NACA RM E54L20

RESEARCH MEMORANDUM

EFFECTS OF TURBINE COOLING WITH COMPRESSOR AIR BLEED ON GAS-TURBINE ENGINE PERFORMANCE

By Jack B. Esgar and Robert R. Ziemer

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

CLASSIFICATION CANCELLED

Authority Naca Res. abs Date 1-10-57
4 RN-111
By NB 1-30-57 See

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
March 15, 1955

 [redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

EFFECTS OF TURBINE COOLING WITH COMPRESSOR AIR BLEED ON GAS-TURBINE ENGINE PERFORMANCE

By Jack B. Esgar and Robert R. Ziemer

SUMMARY

The engine performance gains made possible by use of turbine cooling to permit engine operation at high turbine-inlet temperatures are generally well known, but information on the engine performance variations resulting from bleeding air from the compressor for turbine cooling is lacking. A thermodynamic-cycle investigation was conducted to determine the magnitude of these performance variations for a wide range of operating conditions for turboprop engines and both afterburning and nonafterburning turbojet engines.

Bleeding air from the compressor for turbine cooling at constant turbine-inlet temperature results in a power (thrust or horsepower) reduction and generally an increase in specific fuel consumption for both turbojet and turboprop engines. The effects are smaller at high than at low turbine-inlet temperatures. For both afterburning and nonafterburning turbojet engines at high flight speeds, reductions in thrust and increases in specific fuel consumption resulting from bleeding a given quantity of compressor air for such purposes as cabin cooling and accessory drives (overboard bleed) are very much greater than for the case where the bleed air is used for turbine cooling. For turboprop engines the effects on engine performance due to bleeding the compressor are approximately the same whether the air is bled overboard or used for turbine cooling.

Generally, the performance variations resulting from compressor bleed for turbine cooling are not prohibitive. The net effect of using air bled from the compressor for turbine cooling to permit operation at higher turbine-inlet temperatures is improved power and specific fuel consumption for turboprop and afterburning turbojet engines. For nonafterburning turbojet engines, very large increases in thrust are obtained by increasing gas temperature; the resulting increases in specific fuel consumption are primarily due to increased temperature and not to air-cooling.

[redacted]

INTRODUCTION

One of the factors affecting the use of turbine cooling to permit operation of gas-turbine engines at higher turbine-inlet temperatures than for present engines, and thus higher power outputs, is the effect of the cooling on the engine power and specific fuel consumption. Therefore, an analytical investigation was conducted at the NACA Lewis laboratory to investigate the effects of bleeding air from the engine compressor for turbine-cooling purposes.

3516

Some of the effects of cooling on engine performance have been previously investigated and reported in references 1 to 5. References 1 and 2 discuss the effects of liquid- and air-cooling on the cycle efficiency of turbine engines delivering shaft power. References 3 and 4 present the effects of air-cooling on the performance of a specific turbojet engine. The primary purpose of reference 5 is to show the performance potentials of turbojet engines utilizing turbine cooling and operating at a flight Mach number of 2 in the stratosphere. In the study of reference 5, only engines with single-stage turbines operating near their aerodynamic limits are considered.

In order to provide studies of a more general nature that include both turbojet and turboprop engines operating over a wide range of conditions, the present investigation was undertaken. This investigation is based on a thermodynamic-cycle analysis; and, with the exception of the component efficiencies, the compressor and turbine characteristics such as aerodynamic limits, flow capacity, or number of stages were not specified. In this manner, the results of the analysis can be generalized to show the effects of air-cooling on engine performance for many conditions of engine design. This report does not emphasize the performance potentials obtainable through use of turbine cooling to permit operation at high turbine-inlet temperatures, but presents the relative performance changes due to air cooling that are obtained at various operating conditions and with various types of gas-turbine engines. The amount of coolant flow required for any given set of conditions is not specified. The quantity of coolant required for a given set of engine conditions can vary greatly as the result of various air-cooled turbine blade design techniques. It was thought, therefore, that the results of this report would be more useful if coolant flow was left an independent variable. In this way the amount of effort that may be necessary to minimize coolant-flow requirements for different modes of operation can be determined.

The effect of cooling on engine performance was studied for turbojet afterburning (to 3500° R) and nonafterburning engines for a range of compressor pressure ratios from 4 to 10, turbine-inlet temperatures from 2000° to 3000° R, flight Mach numbers of zero and 2, compressor adiabatic efficiencies of 83 and 88 percent, turbine polytropic efficiencies from 70 to 90 percent, and with the turbine cooling air from zero to 9 percent of the compressor flow bled from either the compressor discharge or from an intermediate stage. Calculations for turboprop engines were made at the same conditions, except the range of compressor pressure ratios was from 8 to 16 and the flight Mach numbers considered were zero with no jet thrust and 0.9 with optimum jet thrust.

