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RESEARCH MEMORANDUM

ANALYSIS OF COOLING-AIR REQUIREMENTS OF CORRUGATED-

INSERT-TYPE TURBINE BLADES SUITABLE FOR A

SUPERSONIC TURBOJET ENGINE

By Robert R. Ziemer and Henry O. Slone

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

An analysis was made to determine the turbine cooling-air require-
ments of a particular turbojet-engine design believed to be representative
of future low-specific-weight engines designed for supersonic flight at
high altitudes. The engine, which was operated over a wide range of
flight Mach numbers and altitudes, had an eight-stage transonic compres-
sor and a high-stress two-stage turbine having a turbine-inlet temperature
of 2500° R. The mode of compressor operation considered was a combination
of constant equivalent and mechanical speeds.

The results of this study show that the required coolant-flow ratio
increases as flight Mach number increases for a given altitude, and for
a given flight Mach number, an increase in altitude causes an increase in
the required coolant-flow ratio. At an altitude of 40,000 feet, a change
in flight Mach number from 0.90 to 2.5 results in an increase in the re-
quired coolant-flow ratio of about 75 percent. A change in altitude from
40,000 to 70,000 feet at a flight Mach number of 2.5 results in an increase
in the required coolant-flow ratio of about 35 percent. In this particular
engine, a maximum total coolant-flow ratio of 0.088 occurred at the maxi-
mum flight Mach number of 2.5 and the maximum altitude of 70,000 feet.
This coolant-flow ratio was obtained when the rotor-blade cooling air was
obtained from the seventh compressor stage while supplying the stator
blades with combustor secondary air. Moving the rotor-blade bleed point
from the seventh to the fifth compressor stage results in about a 16-
percent reduction in the required total coolant-flow ratio.

INTRODUCTION

The use of high-altitude supersonic turbojet-powered airplanes de-
mands engines of low specific weight. Cooling of the turbine blades and
disks is an important factor in realizing low-specific-weight engines.
Some of the effects of cooling on engine performance when using compressor

bleed for the cooling air have been investigated and reported in refer-
ence 1. In order to evaluate the effects of cooling on engine perform-
ance, the required coolant-flow ratios obtained for a given blade-cooling
configuration must be determined. The required coolant-flow ratios are
presented in reference 2 for design-point operation of turbojet engines
equipped with single-stage turbines utilizing corrugated-insert-type
turbine blades. In general, for the design-point engines considered in
reference 2, the cooling problem became more severe as either the flight
Mach number or altitude increased. However, the effects of flight Mach
number and altitude on the cooling requirements of a given turbojet engine
operating over a wide range of flight speeds and altitudes have not been
evaluated. Therefore, an analysis was made to determine the cooling-air
requirements for a particular turbojet engine believed to be representative
of future low-specific-weight engines designed for supersonic flight at
high altitudes. This engine utilized two turbine stages and was operated
over a wide range of flight Mach numbers and altitudes up to a maximum
Mach number of 2.5 and altitude of 70,000 feet.

Low specific weight can be obtained by improved structural design
and by obtaining high thrust per pound of engine air flow and high air
flow per square foot of engine frontal area. Higher thrust per unit of
engine size must be obtained by the use of components of advanced design,
such as high-flow transonic compressors, high-velocity combustors, and a
turbine having good mass-flow handling capacity. The use of a high
turbine-inlet temperature will increase the thrust per pound of engine
air flow. Matching the characteristics of a transonic compressor, designed
near its aerodynamic limits, to a turbine with a frontal area near that
of the compressor results in a two-stage turbine which has quite high
centrifugal stresses at the rotor-blade root.

Turbine cooling provides an important contribution to a compact
engine design, since it permits realization of high turbine-blade stresses
and high turbine-inlet temperature, thus exploiting the advantages of the
transonic compressor and the higher thrust per pound of air. Not only
do high turbine stress and temperature affect cooling-air requirements,
but changes in flight conditions also can alter the cooling requirements
substantially (ref. 2). Therefore, the selection of the engine operating
line for a given application has a marked influence on the cooling-air
requirements.

In this analysis, an effort was made to obtain an engine design of
low specific weight by matching the characteristics of a transonic com-
pressor to an air-cooled two-stage turbine. The engine has a turbine-
inlet temperature of 2500° R and a centrifugal stress at the blade root
of the last turbine stage of 40,000 pounds per square inch. It was as-
sumed that the engine uses a combination of constant mechanical and
equivalent speed in its operation. A corrugated-insert blade was used

in the air-cooled turbines because, as pointed out in reference 2, it provides a good cooling effectiveness and is amenable to analytical treatment for determining coolant flows.

