 3. As shown in figure 4, an increase in m results in a decrease in Y and a rise in C_R. These trends are a result of the change in the coolant heat-transfer coefficient necessary to offset the change in the number of equivalent fins (or surface area). In addition, figure 4 shows that an increase in τ results in an increase in Y and a decrease in C_R. The principal effect of increasing τ is to increase the corrugation cross-sectional area normal to the heat flow. This, in turn, results in

decreased temperature gradients and thus improved effectiveness of the corrugated surfaces. When the value of τ is changed from 0.005 to 0.010 inch for constant m, Y is increased approximately 20 percent and C_R is decreased about 10 percent. In changing m from 0.05 to 0.02 for a constant τ, Y is increased about 25 percent and C_R is decreased almost 25 percent.

Although figure 4 is typical of the cases analyzed, the percentage variations in Y and C for values of m and τ may differ somewhat. In general, if the design conditions are such that Y tends to become excessively large, a small value of τ and a large value of m should be chosen. Such a choice also tends to decrease the blade weight and, therefore, blade stresses. Conversely, if coolant flow becomes the criterion, a large value of τ and a small value of m should be chosen.

Coolant-Flow Requirements and Performance Results

Tables I(a) and (b) summarize all the computed solutions of required corrugation geometries and their corresponding coolant flows for the turbine rotor and stator blades, respectively. Each corrugation-geometry combination presented in table I is within limitations of the pressure levels through the engine and the blade pressure drop and was selected by using a curve similar to the bottom curve of figure 3 for solving for the minimum value of Y. All results shown in table I are for single-stage turbines of turbojet engines operating at a flight Mach number of 2.0 at 50,000 feet. Because a large number of calculations are involved in obtaining each corrugation geometry and the corresponding coolant flow, every effort was made to minimize the number of engine and corrugation designs considered. For this reason, only extreme turbine blade tip speeds and hub-tip radius ratios were considered for each turbine-inlet temperature. Also, only enough values of m and τ and of compressor bleed points were considered to establish trends. The cases for which it was impossible to obtain adequate cooling designs are indicated in table I by "No solution." For a turbine-inlet temperature of 2000° R the cooling-air flows were either zero or negligible.

Once again, it should be emphasized that the coolant flows shown in table I are the approximate minimum coolant flows for the corrugated-insert blade used in single-stage turbines. Also, a comparison of the results presented in the tables and in figures 3 and 4 shows similar trends regardless of engine design specifications.

<u>Effect of compressor bleed point on rotor blade coolant flow.</u> - The performance results of reference 2 are based on the assumption that the rotor cooling air is bled at an intermediate stage of the compressor, so that the total work done on the cooling air is equivalent to the work required to compress it to the compressor-outlet conditions. The

corresponding bleed point for table I is indicated as compressor bleed point 1, and its relative location is shown in figure 2(a).

For all the engine designs considered, the rotor cooling air was initially supplied from compressor bleed point 1. In some cases, no solution was obtained with point 1, because there was not sufficient coolant pressure available at the blade inlet. When this happened, the compressor bleed point was changed to either a point midway between bleed point 1 and the compressor discharge (point 2) or to the compressor discharge (point 3). Although moving the compressor bleed point toward the compressor discharge increases the available blade-inlet coolant pressure, 3644
the blade-inlet coolant temperature also rises. Thus, for some engine designs, solutions cannot be obtained with compressor bleed point 3, because the blade-inlet coolant temperature is too high for adequate cooling.

In order to show better the effects of compressor bleed point, a variation of blade-outlet coolant static pressure $p_{a,o}$ and rotor coolant-flow ratio C_R with corrugation amplitude Y is presented in figure 5 for a particular engine design. With bleed point 1, no solution was obtainable even with $Y = 0.20$ inch for $m = 0.020$ and $\tau = 0.005$ inch. Changing the bleed point from 1 to 2, however, permitted a solution with $Y = 0.077$ inch. The blade-inlet total coolant pressure $p''_{a,in}$ and temperature $T''_{a,in}$ were 2700 pounds per square foot and 844° R for bleed point 1, and 4710 pounds per square foot and 1050° R for bleed point 2. When an attempt was made to change the bleed point to compressor-discharge conditions (compressor bleed point 3), no solution was obtained because of the high blade-inlet coolant temperature. Even though $p''_{a,in}$ increased to 11,600 pounds per square foot, which would be beneficial, $T''_{a,in}$ increased to 1324° R. Because $T''_{a,in}$ is only 62° R below the allowable blade temperature at the blade root for this design, cooling was impossible.

Along with the change in bleed point, however, a change in rotor coolant flow takes place. Figure 5 shows that, for the change in compressor bleed point from 1 to 2, the rotor coolant-flow ratio almost doubles for the engine conditions considered in this plot. The increase in rotor coolant flow is required to offset the increase in coolant temperature. The trends shown by the curves in figure 5 are duplicated by the results in table I. Possibly other compressor bleed points intermediate between the ones already considered would be more beneficial in the final selection of a corrugation geometry. The main point suggested by the results of this analysis is that, when selecting the most suitable air-cooled corrugated-insert blade for a specific engine design, the choice of the best compressor bleed point is extremely important.

Effect of compressor bleed point on stator blade coolant flow. - The performance results of reference 2 are based on the assumption that the stator cooling air is bled at the compressor discharge. Therefore, the turbine stator cooling requirements for this analysis were evaluated with compressor-discharge bleed for each engine design studied. These results, tabulated in table I(b), are indicated by compressor bleed point 3. The possibilities of using either ram air or air bled from an intermediate compressor stage were also considered. In general, these considerations revealed little or no advantage of bleeding ahead of compressor discharge for the engine designs studied.

The resulting cooling requirements for ram-air bleed are not listed in table I(b). In all cases, however, the minimum required corrugation amplitude was about 0.150 inch or greater; in general, the higher the turbine-inlet temperature, the higher the corrugation amplitude. The values of corrugation amplitude obtained with ram air are considered to be excessive for the trailing edge of air-cooled turbine blade designs and are, therefore, impractical. In most cases, even the possible reductions in cooling-air requirements by the use of ram air rather than compressor-discharge air are small, although the cooling-air temperature is lower for ram air.

The cooling requirements for bleed from an intermediate compressor stage, which are listed in table I(b) for three engine designs, are indicated by compressor bleed point 2. The position of this bleed point was selected arbitrarily about two-thirds of the way through the compressor. Since the purpose of considering this bleed point is to establish the trends, the specific bleed point is unimportant. In the interest of minimizing the calculations, only the higher turbine-inlet temperatures and turbine hub-tip radius ratios were considered. Comparison of the results for bleed points 2 and 3 shows that, as the bleed pressure is reduced, the required corrugation amplitude increases significantly. The maximum value is obtained for ram air as previously discussed. For the cases in which the two bleed points are shown in table I(b), for a turbine blade tip speed of 1100 feet per second the increase in Y is accompanied by a relatively small change in coolant flow. For these cases, the advantage of the early bleed seems to be offset by the change in amplitude required. This is particularly true for the design with a turbine-inlet temperature of 3500° R, since a value of 0.150 inch for Y seems excessive. For the single case with a turbine tip speed of 1700 feet per second, changing the bleed point resulted in an increase in both Y and the cooling-air requirements, both of which are undesirable. Intermediate bleed was not considered for a turbine-inlet temperature of 2500° R, since the possible changes in cooling-air requirements were small. It is only for these cases that relatively large changes in Y might be tolerated. Benefits of bleeding ahead of the compressor discharge were not realized for turbine stator cooling as for turbine rotor cooling, possibly for two reasons: (1) The stator effective gas temperature is higher than that for the rotor, and (2) the impeller does additional work on the rotor coolant after compressor bleed.

For either compressor bleed point 2 or 3, the cooling air can be discharged into the tail pipe. However, for ram-air bleed, the cooling air must be discharged to a region of lower pressure than available in the tail pipe. This adds undesirable complications to the engine design.

Coolant-flow range. - The most desirable corrugation design for each of the engine designs may now be selected from table I. In many cases, this selection may be quite arbitrary, since the required coolant flows and corrugation amplitude are either small or essentially unaffected by the corrugation spacing and thickness. In general, however, the selection 3644
should be based on minimum coolant flows with acceptable values of corrugation amplitudes. The coolant-flow requirements selected for this report are as follows:

Turbine tip speed, U_t, ft/sec	Turbine hub-tip radius ratio, r_h/r_t	Turbine-inlet temperature, T_4', $^{\circ}$R					
		2500		3000		3500	
		Rotor coolant-flow ratio, C_R	Stator coolant-flow ratio, C_S	Rotor coolant-flow ratio, C_R	Stator coolant-flow ratio, C_S	Rotor coolant-flow ratio, C_R	Stator coolant-flow ratio, C_S
1100	0.60	0.014	0.025	0.036	0.046	0.078	0.077
	.75	.012	.029	.027	.064	.074	.103
1700	0.60	0.032	0.036	0.100	0.060	-----	-----
	.75	.021	.049	.048	.104	-----	-----

In making these selections, emphasis was placed on choosing small values of corrugation amplitude, except where coolant-flow changes are large. For the rotor blades, the required corrugation amplitude never exceeded 0.090 inch. For the stator blades, the required corrugation amplitude exceeded 0.10 inch for only one case, $T_4' = 3500^{\circ}$ R, $U_t = 1100$ feet per second, and $r_h/r_t = 0.60$. A plot showing the band of total cooling-air requirements, which includes all values given in the preceding table, is shown on figure 6. For each given turbine blade tip speed, the total cooling-air requirements are only slightly affected by the turbine hub-tip radius ratio. The lower limiting line represents the minimum cooling-air requirements obtained for 1100-foot-per-second turbine blade tip speed. The upper limiting line represents the maximum cooling-air requirements obtained for U_t of 1700 feet per second.

A real limit in the application of the corrugated-insert blade is established by the trailing-edge thickness, if the corrugation is to extend into the trailing-edge region. For present designs, the trailing-edge thickness for uncooled turbine blades is considerably below 0.10 inch. The extent that this dimension can be relaxed to accommodate air-cooling designs is as yet unknown. The extent the trailing edge may transfer its load to the central portion of the blade has not been established, nor, therefore, the extent it must be cooled. For the present case, in which only average requirements are being considered rather than local requirements around the periphery of the blade, this limit is arbitrarily set at a value of corrugation amplitude of about 0.10 inch. The wavy line representing the limit on figure 6 approximates this condition. Table I can be used for estimating limitations for the corrugated-insert blade for other values of the maximum allowable corrugation amplitude. It should be realized that the cooling requirements for the corrugated-insert blade may be reduced if either the blade metal strength is increased or the cooling-air temperature is reduced without expanding to a lower pressure. Both of these factors are presently being investigated.

Effect of coolant-flow ratio on engine performance. - The variation of the performance of both nonafterburning and afterburning (afterburning to 3500° R) single-stage-turbine engines operating at a flight Mach number of 2.0 at 50,000 feet is shown in figures 7(a) and (b) for hub-tip radius ratios of 0.75 and 0.60, respectively. In all cases the compressor pressure ratio and engine mass flow are the maximum values obtainable for single-stage turbines with the aerodynamic limits imposed (see ref. 2). These figures, which cover a complete range of turbine-inlet temperatures and tip speeds, are presented for zero coolant-flow ratio and the estimated minimum coolant-flow ratios required with corrugated-insert air-cooled blades in the turbine rotor and stator. The cooling requirements were taken from the table of the preceding section. Because the compressor pressure ratios and the engine weight flows are maximum values for the conditions imposed, while the coolant flows are the estimated minimum values, these performance results represent the upper limit expected for engines equipped with single-stage turbines. With each value of coolant-flow ratio, the engine performance was evaluated in terms of thrust and thrust specific fuel consumption from the results presented in table I of reference 2. As previously discussed, effects of compressor bleed point on engine performance were neglected, since reference 1 shows these effects to be small.