ANALYTICAL PROCEDURE AND ASSUMPTIONS

The amount that engine performance is affected by bleeding air from the compressor for turbine cooling is influenced by other engine variables such as turbine-inlet temperature, compressor and turbine efficiency, compressor pressure ratio, over-all engine pressure ratio as influenced by flight speed, and the compressor stage from which the cooling air is bled. The pressure losses in the cooling air determine the point where the compressor must be bled. In this analysis two compressor bleed points were assumed for the calculations. The low bleed point, which was at a pressure ratio of half the total compressor pressure ratio, would probably be about the minimum that could be expected for a low-pressure-loss system. For a cooling-air system with high pressure losses in the ducting and blades and with a throttle for controlling the flow, compressor-discharge bleed would probably be required. This was the other bleed point considered.

The effect of compressor bleed on air-cooled engine performance was investigated for ranges of the engine variables mentioned using the computational methods presented in reference 6 with some variations to be discussed later. A range of engine conditions was assigned that is believed to be representative of the type of engine operation to be expected for engines using turbine cooling. Calculations for both turbojet and turboprop engines were made that included the following conditions (symbols are given in appendix A, and numerical subscripts indicate stations shown in fig. 1):

Engine variable	Turbojet engine	Turboprop engine
Flight Mach number	0 and 2	0 and 0.9
Flight altitude	Sea level and stratosphere	Sea level
Ram recovery, p_1'/p_0'	1.00 at M = 0 0.85 at M = 2	0.96 at M = 0.9
Compressor pressure ratio (at specified flight conditions), p_2'/p_1'	4, 6, 8, 10	8, 12, 16
Compressor adiabatic efficiency, η_C	0.83 and 0.88	0.83 and 0.88
Compressor bleed point	Compressor discharge and at pressure ratio equal to half compressor pressure ratio	Compressor discharge and at pressure ratio equal to half compressor pressure ratio
Primary-combustor pressure ratio, p_3'/p_2'	0.95	0.95
Primary-combustor efficiency, $\eta_{B,2-3}$	0.98	0.98
Turbine-inlet temperature, °R	2000, 2500, 3000	2000, 2500, 3000
Turbine polytropic efficiency, $\eta_{T,\infty}$	0.70, 0.75, 0.80 0.85, 0.90	0.70, 0.75, 0.80 0.85, 0.90
Turbine tip speed, ft/sec	1200	1200
Turbine-discharge pressure for turboprop with no jet thrust	----	$p_5' = p_0/0.92$
Tail-cone pressure ratio with no afterburner, p_7'/p_5'	0.99	0.99
Afterburner and tail-cone pressure ratio, p_7'/p_5'	0.90	----
Afterburner efficiency, $\eta_{B,6-7}$	0.90	----
Afterburner temperature, °R	3500	----
Type of exhaust nozzle	Convergent-divergent	Convergent
Exhaust-nozzle efficiency, η_n	0.95	0.95
Turboprop gearbox efficiency, η_G	----	0.95
Propeller efficiency, η_p	----	0.85

3516

Unless specified otherwise on the figures, all calculations were made for a compressor adiabatic efficiency of 0.88, a turbine polytropic efficiency of 0.85, and cooling air bled from the compressor discharge for both turbojet and turboprop engines. In addition, the flight Mach number was 2 for the turbojet and zero for the turboprop unless specified differently.

In this analysis, the compressor adiabatic efficiency is assumed to be invarient with compressor pressure ratio. Experience in modern compressors indicates that an adiabatic efficiency of 0.88 can be obtained for a wide range of compressor pressure ratios.

For the turbine, the turbine stage efficiency was assumed to remain constant. As the compressor pressure ratio increases, the turbine work increases, and consequently the greater will be the number of turbine stages required. It was believed that constant turbine polytropic efficiency for all compressor pressure ratios would therefore be a better assumption than constant turbine adiabatic efficiency.

Heat Removed by Turbine Cooling

During the process of cooling turbine blades, heat is removed from the gas driving the turbine and is transferred to the cooling air. The cooling air is usually ducted back into the gas stream so that the heat transferred to the cooling air ultimately finds its way back to the combustion gases. For a turbojet engine, this heat is still available for jet thrust. The heat removed by turbine cooling has a negligible effect on turbojet-engine performance and is therefore neglected for the turbojet calculations.