This report presents the cooling-air requirements of a "paper" air-cooled turbojet engine operating over a wide range of flight Mach numbers and altitudes. The results are presented for a two-stage-turbine turbojet engine having a turbine-inlet temperature of 2500° R and operating at sea-level static conditions and flight Mach numbers from 0.90 to 2.5 and flight altitudes from 40,000 to 70,000 feet. A comparison is made of the required coolant-flow ratios of the first- and second-stage turbine-rotor blades when both sets of blades are supplied with cooling air from the same compressor bleed point. In addition, the effect of decreasing the blade-inlet cooling-air temperature on the required coolant-flow ratio is presented.

DESCRIPTION OF ENGINE

General Considerations

In order to determine the cooling-air requirements for the turbine of an air-cooled turbojet engine operating at supersonic flight speeds and high altitudes, it is necessary to have a given engine design and operating conditions. In an effort to obtain an engine of low specific weight for this cooling analysis, a turbine-inlet temperature of 2500° R, an eight-stage transonic compressor, and a centrifugal stress at the blade root of the last turbine stage of 40,000 pounds per square inch were assumed. The turbine configuration selected had two stages of free vortex design and a 15-percent-larger frontal area than the compressor. The turbine had a tip speed of about 1200 feet per second, and the hub-tip radius ratio of the first stage was 0.683 and that of the second stage was 0.565. The blade lengths of the first- and second-stage rotor blades were 4.62 and 6.33 inches, respectively.

The transonic compressor has an assumed flow capacity of 37.5 pounds per second square foot, an inlet hub-tip radius ratio of 0.4, and a sea-level compressor pressure ratio of approximately 8 to 1. A high centrifugal stress at the blade root of the last turbine stage was assumed so that the engine could operate at higher rotative speeds than would be possible with some lower value of stress. This increase in rotative speed thus enables the compressor to produce the specified pressure ratio in fewer stages. A partial list of some of the engine operating conditions is included in table I.

Cooling System

The turbine-cooling system assumed in this analysis is shown in figure 1. In order to have sufficient pressure to pass the cooling air through the turbine rotor blades, it was assumed that the rotor cooling air for both turbine stages was bled from the seventh stage of the eight-stage transonic compressor. The cooling air was assumed to be divided between the two rotor stages according to their cooling requirements and then discharged at the rotor-blade tips into the combustion-gas stream.

The first-stage stator cooling air was assumed to be bled from the primary-combustor secondary air at a position just ahead of the combustor transition liner and ducted to the inner shroud of the stator ring. The cooling air from the first-stage stator was assumed to be discharged into the area between the engine shell and the engine nacelle. The primary reason for selecting the combustor secondary air for the first-stage stators was that it was believed that the air could be delivered to the blades with a minimum amount of ducting. There is, however, some penalty in using this air for cooling. Because of burning within the combustor liner, the secondary air is heated as it flows toward the turbine. Unpublished results obtained from a primary combustor with a turbine-inlet temperature of 2500° R indicated that for the particular configuration tested, the secondary air experienced a temperature rise of about 100° F from the combustor inlet to the transition-liner inlet. Therefore, the cooling-air supply temperature was assumed to be 100° F higher than the compressor-exit temperature for calculating the first-stage-stator cooling-air flow ratio.

If air is required for cooling the second-stage stator blades, it is assumed that combustor secondary air will be used and the air will be discharged from the blades into the gas stream at the turbine inner radius.

ANALYTICAL PROCEDURES

The assumptions and constants used in the present cooling analysis are presented in appendix A, and the symbols are defined in appendix B. The procedures used are described in detail in references 3 and 4, and only a brief resume will be given herein. These references present a quick method of evaluating, within specified pressure limits, the required cooling-air flow for corrugated-insert-type air-cooled blades. In order to use the procedures described in references 3 and 4, a heat-transfer coefficient on the outside of the turbine blades must be determined.

Evaluation of Gas-to-Blade Heat-Transfer Coefficient

The gas-to-blade heat-transfer coefficients were determined by the methods outlined in reference 2. The heat-transfer correlation equation used herein (see ref. 2) is

$$Nu = 0.092 \ Re^{0.70} \ Pr^{1/3} \tag{1}$$

where

$$Nu = \frac{h \ \dfrac{l_o}{\pi}}{k_b} \tag{2}$$

and

$$Re = \frac{pV \ \dfrac{l_o}{\pi}}{\mu_b g R T_b} \tag{3}$$

The coefficient 0.092 and the exponent 0.70 were assumed in equation (1) and are thought to be representative values for the blade profiles obtained for this turbine. The values of the product pV at the midspan position at the turbine stage exit are used in equation (3).

Equation (1) was used to calculate the average gas-to-blade heat-transfer coefficient h for both turbine stator and rotor blades. The values of h computed from equation (1) may be somewhat high for the reaction blading of the stator blades, since the favorable pressure gradient may permit laminar flow over a large portion of the blade surface. Consequently, the required coolant-flow ratios for the first-stage stator blades may be slightly conservative.