The engine performance results are presented in figure 7 in terms of the total thrust per unit of turbine frontal area (as based on turbine weight-flow capacity) and the thrust specific fuel consumption. In general, the effects of bleeding cooling air from the compressor are a decrease in thrust and an increase in thrust specific fuel consumption for both the nonafterburning and afterburning engines. However, the effect of cooling on the thrust specific fuel consumption is considerably less for the nonafterburning than for the afterburning engine, because the coolant reduces the exhaust-gas temperature. In fact, at high turbine-inlet temperatures, cooling has little or no effect on the thrust specific fuel consumption of the nonafterburning engine.

As shown by figure 7, little improvement in engine performance is possible by increasing the turbine tip speed (or compressor pressure ratio) for a turbine-inlet temperature of 2000° R. Large gains in thrust per unit of turbine frontal area can be obtained, however, by a simultaneous increase in the turbine-inlet temperature and the turbine blade tip speed for either the nonafterburning or the afterburning engine. The changes in thrust specific fuel consumption, however, are different for the two types of engine, decreasing significantly for the afterburning engine as either the turbine-inlet temperature or the turbine blade tip speed increases. For increasing blade tip speed, this effect is due to the increasing compressor pressure ratio caused by the increased turbine work capacity. However, for the nonafterburning engine, as the turbine-inlet temperature rises at constant turbine blade tip speed, the thrust specific fuel consumption increases. In order to maintain constant thrust specific fuel consumption as the turbine-inlet temperature is elevated, the turbine blade tip speed must also be increased to increase the compressor pressure ratio.

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In general, as was shown in figure 6, an increase in the turbine-inlet temperature or the turbine blade tip speed results in larger cooling requirements for the corrugated-insert blade. As illustrated by figure 7, the effect of cooling air is to shift the performance toward lower thrust and higher thrust specific fuel consumption. However, it is important to notice that the effects of the actual cooling requirements on the thrust specific fuel consumption of the nonafterburning engine diminish as the turbine-inlet temperature rises. For the turbine hub-tip radius ratio of 0.60 (fig. 7(b)) at a turbine-inlet temperature of 3000° R and at tip speeds of 1100 and 1700 feet per second, the increase in thrust specific fuel consumption due to cooling for the nonafterburning engine is only approximately 1 and 3 percent, while the corresponding reduction in thrust per unit of turbine frontal area is between approximately 8 and 25 percent. However, for the afterburning engine with a turbine-inlet temperature of 3000° R and an afterburner-outlet temperature of 3500° R, the thrust specific fuel consumption increases by approximately 5 and 8 percent as the coolant-flow ratio changes from zero to the required amount, while the thrust per unit of turbine frontal area is reduced by about 3 and 15 percent for turbine blade tip speeds of 1100 and 1700 feet per second, respectively.

Figure 7 illustrates the large net performance gains based on the aerodynamically limited turbine possible by increasing the turbine-inlet temperature and tip speed of a single-stage turbine with corrugated-insert air-cooled blades. Even for the nonafterburning engine, it should be possible to maintain the thrust specific fuel consumption at present design value while obtaining significant increases in thrust by cooling the turbine. This requires turbine-inlet temperatures somewhat higher than 2000° R and an increase in the turbine blade tip speed. For the afterburning engine with a turbine hub-tip radius ratio of 0.60, only marginal improvements in the performance are shown for turbine-inlet temperatures above 2500° R. For a turbine hub-tip radius ratio of 0.75, the gains for turbine-inlet temperatures above 2500° R are increased.

An estimation of the limiting operating line for the air-cooled corrugated-insert blade limited to an amplitude of approximately 0.10 inch as previously discussed is indicated in figure 7 by the wavy lines. These limits are only applicable for direct compressor bleed. The limits may be extended by reducing the coolant temperature or increasing the
3644 blade metal strength. For the nonafterburning engines operating at a turbine tip speed of 1700 feet per second, this limit is estimated to occur at or near a turbine-inlet temperature of 3000° R. As the turbine tip speed is decreased, the allowable turbine-inlet temperature can be increased. At a turbine tip speed of 1100 feet per second, the limiting temperature is increased to about 3500° R.

It is well to realize that the values of engine thrust presented in figure 7 depend on the mass-flow capacity of the turbine. Should other components limit the engine mass-flow capacity, the effect would be a lower thrust per unit of engine frontal area. This effect is discussed and illustrated in references 2 and 8.

Effects of Stress-Ratio Factor, Flight Mach Number, Altitude, Impeller Efficiency, and Outside Heat-Transfer Coefficient

The cooling requirements presented in table I and the performance results presented in figure 7 are in most cases based on optimistic assumptions. Consequently, the results present the outer limits that can be expected with single-stage air-cooled turbines using corrugated-insert blades. In this way, the results present the goals that may be sought by research with single-stage turbines using turbine blade cooling. These goals can be reached only by improving the design of the cooling system. It is therefore important to establish the effects of the factors influencing the cooling requirements to illustrate what improvements are more desirable and where research can best be emphasized. For this reason, some of the primary factors affecting cooling have been investigated for illustrative cases and the results are presented in the following paragraphs. In all cases the corrugated designs are within the limitations of the pressure levels through the engine and the blade pressure drop. In addition, it is realized that flight Mach numbers and altitudes other than 2 and 50,000 feet, respectively, are of interest. For this reason, calculations were also made to illustrate the effects of flight Mach number and altitude on the cooling requirements.

Stress-ratio factor. - The effects of stress-ratio factor on corrugation amplitude and coolant flow are shown in figure 8 for three engine designs for a flight Mach number of 2.0 at 50,000 feet. The cooling requirements presented in table I(a) and those incorporated in the performance results are based on a stress-ratio factor of 1.5 for the rotor blades. However, as previously pointed out, experimental tests have shown that stress-ratio factors as high as 2.3 are required for a particular blade. Therefore, a range of stress-ratio factor from 1.5 to 2.3 is considered in figure 8. For a turbine-inlet temperature of 3000° R and a turbine blade tip speed of 1700 feet per second, corrugated-insert blade designs could not be obtained for the higher stress-ratio factors. 3644

For a turbine-inlet temperature of 2500° R and a tip speed of 1700 feet per second, an increase in stress-ratio factor from the assumed value of 1.5 to 2.3 results in an increase in corrugation amplitude from 0.032 to 0.059 inch, or about 85 percent, with a corresponding increase in coolant-flow ratio from 0.024 to 0.048, or about 100 percent. The amplitude is necessary in order to pass the required coolant flow within the prescribed pressure limitations without choking. An increase in T_4' to 3000° R with a reduction in U_t to 1100 feet per second results in very little change in Y and no change in coolant flow for a change in stress-ratio factor. The required coolant flow does not increase in this case because the coolant is either in the laminar or transition region where the minimum coolant flow does not exactly correspond to minimum Y, as was discussed thoroughly in connection with figure 3. However, for T_4' of 3000° R and U_t of 1700 feet per second, the change in Y and coolant flow is again significant for the corresponding increase in stress-ratio factor from 1.5 to 2.0. For this case, the corrugation amplitude increases from 0.076 to 0.138 inch, while the coolant-flow ratio increases from 0.040 to 0.070.

The difference between the cooling-air temperature at the blade root and the allowable blade temperature essentially accounts for the different effects of stress-ratio factor at the two tip speeds. As the tip speed is raised, the compressor pressure ratio increases directly with the increase in turbine work capacity. This is reflected in a higher cooling-air temperature at the bleed condition. In addition, the increase in the tip speed causes higher blade stresses and therefore lower allowable blade temperatures. Both these effects combine to reduce the difference between the allowable blade temperature and the cooling-air temperature and thus cause a more critical design condition. Thus, a change in allowable blade temperature has a large percentage effect on C and Y at the high tip speeds.

It is expected that the results on figure 8 are typical of those for other conditions. From these few cases, it may be concluded that the effect of changing stress-ratio factor from 1.5 to 2.3 is quite significant on the corrugated-insert blade for engines with high tip speeds. It should also be realized that the more severe stresses for lower turbine hub-tip radius ratios may further increase this effect. Until experimental verification of stress-ratio factors is available for the corrugated-insert blade operating under various conditions, however, the blade designer should be conservative in choosing a stress-ratio factor.

Flight Mach number. - The foregoing results have all been presented for a flight Mach number M of 2.0 at a 50,000-foot altitude. The effect of a change in M on rotor blade corrugation amplitude and coolant-flow ratio for two engine designs operating with a turbine-inlet temperature of 3000° R and a turbine hub-tip radius ratio of 0.75 at 50,000 feet is shown in figure 9.

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In order to hold a constant turbine-inlet temperature and tip speed over the range of flight Mach number considered, the compressor pressure ratio decreases (for aerodynamically limited turbines) and the outside heat-transfer coefficient increases at constant altitude as M changes from 1.65 to 2.80. The conditions chosen to illustrate the effects of Mach number change are considered typical of those expected for future air-cooled engine designs. The design conditions used for computing the results shown in figure 9 were obtained by generalizing the engine results presented in references 2 and 8. Thus, the M range shown was determined by limits of the generalized engine results. Furthermore, it was assumed that the values of $\overline{F}$ and z of the outside heat-transfer equation (eq. (1)) are independent of Reynolds number. The validity of this assumption for turbine blades is indicated by the linear heat-transfer correlation presented in reference 14.

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For the engine with U_t of 1100 feet per second, an increase in M from 1.65 to 2.80 results in a decrease in Y from 0.0855 to 0.0485 inch, accompanied by a slight change in the required coolant-flow ratio. For U_t of 1700 feet per second, however, Y first decreases and then increases as M rises, while the coolant-flow ratio increases steadily. The difference in the behavior of the cooling requirements is due both to differences in the coolant-flow region and the coolant temperature rise. In both cases, as the Mach number rises, the coolant flow becomes more turbulent. For U_t of 1100 feet per second, this effect essentially compensates for the rise in coolant-air temperature. At U_t of 1700 feet per second, however, the coolant-air temperature has a larger effect and the coolant-flow ratio increases. It should be pointed out that the engine mass flow increases with M because of the higher ram pressures. Consequently, the actual coolant mass flow increases much more than the coolant-flow ratio shown in figure 9.

The effect of compressor bleed point was investigated for only one of the engine design conditions considered in figure 9. However, for this example, the importance of bleed point with changing flight Mach number is illustrated. The curves presented in the figure were constructed with the use of compressor bleed point 1 for all values of M. The cooling conditions become less critical as the flight Mach number is reduced because of the lower ram temperatures. However, for this fixed compressor bleed point, as M is reduced between 2.0 and 1.65, the required value of Y increases rapidly. Therefore, at M of 1.65, the rotor bleed point was changed from 1 to 2 for U_t of 1100 feet per second. These results are shown by the circled points. Although the coolant-flow ratio was not affected much, Y was reduced approximately 50 percent. This is evidence of the tendency for the best rotor bleed point to move toward the compressor discharge as the flight Mach number is reduced. For engine operation over a range of flight Mach numbers, a compromise may be required in the design of the engine coolant system if a single bleed point is used. Although figure 9 is only meant to illustrate the effects of flight Mach number on corrugation geometry and coolant flow, it points out the necessity of investigating these effects in the design of an air-cooled engine.

<u>Altitude</u>. - The effect of a variation in altitude on corrugation amplitude and coolant-flow ratio for both the turbine rotor and stator blades is shown in figure 10. The engine design conditions chosen for this illustration are the same as used in figure 9; that is, a turbine-inlet temperature of 3000° R, hub-tip radius ratio of 0.75, and turbine tip speeds of 1100 and 1700 feet per second. In addition, the results are for operation at a flight Mach number of 2.0. The engines of figure 10 have the same pressure and temperature ratios. A change in altitude changes only the ambient pressure, which is reflected throughout the engine, and thus the outside heat-transfer coefficients. As before, it was assumed that the values of $\overline{F}$ and z of the outside heat-transfer equation (eq. (1)) are independent of Reynolds number and thus altitude.