For the turboprop engine, however, the effect of heat removal from the cooled portion of the turbine has a greater effect on the engine performance than for the turbojet engine. Although small, this effect is accounted for in the calculations, and the method of calculating the quantity of heat removed is discussed in appendix B.

Division of Work Between Cooled and Uncooled Turbine Stages

In a convection-air-cooled gas-turbine engine the cooling air discharges from the tips of the rotor blades. This cooling air mixes with the combustion gases to increase the mass flow and decrease the gas temperature for the following turbine stages. The cooling air from the stator blades probably will not be discharged at the blade tips but instead will be ducted to discharge into the main gas stream downstream of the cooled turbine stage. The quantity of work that can be obtained from the cooling air in subsequent turbine stages is uncertain, because the air is at a low energy level until mixed with the exhaust gases. In the analytical procedure of reference 6, it was assumed that no work was obtained from the rotor or stator cooling air in any turbine stage. In the present analysis, however, it is assumed that no work is obtained from the cooling air in any of the cooled turbine stages, but that the cooling air from both the rotor and the stator is available for producing work in the uncooled turbine stages.

As stated previously, the effect of heat rejection to the cooling air is neglected in the turbojet-engine calculations. For turboprop engines, however, it is assumed that the heat rejected to the cooling air is made available for producing work in the uncooled turbine stages.

The work output of the cooled turbine stages was determined by specification of the gas temperature at the exit of the cooled turbine stages. For some cases for turbojet engines with high turbine-inlet temperatures, the turbine work required for driving the compressor was not high enough to reduce the turbine-exit temperature to the specified temperature. For these special cases all the turbine stages were cooled; and, therefore, the work output of the cooled turbine stages would be the total turbine work determined in the manner explained in reference 6.

For a turbine-inlet temperature of 2000° R, cooling probably would not be required unless the turbine blades were made from noncritical materials. It was assumed that noncritical blades were used at a turbine-inlet temperature of 2000° R and that uncooled turbine blades could be utilized when the gas temperature was reduced to 1600° R. This temperature (1600° R), therefore, was the specified exit temperature for the cooled turbine stages. For turbine-inlet temperatures of 2500° and 3000° R, it was assumed that the turbine blades were made of high-temperature materials and that uncooled turbine blades could withstand stage-inlet temperatures of 2100° R (the specified exit temperature for the cooled turbine stages at the higher gas temperatures).

Optimum Jet Thrust for Turboprop Engine

Reference 7 shows that the jet velocity required to obtain optimum jet thrust for a turboprop engine without compressor bleed can be closely approximated by the expression

$$V_{8,opt} = \frac{\eta_n V_0}{\eta_T \eta_G \eta_p} \tag{1}$$

When air is bled from the compressor, a higher jet velocity is required to obtain optimum jet thrust, because a larger pressure drop is necessary across the turbine, the same as if the turbine efficiency were lowered.

Calculations showed that a jet velocity 10 percent higher than that given by equation (1) resulted in very nearly an optimum jet thrust for the ranges of compressor bleed flow, component efficiency, and flight velocity considered in this report; consequently, this higher jet velocity was used in all optimum jet thrust calculations. Exactly optimum values of jet velocity are not important, however, because the maximum equivalent horsepower and minimum equivalent specific fuel consumption are relatively insensitive to the jet velocity for velocities near the optimum value. The pressure ratio across the exhaust nozzle required to obtain the specified jet velocity, the over-all engine pressure ratio, and the specified pressure losses were then used to determine the pressure ratio available across the turbine for obtaining turbine and shaft power.

Definition of Engine Performance Terms

Throughout this report the engine performance is given in terms of the engine power and the specific fuel consumption. The definition of these terms varies somewhat with different engines as explained in the following paragraphs.

Afterburning and nonafterburning turbojet engines. - For turbojet engines, the power of the engine is always in terms of the net specific thrust in pounds of thrust per pound of compressor-inlet air flow per second. The net thrust is the gross or jet thrust minus the inlet momentum of the air flowing into the engine.

The specific fuel consumption is the fuel flow divided by the net thrust; therefore, it is a thrust specific fuel consumption in pounds of fuel per hour per pound of thrust.

Turboprop engines with no jet thrust. - Without jet thrust, the turboprop-engine power is taken equal to the specific shaft horsepower, which is the horsepower per pound of compressor-inlet air flow per second.

The specific fuel consumption is the fuel flow divided by the shaft horsepower; therefore, it is a brake specific fuel consumption in pounds of fuel per hour per shaft horsepower.