Selection of Corrugation Geometry

When analyzing the coolant requirements of a corrugated-insert-type turbine blade for a given engine application, it is important to select a corrugation geometry which gives the lowest possible coolant flow within limitations of the pressure levels through the engine and of the blade pressure drop. The selection of a suitable corrugation geometry also depends on the blade profile being considered and the ease of fabricating the corrugation configuration. In general, small values of corrugation height or amplitude are required for air-cooled turbine blades in order to provide trailing-edge cooling with minimum trailing-edge thickness.

The influence of corrugation amplitude, thickness, and pitch on the coolant-flow requirements of a representative turbine rotor blade is discussed in reference 2. From the preceding discussion and the results of reference 2, a corrugation geometry having small values of corrugation amplitude, thickness, and pitch seems to be most suitable for the engine applications considered herein.

The corrugation geometry configuration selected for both the turbine stator and rotor blades of the present analysis has an amplitude and a pitch of 0.050 inch each, and a thickness of 0.005 inch. A sketch of this configuration is shown in figure 2(a). As a matter of interest, other corrugation configurations having larger values of amplitude (0.070 and 0.10 in.), pitch (0.080 in.) and a thickness of 0.005 inch were analyzed at a flight Mach number of 2.5 and an altitude of 70,000 feet. As was expected, all of these configurations resulted in higher coolant-flow ratios than the configuration selected.

The small amplitude of 0.050 inch enables the placement of the selected corrugations almost to the trailing edges of the blades. Because these corrugations are very small, there was some question as to whether or not the passages would fill up with braze material when brazing the corrugations to the blade shell. In order to determine the practicality of the corrugation design, a small section of corrugations of this size was fabricated and brazed to a simulated blade shell and insert (fig. 2(b)). As can be seen in figure 2(b), there seems to be no problem of braze material clogging the air passages. Also, the corrugations are quite uniform and appear to be fastened securely to both members.

RESULTS AND DISCUSSION

Following the procedures outlined in reference 3, the required coolant-flow ratios were obtained for the turbine rotor blades and first-stage stator blades over a range of flight Mach numbers and altitudes. The methods outlined in reference 4 were used to make sure that there was sufficient pressure available to pass the required cooling-air flow through the blades for all conditions. Since the limiting allowable blade temperature assumed for the second-stage stator blades was approximately equal to the gas temperature, it was assumed that these blades required a coolant-flow ratio of 0.005 for all flight conditions. A summary of some of the engine operating conditions and a breakdown of the individual blade coolant-flow ratios are given in table I.

Effect of Flight Mach Number and Altitude on Turbine Required Coolant-

Flow Ratio Using Seventh-Stage Compressor Bleed for Cooling

The required coolant-flow ratios for the specific engine of this analysis operating over a range of flight Mach numbers M and altitudes are shown in figure 3. The ordinate of figure 3 is the total turbine coolant-flow ratio C_{tot}, which is the sum of the required coolant-flow ratios for the turbine rotor and stator blades (see table I). In results presented in figure 3, the rotor cooling air was assumed to be supplied by the compressor seventh stage, and the combustor secondary air was used for supplying the stator blades. Also, 15 percent of the total pressure available at the compressor bleed point was assumed to be lost by the time the air reached the base of the turbine blades. This loss was assumed to include any loss entailed at the bleed point and any subsequent ducting losses. Then, the supply pressure at the base of the rotor blades must be sufficient to pass the required cooling-air flow and discharge it into the combustion-gas stream at the blade tip.

For any particular altitude, the coolant-flow ratio increased with flight Mach number primarily because of the rise in the blade-inlet cooling air temperature (fig. 3). At an altitude of 40,000 feet, a change in M from 0.90 to 2.5 results in an increase in C_{tot} of about 75 percent, and a corresponding increase in the blade-inlet cooling-air temperature for the stator and rotor blades (see table I) of approximately 50 and 60 percent, respectively.

At any given flight Mach number, the required value of C_{tot} also increases as the altitude is increased (fig. 3). A change in altitude from 40,000 to 70,000 feet at a flight Mach number of 2.5 results in an increase in C_{tot} of about 35 percent. This trend is similar to that observed in reference 2 where minimum coolant-flow ratios were obtained over a range of altitudes for a constant flight Mach number by changes in corrugation geometry.

The maximum coolant-flow ratio obtained for the engine considered herein is 0.088 and occurs for a flight Mach number of 2.5 and an altitude of 70,000 feet.

Effect of Blade-Inlet Cooling-Air Temperature and Compressor Bleed

Point on Required Coolant-Flow Ratio

Performance variations resulting from bleeding cooling air from the compressor of turbojet engines for a wide range of engine operating conditions were analytically investigated in reference 1. This reference

shows that the engine specific thrust decreases approximately 1 percent for every percent of turbine rotor cooling air for a nonafterburning turbojet engine with a turbine-inlet temperature of 2500° R and a flight Mach number of 2.0 in the stratosphere. The specific fuel consumption increases about 1 percent for every 6 percent of turbine rotor cooling air. Thus, anything that can be done to reduce the required coolant-flow ratio of the turbojet engine investigated herein will result in improved performance of the engine. In addition, reductions in the required coolant-flow ratio will tend to reduce the size of the cooling-air ducting system, thereby decreasing engine weight.