The coolant supply for the rotor blades was taken from compressor bleed point 1, and that for the stator blades from the compressor discharge. As the altitude changes from 35,000 to 70,000 feet, the blade-inlet coolant pressure is reduced considerably for both rotor and stator blades. For example, an altitude change from 35,000 to 50,000 feet results in a reduction in rotor blade-inlet coolant pressure of approximately 100 percent, whereas a reduction of about 160 percent occurs for a change from 50,000 to 70,000 feet. The blade coolant inlet temperature, however, remains constant for a change in altitude from 35,000 to 70,000 feet, because the ambient temperature is constant in the stratosphere. Therefore, the increase in corrugation amplitude Y as the altitude is raised is due to the considerable reduction in blade-inlet coolant pressure. This change in pressure is reflected in the Reynolds numbers and, therefore, in the heat-transfer coefficients. The fact that curves differ for the rotor and stator blades can only be explained by the differences in the coolant-flow regions in each case. As discussed previously, the value of Y obtained for the rotor blades may be reduced by changing the rotor bleed point from 1 toward the compressor discharge (higher blade-inlet coolant pressures). However, the best bleed point is not expected to vary appreciably with altitude as it does with flight Mach number. For this reason, the effect of bleed was not studied here. As the altitude increases from 35,000 to 70,000 feet, the coolant-flow ratio increases about 30 to 40 percent for the rotor blades, and as much as 75 percent for the stator blades.

The results shown in figure 10 indicate that changes in the altitude may cause significant changes in the required corrugation amplitudes and coolant-flow ratios. These changes certainly affect the limits for application of the corrugated-insert blade. In addition, they illustrate the need for considering the cooling design at several altitudes if the engine design is to operate over a range of altitudes. Although the cooling requirements shown in figure 10 increase with altitude, the trends may be reversed, depending on the coolant- and gas-flow regions.

<u>Impeller efficiency and ducting losses</u>. - A turbine rotor cooling-air-impeller efficiency of 0.83 was used in all of the preceding results. In addition, all cooling-air ducting losses from the compressor bleed point to the inlet of the impeller were neglected. New design techniques are certainly required before either of these conditions can be satisfied. For this reason, an analysis was made to illustrate the importance of turbine rotor impeller efficiency and pressure losses in the cooling ducting system.

The effects of impeller efficiency on corrugation amplitude and coolant-flow ratio are shown in figure 11 for turbine-inlet temperatures of 2500° and 3000° R and for turbine tip speeds of 1100 and 1700 feet per second. Once again, a flight Mach number of 2.0 at 50,000 feet was used, with a turbine hub-tip radius ratio of 0.75. For both turbine tip speeds at 2500° R and for the 1100-foot-per-second tip speed at 3000° R, the corrugation amplitude increases gradually as the efficiency is decreased. For the complete efficiency range covered (0.05 to 0.83), this change in amplitude varies from about 70 to 120 percent.

At the more severe condition of 1700-feet-per-second tip speed and 3000° R turbine-inlet temperature, the effect of impeller efficiency is even greater. In this case, a change in impeller efficiency of 40 percent results in about a 100-percent change in corrugation amplitude. In addition, this change results in excessive amplitudes for air-cooled blade designs. Thus, it is apparent that the impeller efficiency has a large effect on the corrugation geometry.

The effect on the cooling-air requirement is less critical. In fact, for these examples, the coolant-flow ratio, in general, decreases as the impeller efficiency decreases. This behavior of the coolant-flow ratio with changes in impeller efficiency is a result of the nature of the coolant flow. In these cases, the flow varies from that in the transition region to flow in the laminar region. For other conditions, the illustrated trends might be somewhat different.

The effects of cooling-air ducting pressure losses on the corrugation amplitude are shown on figure 12 for the same conditions as in figure 11. The total-pressure losses plotted as the abscissa include all frictional losses in the cooling-air supply system from the compressor bleed points to the blade inlet. The ideal pressure rises due to compression in the turbine rotor impeller are accounted for. Therefore, the 3644
point of zero frictional pressure loss corresponds to the conditions of no ducting losses prior to compression in the impeller and 100-percent impeller efficiency. If all of the losses are sustained in the impeller, the impeller efficiencies for the complete range of losses in figure 12 cover the range of impeller efficiencies for the respective curves in figure 11. Therefore, the abscissa of figure 12 was obtained by transforming the impeller efficiencies of figure 11 to the pressure losses of figure 12. In this way, the pressure losses can be considered to be distributed over the entire cooling-air ducting system. It is apparent from figure 12 that large losses in the cooling-air ducting system are undesirable. In addition, these losses appear to have as large an effect (or larger) at the lower tip speeds as at the higher tip speeds, even though the engine design is less critical. In all cases, the ideal compression in the turbine rotor impeller is relatively high. That is, the actual cooling-air pressure rises for the conditions shown in figure 12 in spite of the frictional pressure losses in the ducting system.

Outside heat-transfer coefficient. - Equation (2) was used to compute the outside heat-transfer coefficients for this analysis. A comparison of equation (2) with the extreme values obtained experimentally from static tests with cascades of turbine blades (ref. 14) is presented in figure 13. The outside heat-transfer coefficient is directly proportional to the ordinate. Each of the curves obtained from reference 14 is extrapolated to the Reynolds number range covered in this analysis. Curves A and C result in the maximum and minimum heat-transfer coefficients obtained experimentally with cascades of impulse and reaction turbine blades. Figure 13 also includes the curve (D) for laminar flow over a flat plate; this curve represents the theoretical minimum obtainable outside heat-transfer coefficient.

Although the heat-transfer relation used in this report was fixed by necessity, it is realized that each design considered would possess a different relation. In addition, the accuracy of computed heat-transfer coefficients for a specific turbine design is presently limited. The estimated maximum changes in outside heat-transfer coefficient which might occur are illustrated by the changes to either the impulse or reaction blading results shown in figure 13. For the designs considered, the average Reynolds number varied from approximately 200,000 to 1,500,000 for both stator and rotor blades. The higher Reynolds numbers correspond to the higher turbine-inlet temperatures and tip speeds. Thus, high Reynolds numbers are in the direction of increasing cooling requirements. For this range of Reynolds number, the increases in outside heat-transfer coefficients from curve B to curve A range from about 20 to 30 percent. In addition, a change to reaction blading (curve B to curve C) results in decreases from approximately 15 to 40 percent.

Calculations were made to illustrate the minimum changes in coolant-
3644 flow requirements for a particular corrugation geometry due to changes in the outside heat-transfer coefficient (fig. 14). The maximum range for the static cascade test results of figure 13 is also shown. Because the minimum percentage change in coolant flow depends somewhat on specific engine design conditions, the results of figure 14 were obtained by using mean conditions for the complete range of engine designs considered in this analysis, resulting in an approximation of less than 5 percent for the results shown. Although the actual changes in coolant flows required are larger than those shown in figure 14, this difference is not expected to be large.

From figure 14 it is apparent that the outside heat-transfer coefficient has a significant effect on the coolant-flow requirements. If the coolant flow is turbulent, the change in the minimum coolant flow is about the same as the change in the outside heat-transfer coefficient. However, for laminar flow, the change in the minimum coolant flow may be as much as twice the change in the outside heat-transfer coefficient. For the cases studied in this analysis, the coolant flow was more commonly in or near the laminar region. If the design assumptions were more conservative, however, the coolant flow would approach the turbulent condition. Regardless of coolant-flow region, however, the results of figure 14 were computed with the assumption of no change in corrugation geometry and, therefore, without considering the pressure requirements. Thus, an increase in coolant flow due to an increase in outside heat-transfer coefficient means a necessary increase in corrugation amplitude in order to pass the coolant flow. Also, the change in corrugation amplitude may, depending upon the coolant-flow region, cause an additional coolant-flow increase.

CONCLUDING REMARKS

This study has been directed toward exploring the outer limits for single-stage air-cooled turbines using corrugated-insert blades and toward illustrating the importance of variables affecting turbine cooling. The results, therefore, are intended to be useful in establishing possible goals for research as well as in illustrating the importance of variables that must be considered to assure a good and dependable corrugated-insert blade. The engine performance results and the cooling requirements of the corrugated-insert blade are in general better than can be expected without improved design techniques. In addition, the turbine is the only component considered for cooling.

The engine performance results and the corresponding cooling requirements show that the performance can be improved considerably above that of present designs by increasing the turbine-inlet temperature and the turbine blade tip speeds through the use of turbine cooling with corrugated-insert blades. The beneficial effect of increasing the turbine blade tip speed is derived from the increased turbine work capacity, which increases both the compressor pressure ratio and the turbine weight flow. For engine operation at a flight Mach number of 2.0 at a 50,000-foot altitude, the turbine-inlet temperature could possibly be increased to about 3000° and 3500° R with the corrugated-insert blade for turbine blade 3644 tip speeds of 1700 and 1100 feet per second, respectively. For the nonafterburning engines and the afterburning engines with a turbine hub-tip radius ratio of 0.75, such increases still seem beneficial in spite of the cooling requirements. This, however, can only be speculative until such results are applied to an actual aircraft. For the afterburning engine with a turbine hub-tip radius ratio of 0.60, the performance improves only slightly if the temperature is increased above 2500° R because of cooling losses.

The effect of flight Mach number and altitude on the performance results was not illustrated. However, the changes in cooling requirements with flight Mach number and altitude were studied. For the designs considered, the cooling problem in general becomes more severe as either flight Mach number or altitude increases. This is particularly true for the higher turbine-inlet temperatures and tip speeds. These results suggest that the limits of application of the corrugated-insert blade will decrease with increasing flight Mach number and altitude. The principal problem seems always to involve the need for reducing the corrugation amplitude. However, for the cases of interest for the immediate future, the corrugated-insert blade still holds promise for flight speeds and altitudes higher than 2.0 and 50,000 feet, respectively.

The effects of the turbine rotor stress-ratio factor, turbine rotor cooling-air-impeller efficiency, and cooling-air-supply ducting losses, and variations in the turbine rotor and stator outside heat-transfer coefficients may be quite significant. Only the individual effects of these variables are illustrated, but from these individual effects it becomes apparent that the combined effects may be quite serious with regard to the cooling problem and, therefore, represent an important problem in research. At a turbine-inlet temperature of 2500° R, the cooling requirements would probably be acceptable with the use of design variables that are presently more realistic than those used in this analysis. An increase in turbine-inlet temperature to 3000° R, however, will permit design variables only slightly more conservative than those used herein.

In all cases, it is desirable to predict accurately and to reduce the turbine rotor stress-ratio factor, cooling-air ducting losses, and outside heat-transfer coefficients.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, March 17, 1955

APPENDIX A

ASSUMPTIONS AND CONSTANTS

The following assumptions were made in this analysis:

(1) The outside heat-transfer coefficient and effective gas temperature are constant chordwise and spanwise on the blade and are equal to the midspan values. 3644

(2) The mean coefficient $\overline{F}$ and exponent z in equation (1) remain constant as the Reynolds number varies.

(3) An average of the inlet and exit gas static pressure and relative velocity product is used in the Reynolds number of equation (1).

(4) The same values of $\overline{F}$ and z are used in calculating the outside heat-transfer coefficients for the turbine rotor and stator blades.

(5) The corrugation geometry is constant chordwise and spanwise.

(6) The ratio of turbine rotor blade metal area at the tip to that at the root is equal to 0.50.

(7) A stress-ratio factor of 1.5 is used for the turbine rotor corrugated-insert blade.

(8) The limiting allowable blade temperature of the stator blades is taken as 2000° R at the stator blade tip.

(9) The required coolant static pressure at the rotor blade outlet is equal to the average gas static pressure across the rotor tip.

(10) The required coolant static pressure at the stator blade outlet is equal to the gas static pressure downstream of the turbine rotor tip.