As stated in the preceding section, a primary reason for increasing values of C_{tot} with rising flight Mach number is the increasing values of the blade-inlet cooling-air temperature. Therefore, if some means could be used to reduce this temperature, the required coolant-flow ratios will be reduced. One such method of reducing the blade-inlet cooling-air temperature would be to bleed the compressor ahead of the seventh stage, provided there is sufficient pressure available to pass the cooling air. Considering the flight condition wherein the maximum coolant-flow ratio was obtained (M, 2.5; altitude, 70,000 ft), calculations indicated that there would be sufficient pressure to pass the cooling air, if the cooling air for the rotor blades were bled at the fifth compressor stage. For this calculation, as before, 15 percent of the total pressure available at any compressor bleed point was assumed to be lost by the time the air reached the base of the turbine blades.

The effects of reducing the blade-inlet cooling-air temperature on the required coolant-flow ratios of the first- and second-stage rotor blades for a flight Mach number of 2.5 and an altitude of 70,000 feet will be discussed subsequently. Then, for these same flight conditions, the effect of bleeding the compressor at the fifth and sixth stages on the engine-required coolant-flow ratio will be presented.

Effects of reductions in blade-inlet cooling-air temperature. - The required coolant-flow ratios for the first- and second-stage rotor blades, $C_{r,1}$ and $C_{r,2}$, respectively, are plotted in figure 4 against the blade-inlet cooling-air temperature for a flight Mach number of 2.5 and an altitude of 70,000 feet. For these flight conditions, $C_{r,1}$ is 0.031 for a blade-inlet cooling-air temperature of 1422° R which is obtained for the seventh-stage compressor bleed. By bleeding at the fifth compressor stage, the blade-inlet cooling-air temperature is reduced to 1325° R and $C_{r,1}$ is reduced to about 0.022.

Arbitrarily decreasing the blade-inlet cooling-air temperature about 200° from the value obtained at the seventh compressor stage (1422° R) results in a reduction in $C_{r,1}$ to a value of 0.0165, or about a 47-percent decrease in $C_{r,1}$ over that required when using seventh compressor

stage bleed (fig. 4). In order to obtain a decrease in the blade-inlet cooling-air temperature below the 1325° R value obtained for the fifth-stage compressor bleed, some auxiliary means must be used because there is insufficient cooling-air pressure available from compressor stages ahead of the fifth stage. One method of reducing the cooling-air temperature might be by the use of heat exchangers.

The effects of reductions in the blade-inlet cooling-air temperature for the second-stage rotor blade on the required coolant-flow ratio, $C_{r,2}$, are clearly shown in figure 4. The second-stage rotor blade requires about 40-percent less coolant flow than the first-stage rotor blade when both blades are supplied with cooling air from the same compressor bleed point. In this case, both blades have the same inlet cooling-air temperature. It should be pointed out that the difference obtained in figure 4 between $C_{r,1}$ and $C_{r,2}$ may change, if the first- and second-stage rotor blades could be supplied from different compressor bleed points.

Effects of compressor bleed point on required engine coolant-flow ratio. - If it is possible to supply the stator blades with cooling air from the compressor discharge instead of from the combustor, a decrease in the stator-blade-inlet cooling-air temperature of about 100° F would result. Such a reduction in cooling-air temperature would probably be accomplished, however, at some expense to the cooling-air duct weight. For comparison purposes, figure 5 shows the reductions in C_{tot} that may be obtained by bleeding the compressor ahead of the seventh stage for the rotor blades and supplying the stator blades with combustor secondary air or compressor discharge air. It is noted that moving the rotor bleed point from the seventh to fifth stage while supplying the stator blades with combustor secondary air results in about a 16-percent decrease in C_{tot}. For a given rotor bleed point, using compressor discharge air to supply the stator blades means a reduction in C_{tot} of approximately 7.7 to 9.2 percent.

Thus, the main point suggested by the results shown in figure 5 is that substantial savings in the required coolant-flow ratio may be obtained by proper choice of the cooling-air supply location. And, as pointed out earlier, for each percent of cooling air saved, the engine specific thrust will increase about 1 percent. For each 6 percent saved in coolant-flow ratio, about a 1-percent saving in specific fuel consumption results. The importance of trying to reduce the required coolant-flow ratio becomes apparent when considering the possible decrease in the cooling-air ducting weight due to reductions in the required engine coolant-flow ratio along with the improved engine performance. Although figure 5 is presented for a given flight condition, similar results could be expected at other flight conditions.

REMARKS

This analytical investigation has been directed toward exploring the required coolant-flow ratios of a corrugated-insert-type blade design suitable for a supersonic turbojet engine with a turbine-inlet temperature of 2500° R. Other flight plans that consider higher flight Mach numbers and higher altitudes than those used herein, or the use of turbojet engines having higher compressor ratios than that considered in this analysis, would all tend to increase the required coolant-flow ratios, and at the more severe conditions, they would probably become excessive. Since maximum flight Mach number and altitude and mode of compressor operation all influence the cooling problem, each engine should be checked over a wide range of operating conditions to obtain the most severe conditions of operation with reference to coolant-flow ratio, compressor bleed point, and cooling-air duct sizes.

SUMMARY OF RESULTS

The results of this analytical investigation to determine the cooling-air requirements of a given turbojet-engine design operating over a wide range of flight Mach numbers and altitudes can be summarized as follows:

1. For a given altitude the required coolant-flow ratio increases as flight Mach number is increased primarily because of the rise in the blade-inlet cooling-air temperature. At an altitude of 40,000 feet, a change in flight Mach number from 0.90 to 2.5 results in an increase in the required coolant-flow ratio of about 75 percent.

2. At any given flight Mach number, the required coolant-flow ratio increases as the altitude is increased. A change in altitude from 40,000 to 70,000 feet at a flight Mach number of 2.5 results in an increase in the required coolant-flow ratio of about 35 percent.

3. The maximum coolant-flow ratio obtained for the turbojet engine considered in this analysis is 0.088 and occurs for a flight Mach number of 2.5 and an altitude of 70,000 feet.

4. A reduction in the blade-inlet cooling-air temperature of 200° results in about a 47-percent reduction in the coolant flow required for the first-stage rotor blades over that required for a blade-inlet cooling-air temperature of 1422° R.

5. The second-stage rotor blade requires about 40-percent less coolant flow than the first-stage rotor blade, when both blades are supplied with cooling air from the same compressor bleed point and the engine is operating at a flight Mach number of 2.5 and an altitude of 70,000 feet.

6. Moving the rotor-blade bleed point from the seventh to fifth compressor stage while supplying the stator blades with combustor secondary air results in about a 16-percent decrease in the required coolant-flow ratio.

7. For a given rotor-blade bleed point, using compressor discharge air to supply the stator blades instead of combustor secondary air means a reduction in the required coolant-flow ratio of approximately 7.7 to 9.2 percent for a flight Mach number of 2.5 and an altitude of 70,000 feet.

Lewis Flight Propulsion Laboratory
    National Advisory Committee for Aeronautics
        Cleveland, Ohio, February 21, 1956

APPENDIX A

ASSUMPTIONS AND CONSTANTS

The following assumptions and constants were used in this analysis:

(1) The corrugation geometry (fig. 2(a)) is constant chordwise and spanwise.

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

(3) A stress-ratio factor (see ref. 5) of 2.0 is used for the turbine rotor corrugated-insert blade.

(4) A high-temperature alloy A-286 was used for the blade corrugation and shell material.

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

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

(7) The ratio of the turbine-blade outside perimeter to the blade chord was 2.30 for both rotor and stator blades.

(8) The cooling air for the turbine rotor blades has a 15-percent loss in pressure between the compressor bleed point and the blade inlet.

(9) The blade chords of the first- and second-stage rotor blades are 2.0 and 2.5 inches, respectively, and the blade chord of the first- and second-stage stator blades is 2.5 inches.

(10) The blade solidity (blade chord/pitch) is 1.8 for the turbine rotor blades and 1.6 for the turbine stator blades.

(11) The blade lengths of the first- and second-stage rotor blades were 4.62 and 6.33 inches, respectively.

(12) The second-stage stator blades require a coolant-flow ratio of 0.005 for all flight conditions.

APPENDIX B

SYMBOLS

C   coolant-flow ratio (ratio of turbine cooling air to compressor weight flow)

c_p specific heat at constant pressure, Btu/(lb)($^\circ$F)

g   standard acceleration due to gravity, 32.174 ft/sec^2

h   average gas-to-blade heat-transfer coefficient, Btu/(sec)(sq ft)($^\circ$F)

k   thermal conductivity, Btu/(sec)(ft)($^\circ$F)

l_o outside perimeter, ft

M   flight Mach number

Nu   Nusselt number of gas, $\dfrac{h \, l_o/\pi}{k_b}$

Pr   Prandtl number of gas, $c_{p,b}\mu_b g/k_b$

p   static pressure, lb/sq ft abs

R   gas constant, ft-lb/(lb)($^\circ$F)

Re   Reynolds number of gas, $\dfrac{pV \, l_o/\pi}{\mu_b g R T_b}$

T   temperature, $^\circ$R

V   velocity relative to blade, ft/sec

μ viscosity of gas, slugs/(sec)(ft)

Subscripts:

b   blade

r   rotor

tot   total

1   first stage

2   second stage

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     1954.

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     and Pressure-Loss Characteristics. NACA RM E52G18, 1952.