The following values of constants were employed herein:

Compressor adiabatic efficiency	0.83
Combustion efficiency	0.95
Turbine adiabatic efficiency	0.83
Impeller efficiency	0.83
Afterburner combustion efficiency	0.88
Afterburner-exit stagnation temperature, °R	3500
Tail-pipe nozzle efficiency	0.95

APPENDIX B

SYMBOLS

The following symbols are used in this report:

A_T	turbine frontal area, sq ft
C	coolant-flow ratio (ratio of turbine cooling air to compressor weight flow)
c_p	specific heat at constant pressure, Btu/(lb)(°F)
F	thrust, lb
$\overline{F}$	mean coefficient in eq. (1)
g	standard acceleration due to gravity, 32.174 ft/sec^2
H_o	outside heat-transfer coefficient, Btu/(sec)(sq ft)(°F)
k	thermal conductivity, Btu/(sec)(ft)(°F)
l_o	outside perimeter, ft
M	flight Mach number
M_o	outlet cooling-air Mach number relative to blade
m	corrugation spacing, in.
Nu_{av}	average Nusselt number of gas, $\frac{H_{o,av} l_o/\pi}{k_b}$
Pr	Prandtl number of gas, $c_{p,b}\mu_b g/k_b$
p	static pressure, lb/sq ft abs
p"	total pressure with respect to rotating passage, lb/sq ft abs
R	gas constant, ft-lb/(lb)(°F)
Re_{av}	average Reynolds number of gas, $\frac{p_{av} W_{av} l_o/\pi}{\mu_b g R T_b}$
r	radius, ft

T	temperature, $^{\circ}R$
T'	total temperature, $^{\circ}R$
T''	total temperature with respect to rotating passage, $^{\circ}R$
U	actual blade speed, ft/sec
W	velocity relative to blade, ft/sec
Y	corrugation amplitude, in.
z	exponent of Reynolds number, eq. (1)
η_i	impeller efficiency
μ	viscosity of gas, slugs/(sec)(ft)
τ	corrugation thickness, in.
1	compressor bleed point, interstage bleed
2	compressor bleed point, between point 1 and compressor discharge
3	compressor bleed point, at compressor discharge

Subscripts:

a	air
av	average
b	blade
e	effective
g	combustion gas
h	hub
in	inlet
o	outside or outlet (of blade, when used with p)
R	turbine rotor
S	turbine stator

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t tip

tot total, when used with C refers to $C_R + C_S$

4 turbine stator inlet

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TABLE I. - COOLING REQUIREMENTS FOR CORRUGATED-INSERT BLADE USED IN SINGLE-STAGE TURBOJET ENGINES

[Flight Mach number, 2.0; altitude, 50,000 ft.]

(a) Turbine rotor blade

Turbine hub-tip radius ratio, r_h/r_t	Corrugation thickness, τ, in.	Turbine tip speed, U_t, ft/sec							
		1100				1700			
		Corrugation spacing, m, in.	Compressor bleed point	Corrugation amplitude, Y, in.	Coolant-flow ratio, C_R	Corrugation spacing, m, in.	Compressor bleed point	Corrugation amplitude, Y, in.	Coolant-flow ratio, C_R
Turbine-inlet temperature, T_4, 2500° R									
0.60	0.005	0.020	1	0.098	0.014	0.020	1	(a)	(a)
		.035	1	.054	.014	.020	2	0.077	0.033
		.050	1	.048	.014	.035	2	.050	.032
		.020	3	.032	.016	.050	2	.061	.039
						.020	3	(a)	(a)
	0.010	0.035	1	0.068	0.014	0.020	2	0.086	0.029
0.75	0.005	0.020	1	0.032	0.011	0.020	1	0.034	0.024
		.050	1	.026	.012	.035	1	.032	.024
						.050	1	.030	.021
						.020	3	(a)	(a)
	0.010					0.020	1	0.045	0.023
Turbine-inlet temperature, T_4, 3000° R									
0.60	0.005	0.020	1	0.160	0.026	0.020	1	(a)	(a)
		.050	1	.106	.045	.020	2	0.130	0.063
		.020	3	.053	.036	.050	2	.090	.100
						.020	3	(a)	(a)
0.75	0.005	0.020	1	0.054	0.027	0.020	1	0.076	0.041
						.050	1	.066	.048
Turbine-inlet temperature, T_4, 3500° R									
0.60	0.005	0.020	1	0.251	0.041	(a)			
		.050	1	.188	.085				
		.020	3	.092	.060				
		.050	3	.075	.078				
	0.010	0.020	3	0.112	0.055				
		.050	3	.090	.069				
0.75	0.005	0.020	1	0.108	0.049				
		.050	1	.090	.074				

[a]No solution.

[redacted]

TABLE I. - Concluded. COOLING REQUIREMENTS FOR CORRUGATED-INSERT BLADE USED IN SINGLE-STAGE TURBOJET ENGINES

[Flight Mach number, 2.0; altitude, 50,000 ft.]

(b) Turbine stator blade. Corrugation thickness, 0.005 inch

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Turbine hub-tip radius ratio, r_h/r_t	Corrugation spacing, m, in.	Compressor bleed point	Turbine tip speed, U_t, ft/sec			
			1100		1700	
			Corrugation amplitude, Y, in.	Coolant-flow ratio, C_S	Corrugation amplitude, Y, in.	Coolant-flow ratio, C_S
Turbine-inlet temperature, T_4', 2500° R						
0.60	0.020	3	0.060	0.023	0.047	0.036
	.035	3	.041	.025	.042	.036
	.050	3	.042	.029	.041	.036
0.75	0.020	3	0.032	0.029	0.039	0.049
Turbine-inlet temperature, T_4', 3000° R						
0.60	0.020	3	0.098	0.046	0.083	0.060
0.75	0.020	3	0.060	0.064	0.071	0.104
	.050	3	.055	.068	.069	.154
	.020	2	.075	.056	.130	.133
Turbine-inlet temperature, T_4', 3500° R						
0.60	0.020	3	0.140	0.077	(a)	
	.050	3	.140	.110		
0.75	0.020	3	0.095	0.103		
	.020	2	.150	.089		

[a]No solution.

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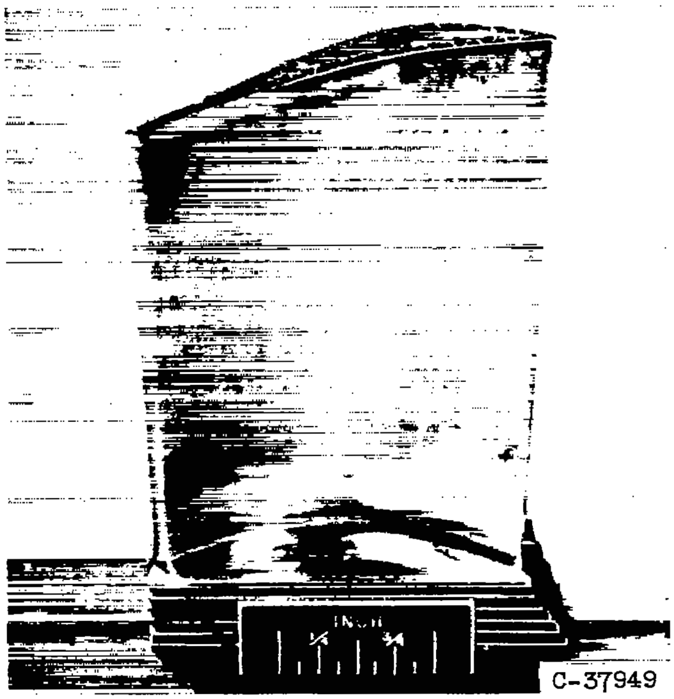
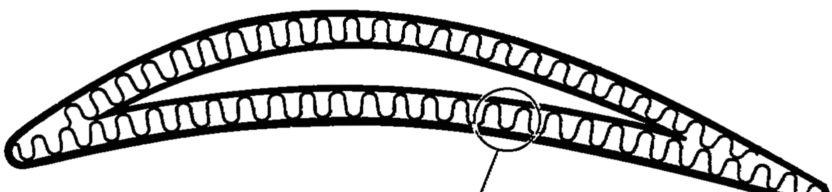
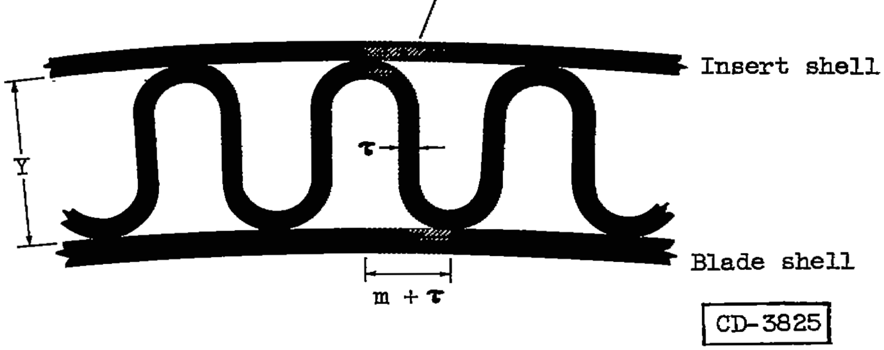


(a) Actual test blade.

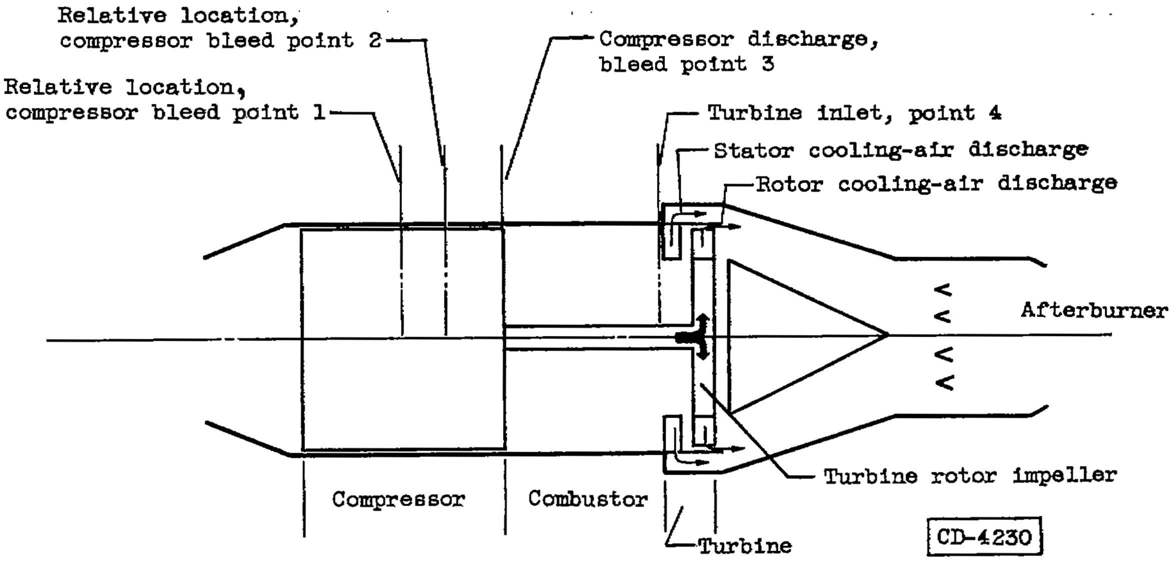


(b) Tip view.

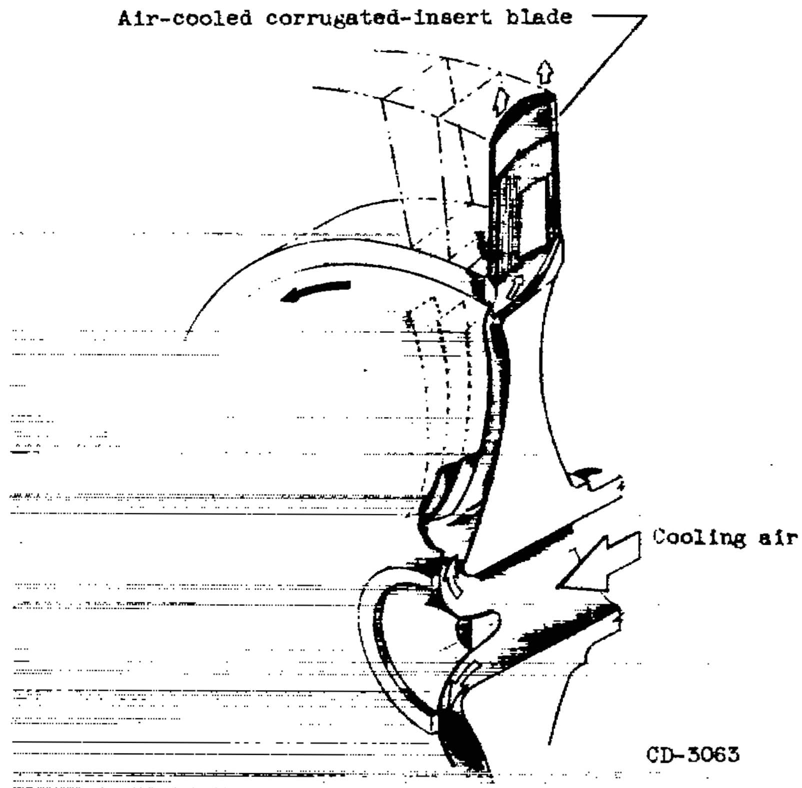


(c) Typical corrugated section.

Figure 1. - Air-cooled corrugated-insert blade.



(a) Air-cooled turbojet engine.



(b) Turbine rotor impeller and air-cooled blade.

Figure 2. - Schematic diagram of single-stage turbojet engine and air-cooled turbine.

[redacted]

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CM-5

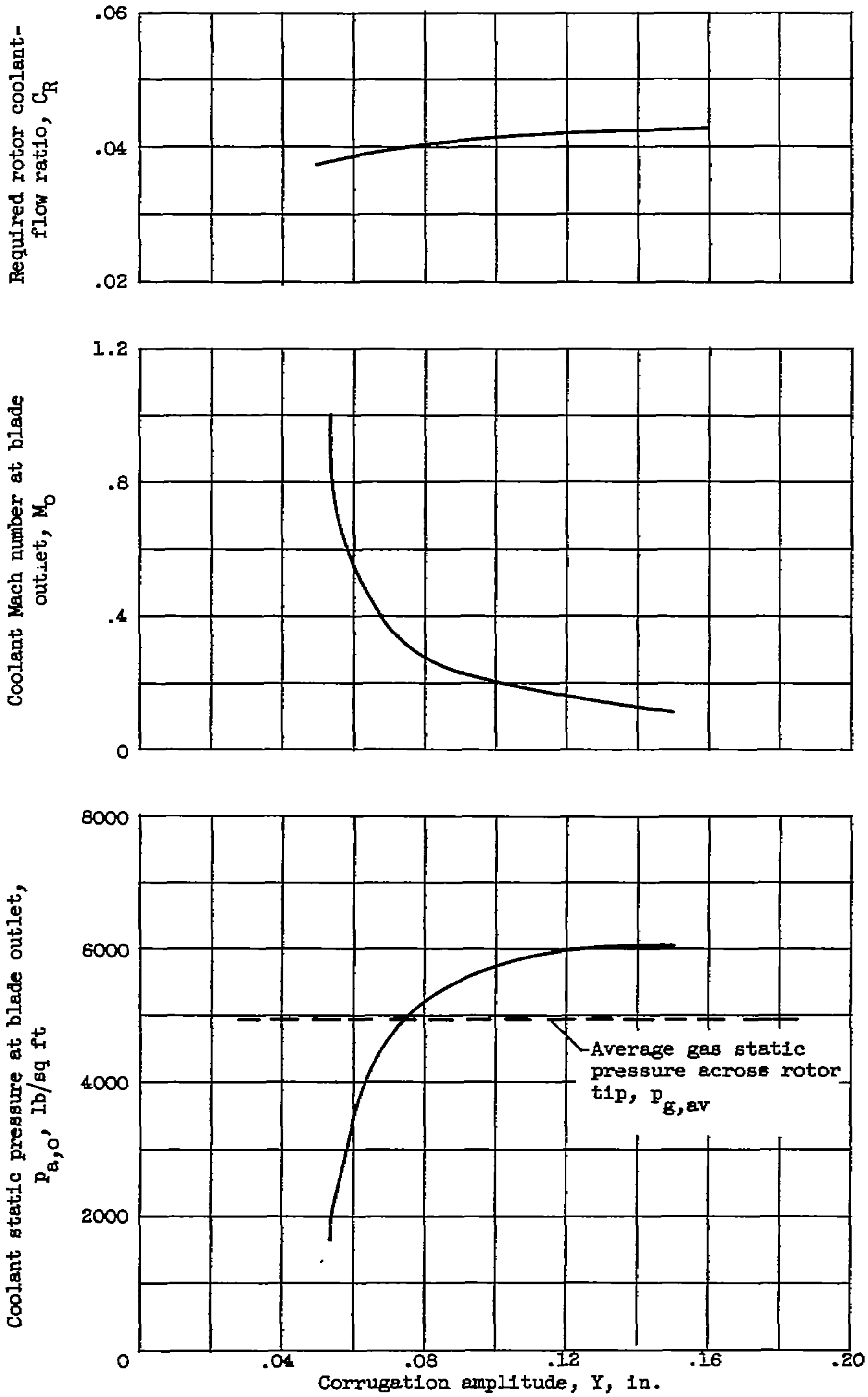

Figure 3. - Variation of required rotor coolant-flow ratio, blade-outlet coolant Mach number, and blade-outlet coolant static pressure with corrugation amplitude.

[redacted]

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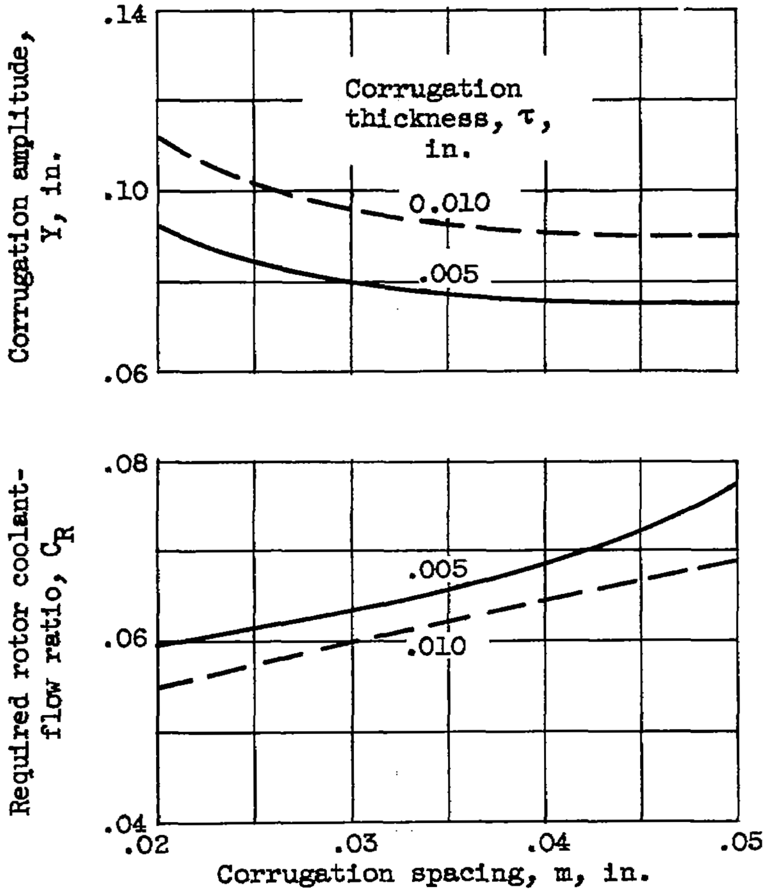

Figure 4. - Variation of corrugation amplitude and required rotor coolant-flow ratio with corrugation spacing for two values of corrugation thickness.

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CM-5 back

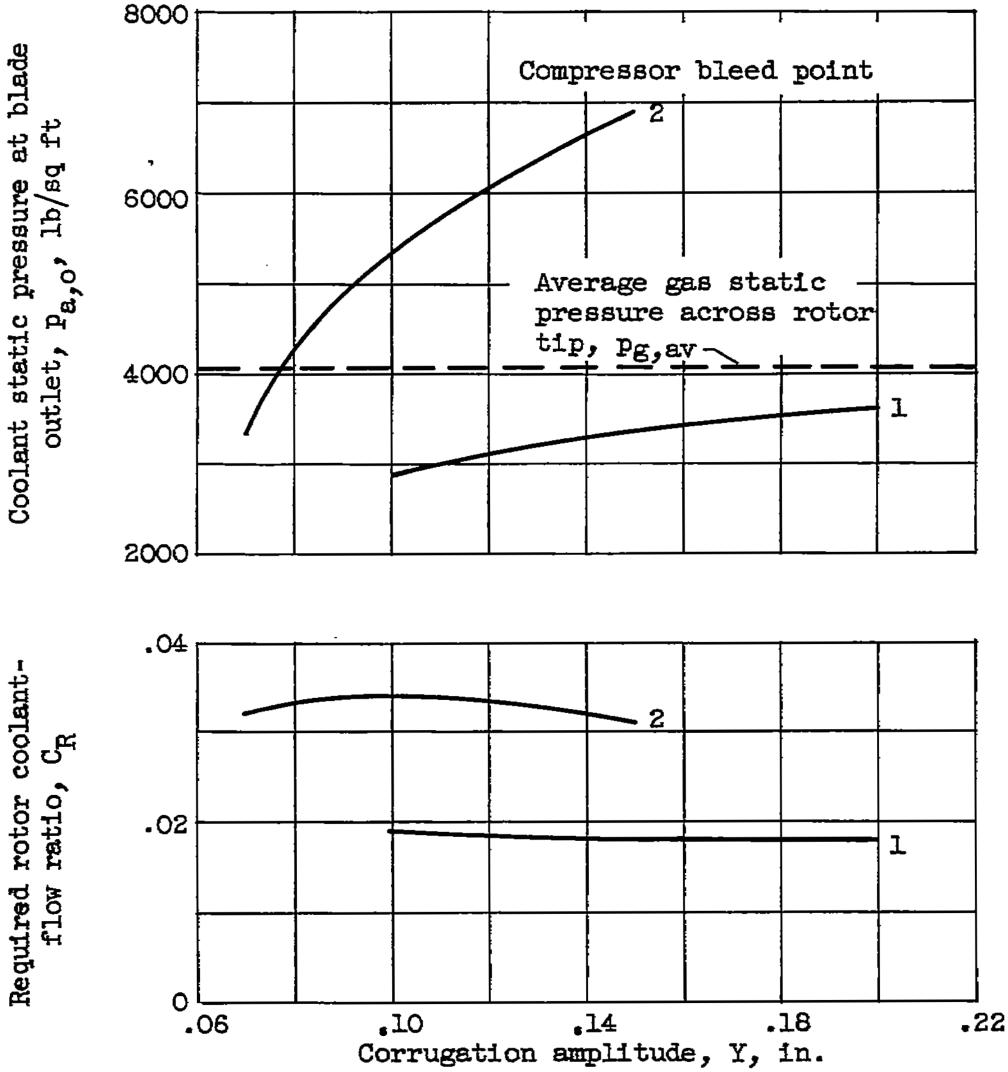

Figure 5. - Variation of coolant static pressure at blade outlet and required rotor coolant-flow ratio with corrugation amplitude for two compressor bleed points. Corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch.