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TABLE I. - COOLING REQUIREMENTS FOR CORRUGATED-INSERT-TYPE

BLADE USED IN TWO-STAGE AIR-COOLED TURBOJET ENGINE

[Turbine-inlet temperature, 2500° R; coolant-flow
ratio for second-stage stator blades, 0.005.]

Flight Mach number, M	Altitude, ft	Compressor inlet weight flow, lb/sec	Compressor pressure ratio	Blade-inlet cooling-air temperature, °R		Coolant-flow ratio		
				First-stage stator blade[a]	First-stage rotor blade[b]	First-stage stator blade	First-stage rotor blade	Second-stage rotor blade
2.5	70,000	49.8	4.59	1510	1422	0.034	0.031	0.018
2.5	65,000	63.3	4.59	1510	1422	.030	.030	.016
2.5	40,000	208.5	4.59	1510	1422	.024	.021	.015
2.2	40,000	176.6	5.60	1420	1303	.020	.018	.010
2.0	65,000	46.6	6.25	1359	1235	.025	.018	.011
2.0	55,000	75.2	6.25	1359	1235	.022	.015	.010
1.53	40,000	100.3	7.98	1207	1078	.018	.011	.010
0.9	40,000	48.1	8.97	1016	901	.018	.010	.008
0	0	147.6	8.42	1122	992	.014	.009	.005

[a] Cooling air for stator blades supplied from combustor secondary air.
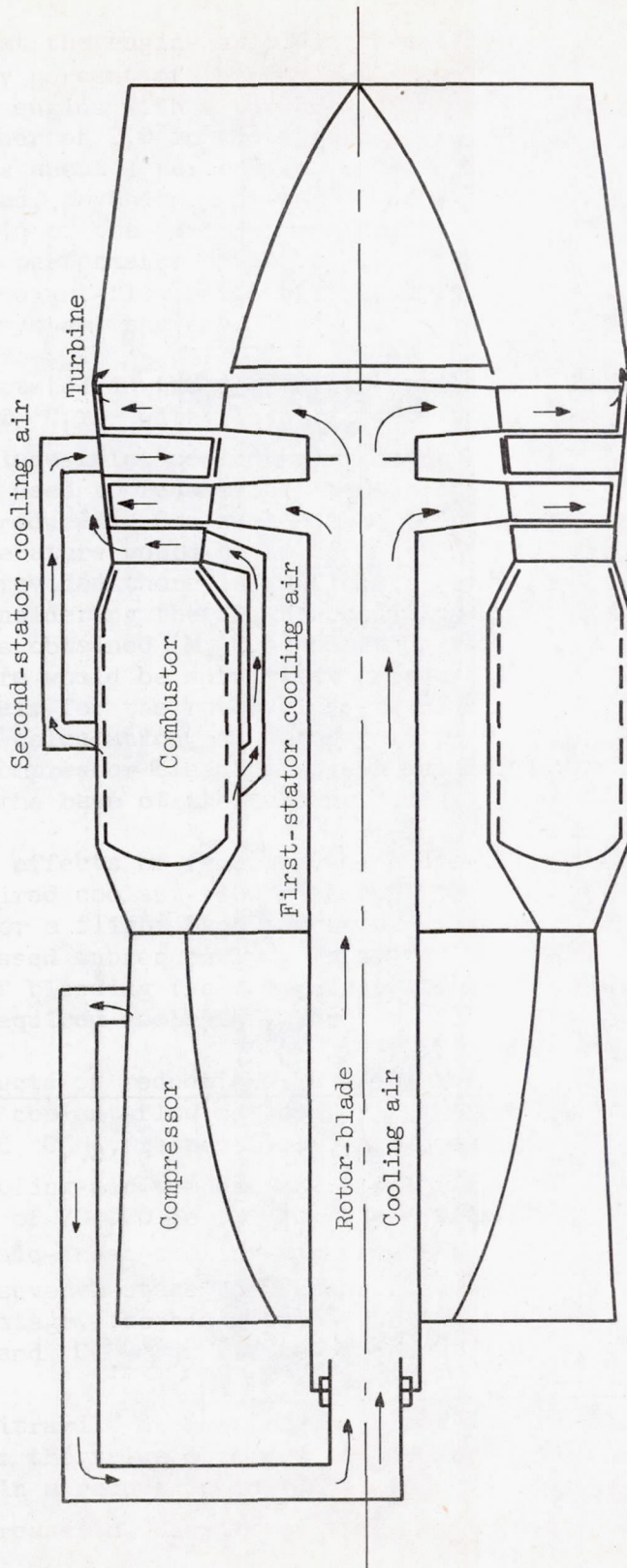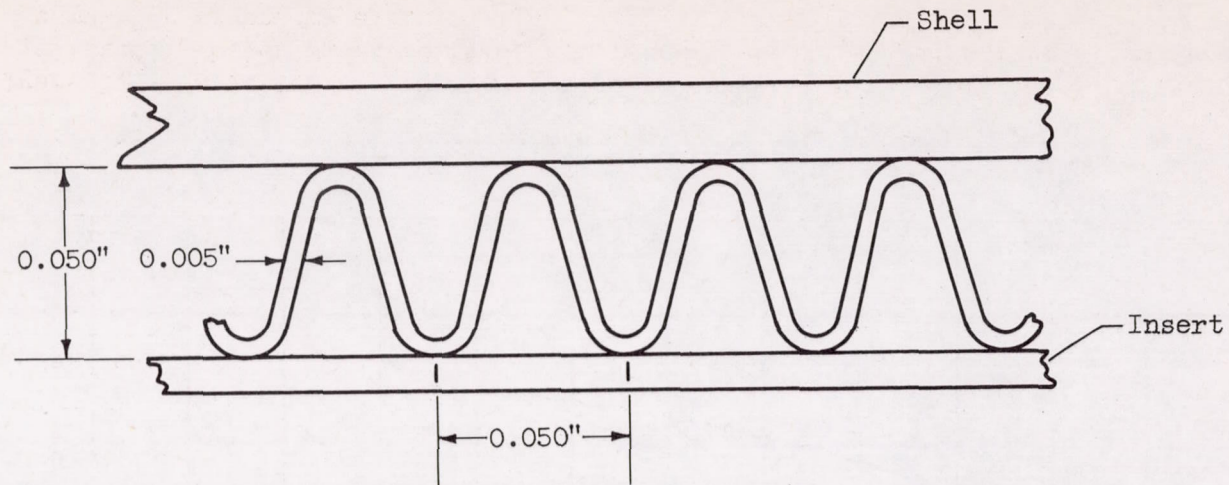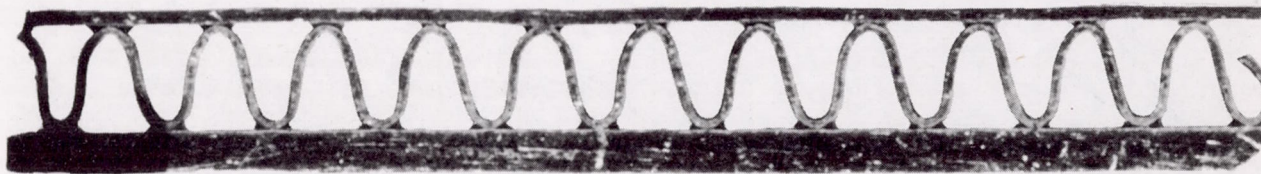[b] Cooling air for rotor blades supplied from seventh compressor stage bleed.

Figure 1. - Coolant system of turbojet engine having two turbine stages.

Shell

0.050" 0.005" →

Insert

←—0.050"—→

(a) Dimensions of corrugations.

C-40965

(b) Sample corrugations.

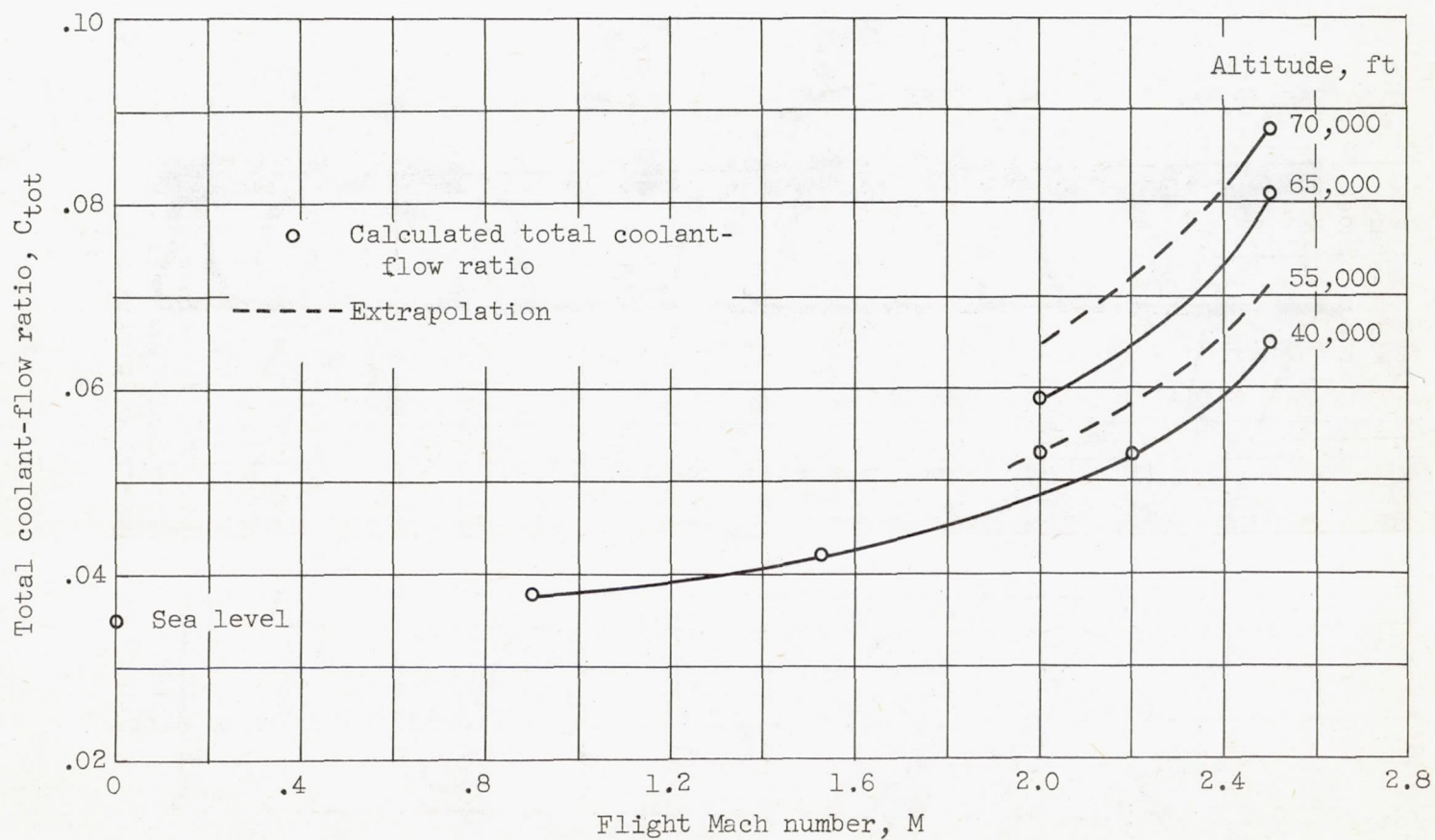Figure 2. - Turbine-blade cooling-air passage configuration.