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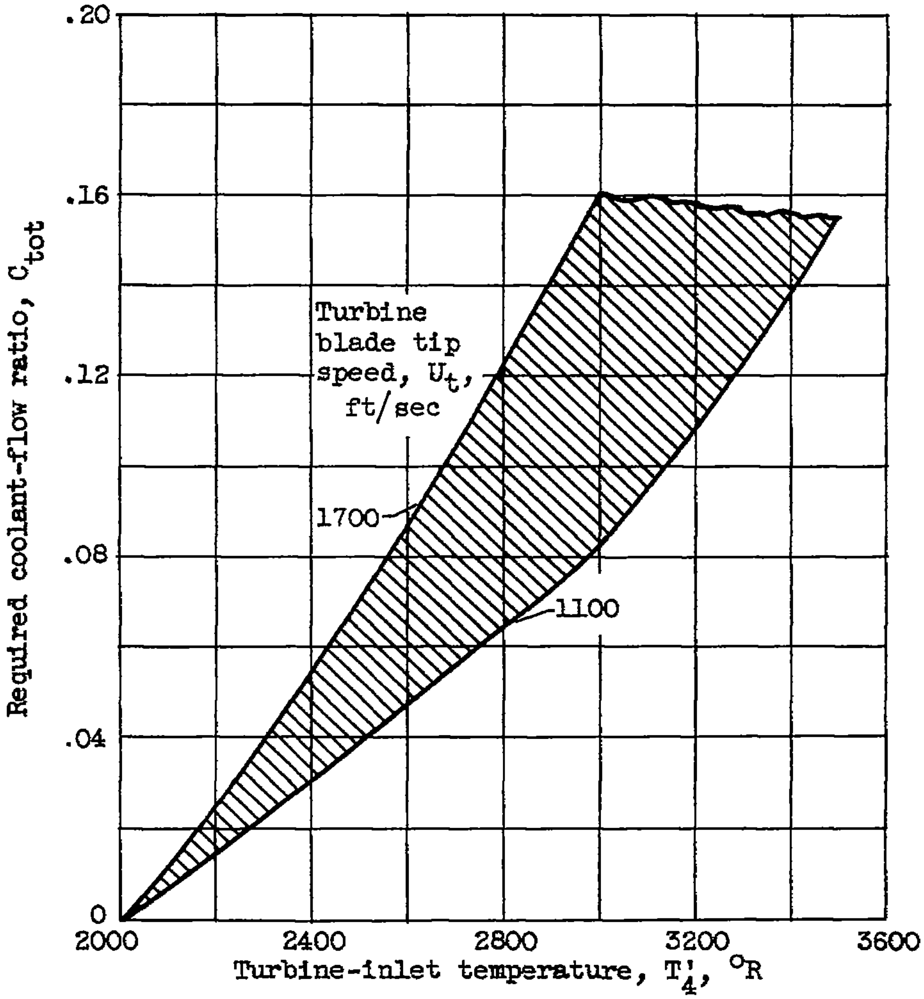
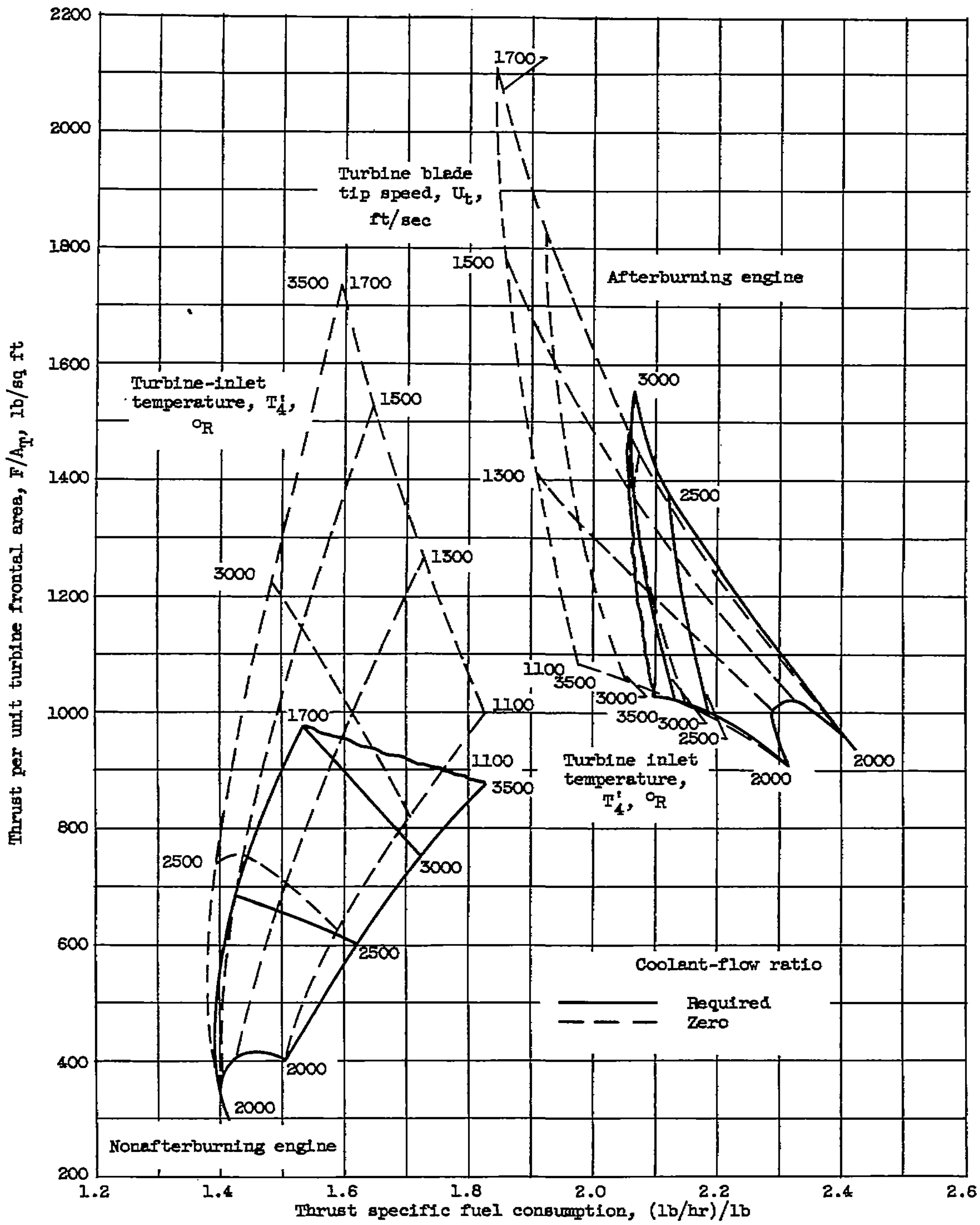
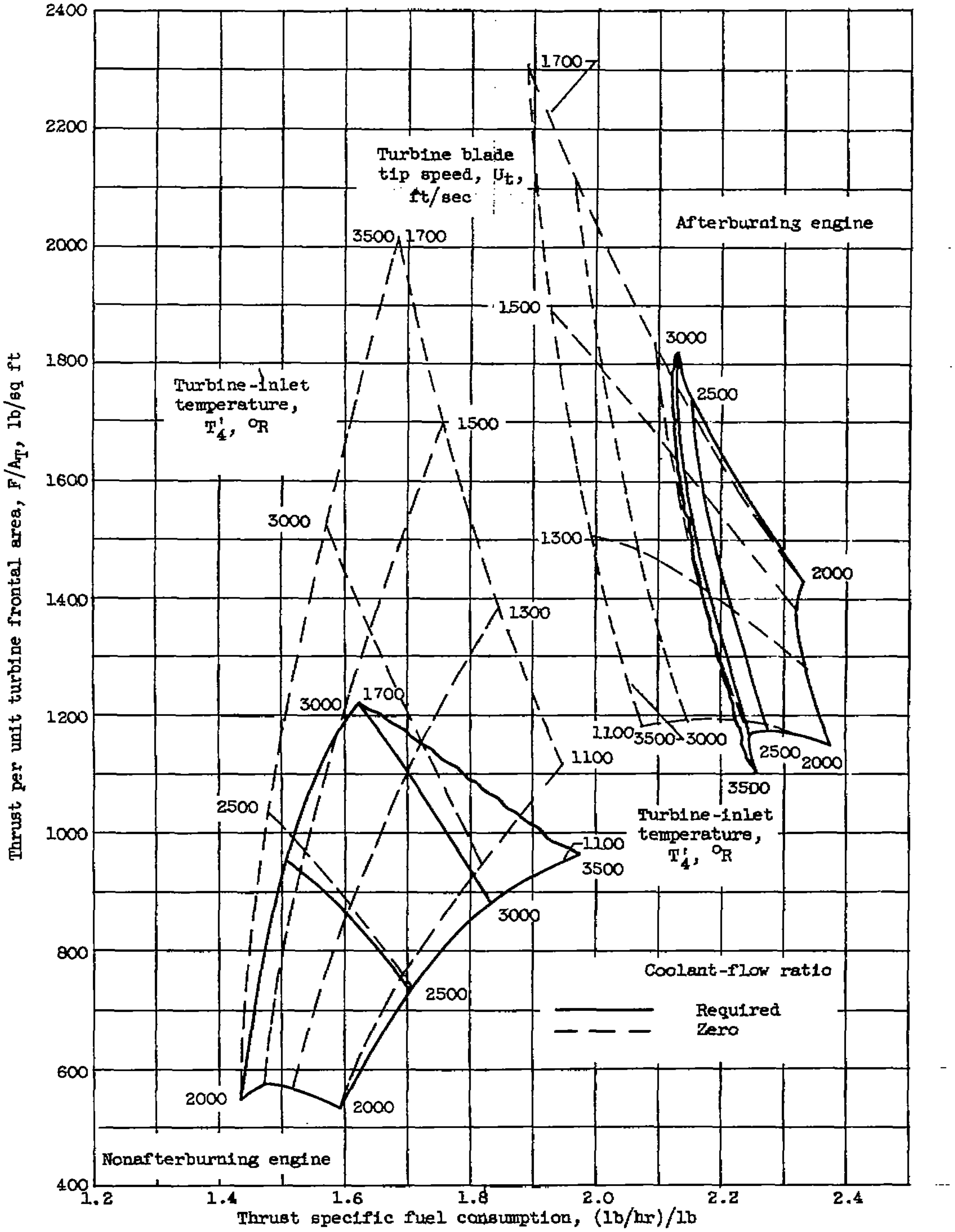


Figure 6. - Approximate minimum coolant-flow range for corrugated-insert blade used for stator and rotor blades of single-stage air-cooled turbine. Flight Mach number, 2.0; altitude, 50,000 feet.



(a) Turbine hub-tip radius ratio, 0.75.

Figure 7. - Afterburning- and nonafterburning-engine performance for range of turbine design variables. Afterburning outlet temperature, 3500° R; flight Mach number, 2.0; altitude, 50,000 feet.



(b) Turbine hub-tip radius ratio, 0.60.

Figure 7. - Concluded. Afterburning- and nonafterburning-engine performance for range of turbine design variables. Afterburning outlet temperature, 3500° R; flight Mach number, 2.0; altitude, 50,000 feet.

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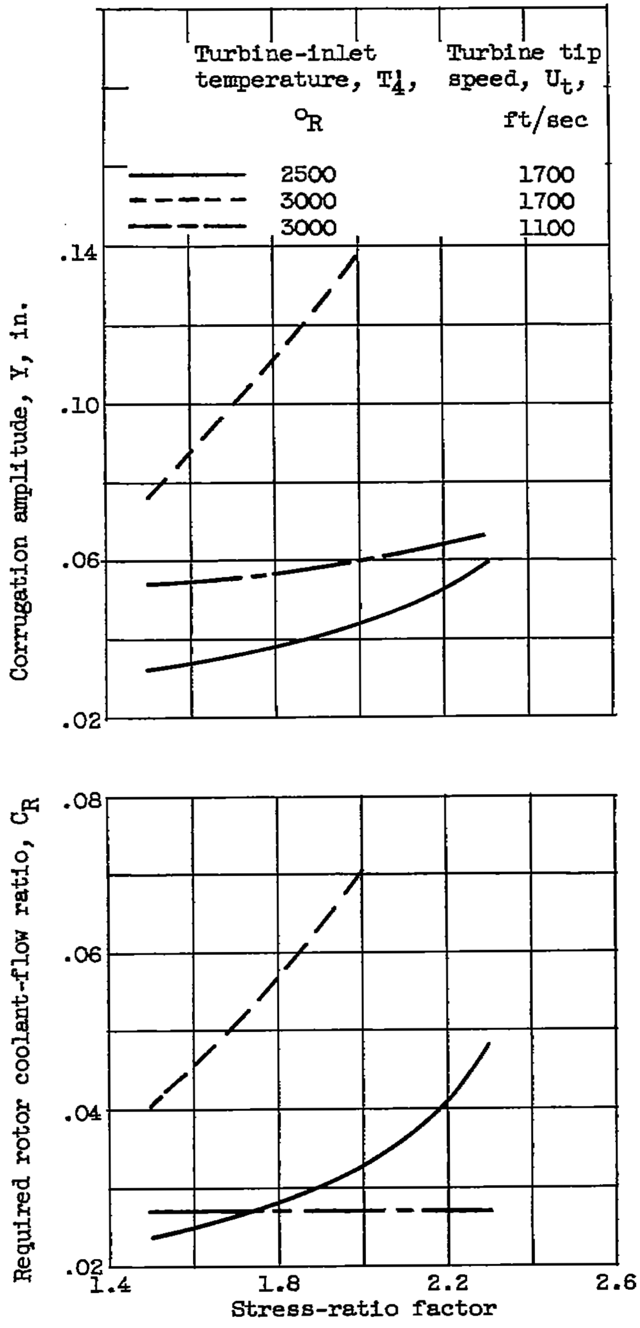

Figure 8. - Variation of corrugation amplitude and coolant-flow ratio with stress-ratio factor. Hub-tip radius ratio, 0.75. Corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch; compressor bleed point, 1; flight Mach number, 2.0; altitude, 50,000 feet.