Figure 3. - Variation of required coolant-flow ratio with flight Mach number and altitude. Rotor bleed point at compressor seventh stage; stator bleed from combustor secondary air.
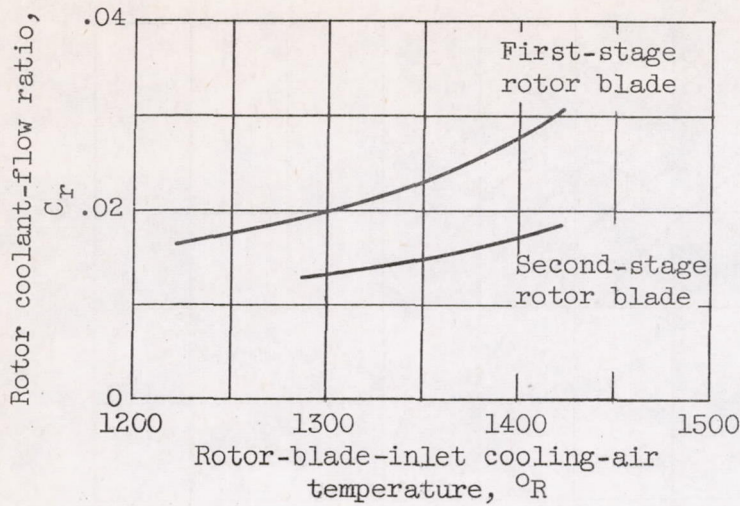
Figure 4. - Effect of blade-inlet cooling-
air temperature on rotor coolant-flow
ratio. Flight Mach number, 2.5; alti-
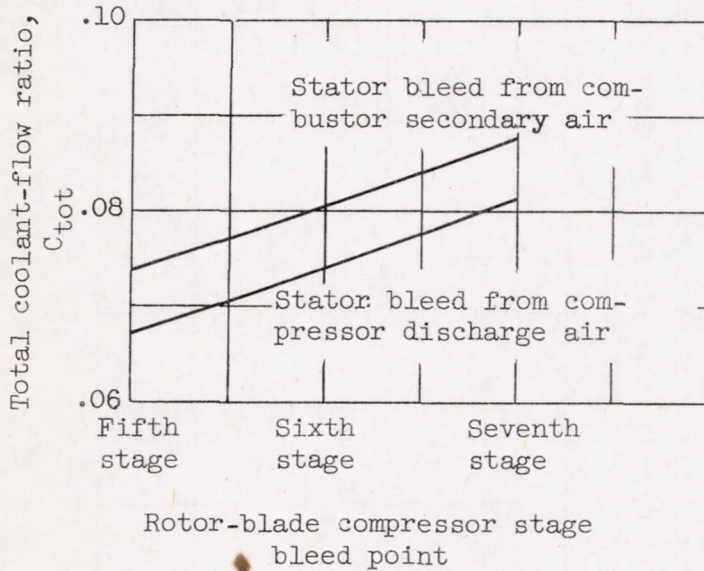tude, 70,000 feet.

Figure 5. - Effect of rotor-blade bleed
point on total coolant-flow ratio.
Flight Mach number, 2.5; altitude,
70,000 feet.