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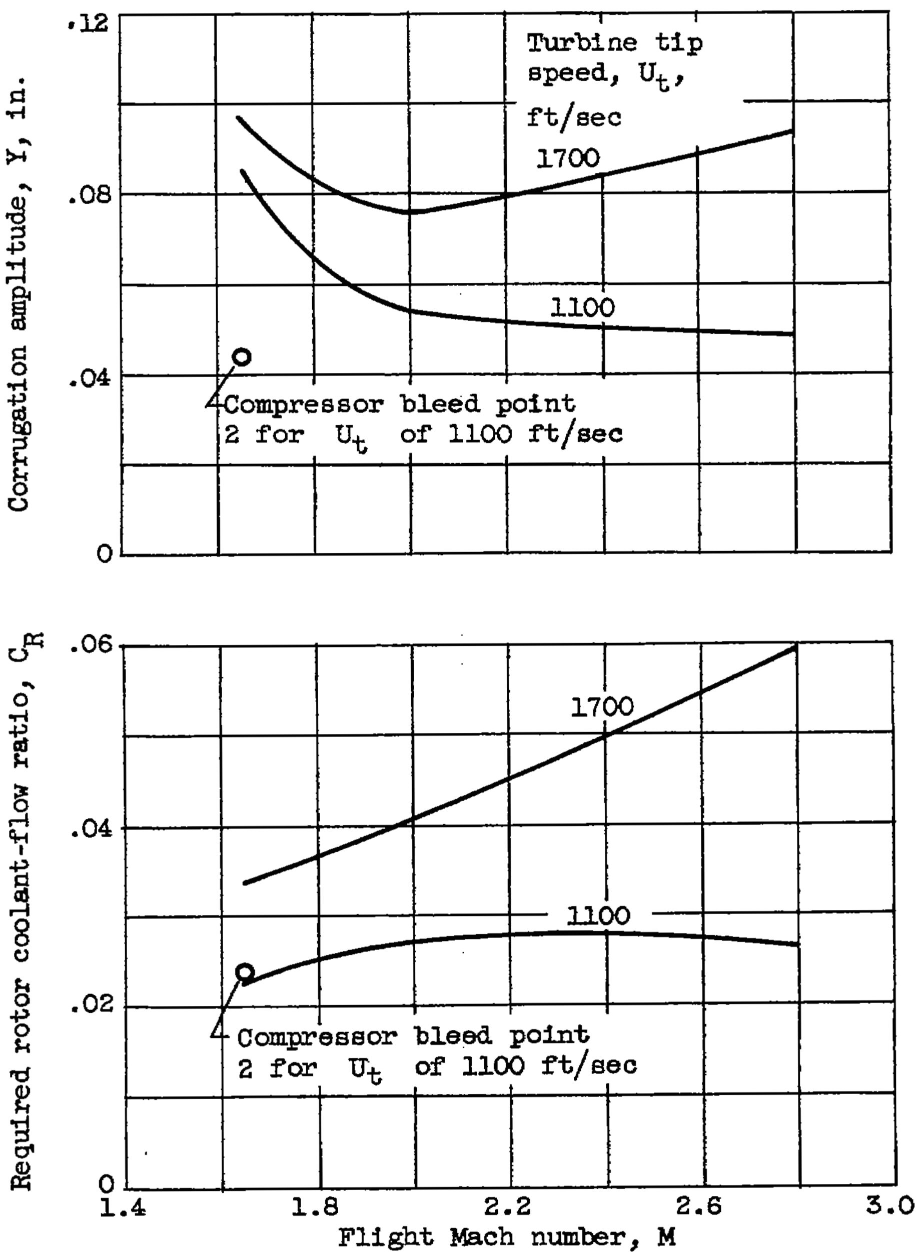

Figure 9. - Effect of flight Mach number on corrugation amplitude and coolant-flow ratio. Turbine-inlet temperature, 3000° R; hub-tip radius ratio, 0.75; corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch; compressor bleed point, 1; altitude, 50,000 feet.

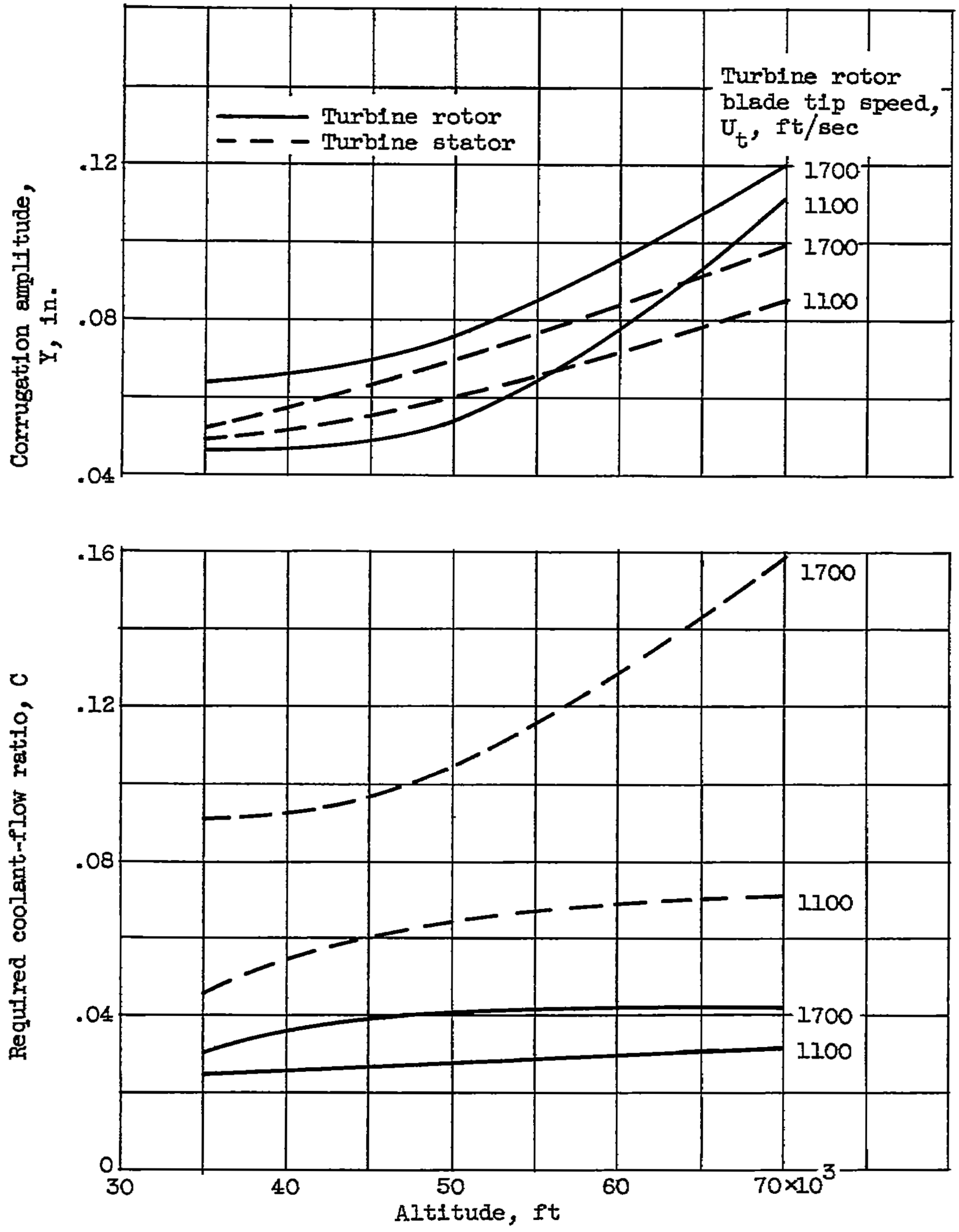

Figure 10. - Effect of altitude on corrugation amplitude and coolant-flow ratio. Flight Mach number, 2.0; turbine-inlet temperature, 3000° R; turbine hub-tip radius ratio, 0.75; corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch.

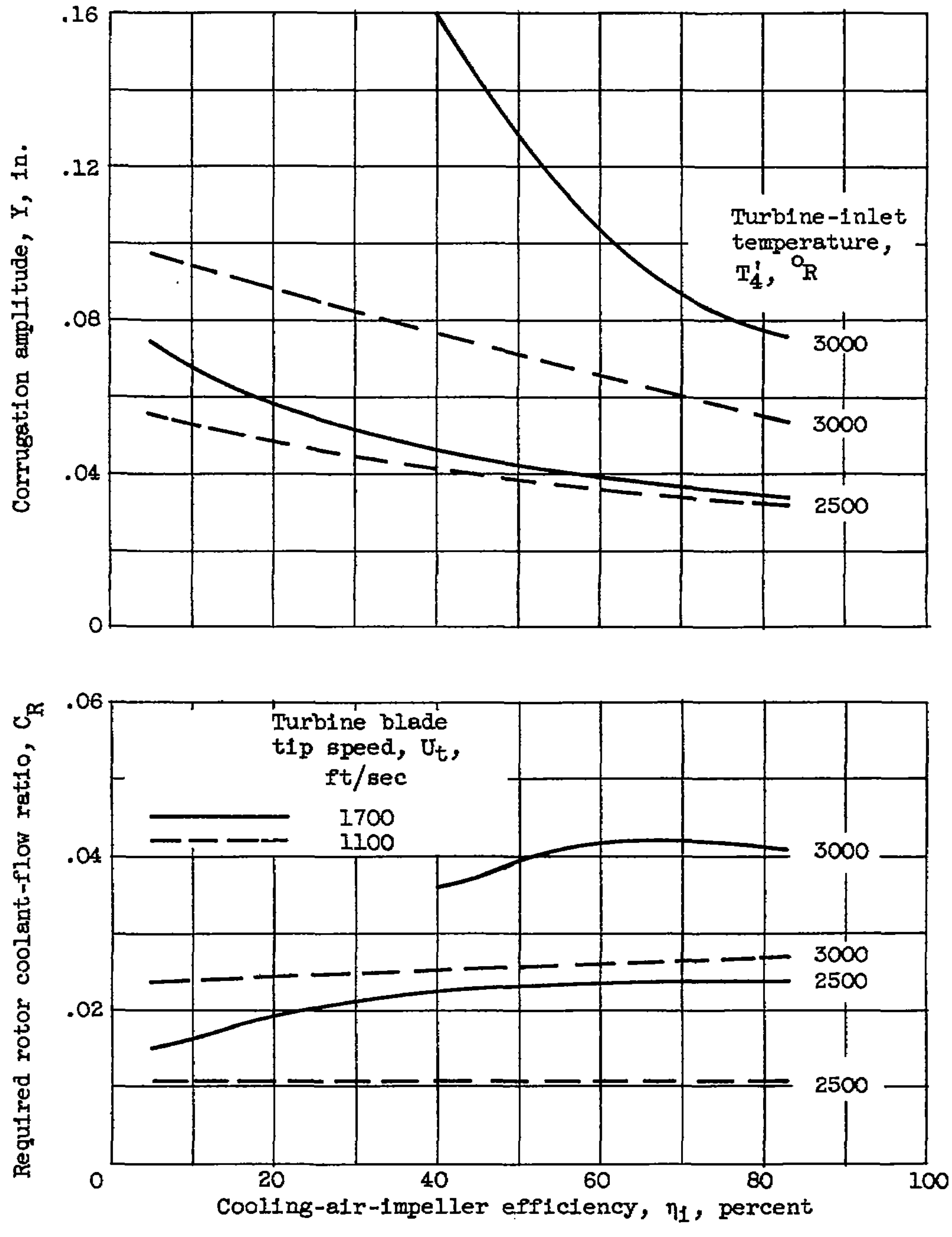

Figure 11. - Effect of impeller efficiency on corrugation amplitude and coolant-flow ratio. Hub-tip radius ratio, 0.75; corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch; flight Mach number, 2.0; altitude, 50,000 feet.

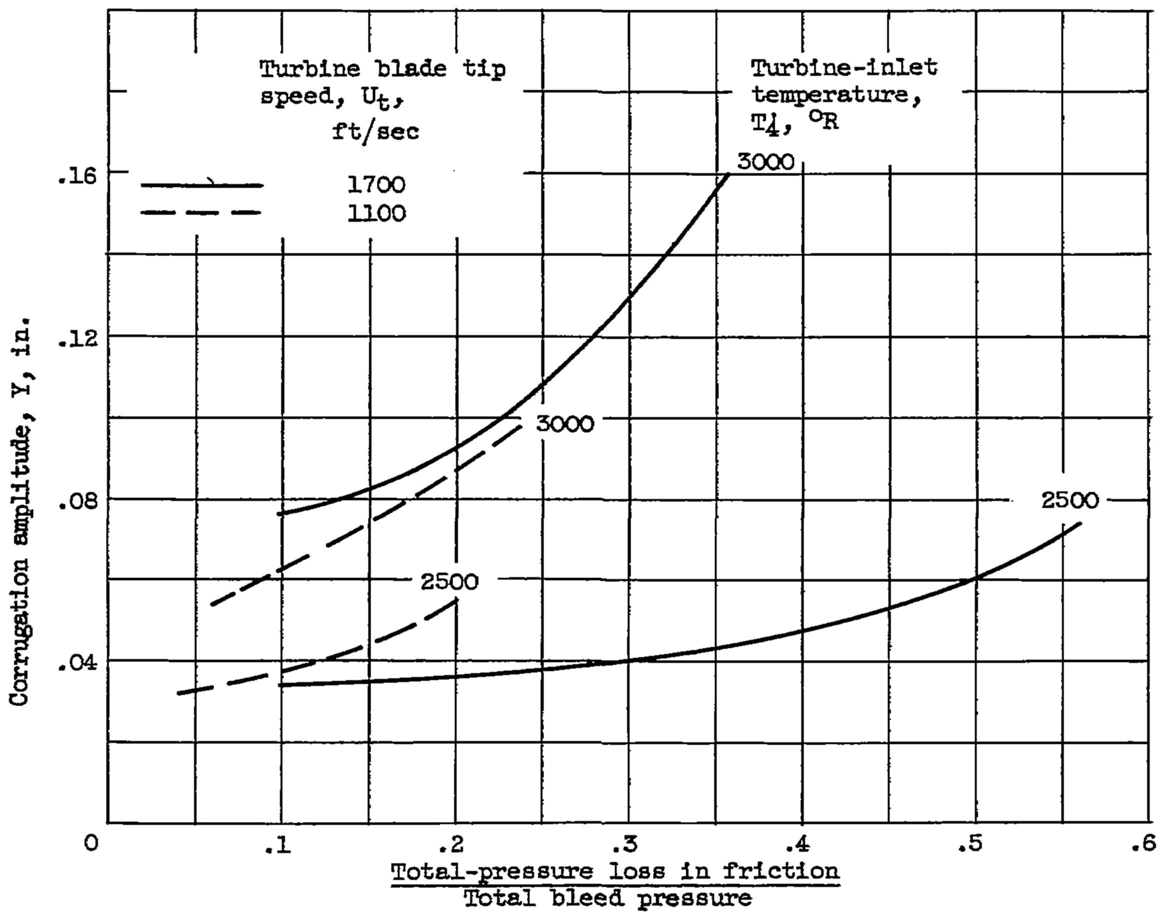

Figure 12. - Effect of cooling-air ducting losses on corrugation amplitude. Hub-tip radius ratio, 0.75; corrugation thickness, 0.005 inch; corrugation spacing, 0.020 inch; flight Mach number, 2.0; altitude, 50,000 feet.

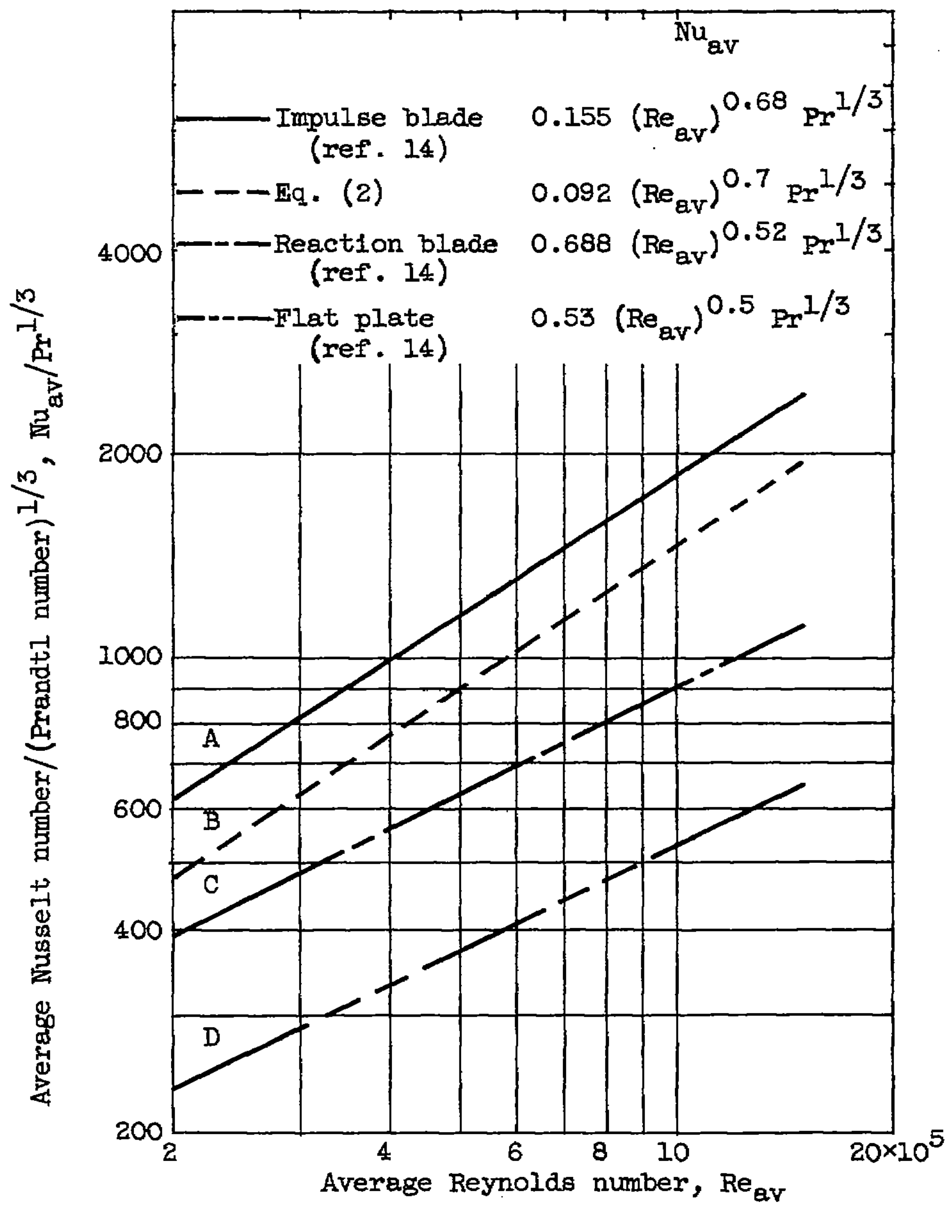

Figure 13. - Comparison of heat-transfer relation used herein with extreme experimental relations.

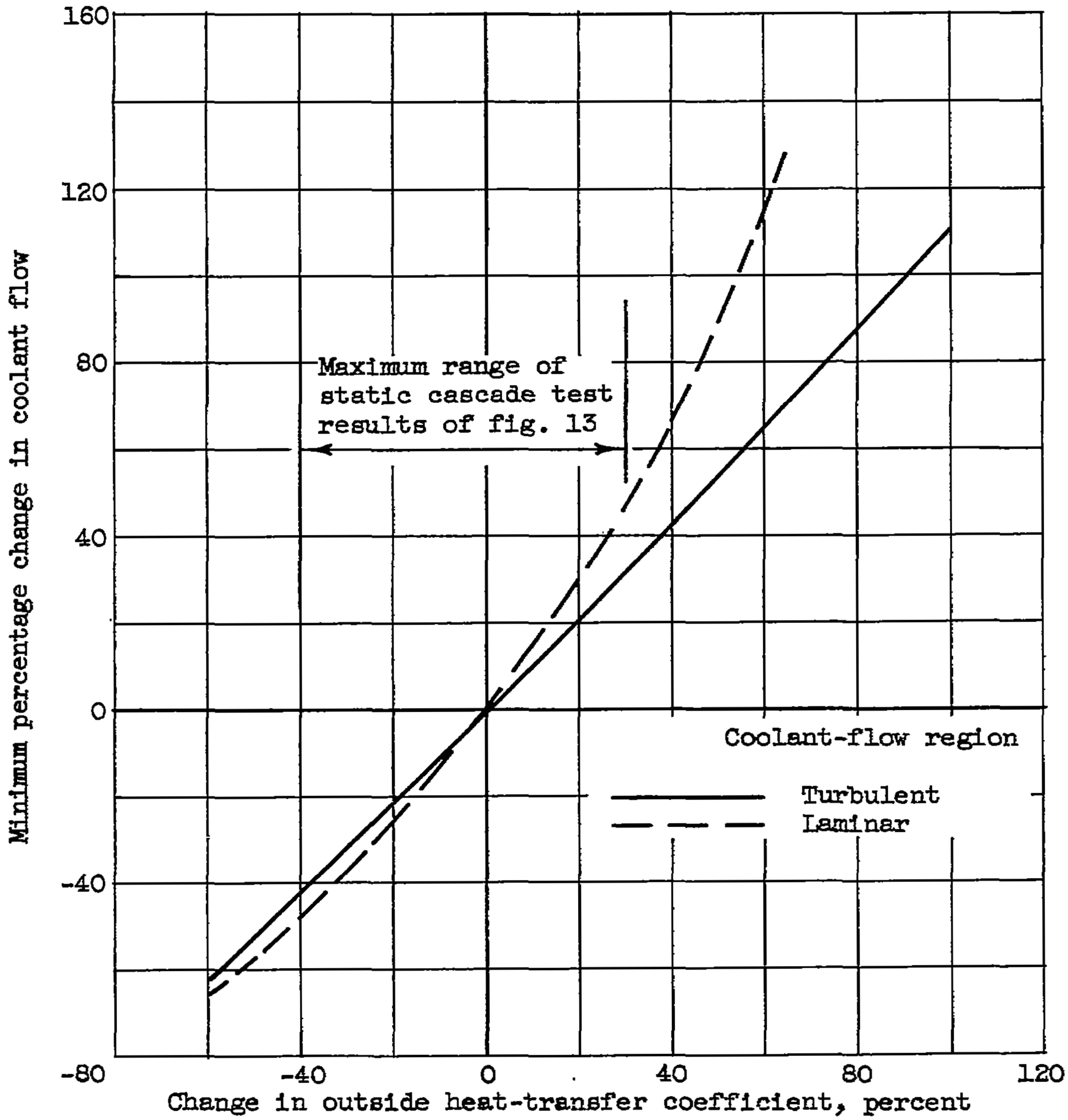

Figure 14. - Effect of changes in outside heat-transfer coefficient on coolant flow.

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