Turboprop engines with jet thrust. - With jet thrust, the turboprop-engine power is taken as the specific equivalent horsepower, which is equivalent horsepower per pound of compressor-inlet air flow per second. The equivalent horsepower of the engine is the sum of the equivalent jet thrust horsepower (product of jet thrust and velocity divided by propeller efficiency with proper conversion units) and the shaft horsepower.

The specific fuel consumption is the fuel flow divided by the equivalent horsepower; therefore, it is an equivalent specific fuel consumption in pounds of fuel per hour per equivalent horsepower.

NONAFTERBURNING TURBOJET ENGINES

As stated in the INTRODUCTION, the primary purpose of this report is to show the performance variations that result from air-cooling of turbines rather than to show the performance potentials that are obtainable through use of cooled turbines to permit higher turbine-inlet temperatures. The best basis for showing performance variations due to cooling is debatable because of the following considerations: The cooling air should be ducted back into the combustion gases after it has served its cooling purposes so that it is available for creating jet thrust. The exhaust-gas temperature is then reduced by dilution from the cooling air. This temperature reduction quite naturally results in a specific thrust reduction. The resulting specific fuel consumption may be either increased or decreased, depending upon the pressure losses in the cooling air. The question then arises whether the effects of air-cooling on engine performance should be determined at a constant thrust level or at constant turbine-inlet temperature. 3516

If comparisons are made at a constant thrust level, the turbine-inlet temperature is a function of the quantity of cooling-air flow, and the basis of comparison is the variation in specific fuel consumption with variations in cooling-air flow. On the other hand, if comparisons are made at constant turbine-inlet temperature, the specific thrust always decreases with increasing quantities of cooling air because of exhaust-gas temperature reduction. The specific fuel consumption may increase, decrease, or be unaffected. It depends to a large extent on cooling-air pressure losses. Comparisons at constant turbine-inlet temperature are probably more useful than comparisons at constant thrust, because there are practical limitations to the amount turbine-inlet temperatures can readily be increased, and because this type of comparison is more familiar. Most air-cooling effects are shown, therefore, at constant turbine-inlet temperature.

Performance with No Cooling Air

In order to permit evaluation of over-all engine performance for a range of turbine-inlet temperatures, the engine performance is first presented on a relative basis in figure 2 for a range of turbine-inlet temperatures for no cooling air. In the remaining figures, the performance variations due to air-cooling at constant turbine-inlet temperature are presented by showing the cooled-engine performance relative to the case with no cooling air with all other engine conditions remaining constant.

The over-all effect of higher turbine-inlet temperatures for a range of compressor pressure ratios can be seen by first observing the performance shown in figure 2 (no cooling air). The net performance for most conditions studied can then be obtained by multiplying the performance indicated in figure 2 by the relative performance figures showing performance variations due to turbine rotor and stator cooling at constant turbine-inlet temperature. In figure 2 the compressor adiabatic efficiency is 0.88, the turbine polytropic efficiency is 0.85, and the flight Mach number is 2 in the stratosphere. By plotting the performance on a relative basis, the results become more general and are approximately correct for somewhat different assumptions concerning efficiencies, pressure losses in the engine, and flight speed. As an example, the results shown in reference 8 are for different assumptions, but on a relative basis the results corroborate those presented herein. In figure 2, all performance is relative to an engine with a compressor pressure ratio of 4 (at the flight condition specified) and a turbine-inlet temperature of 2000° R. The corrected specific thrust is 32.06 pound-seconds per pound, and the corrected specific fuel consumption is 1.162 pounds per hour per pound at this basic condition.

Figure 2 shows that the specific thrust of nonafterburning engines can be approximately doubled by increasing the turbine-inlet temperature from 2000° to 3000° R. This thrust is accompanied by an increase in specific fuel consumption unless the compressor pressure ratio is increased also. The desirability of operating at these higher temperatures is dependent on the use of the power plant. Generally, high temperatures for turbojet engines are most useful at supersonic flight speeds, as discussed in references 9 and 10. At these speeds, high thrust per unit of engine weight or per unit of frontal area can more than overbalance the effects of increased specific fuel consumption at high turbine-inlet temperatures. Because it is expected that high turbine-inlet temperatures will be utilized in turbojet engines at supersonic speeds, most of the study presented herein for turbojet engines is for a flight Mach number of 2 in the stratosphere.

Performance Variations due to Turbine Cooling

Bleeding air from the compressor for cooling of the turbine results in a change in turbojet-engine performance for three main reasons:

(1) Dilution of the exhaust gases by the cooling air causes a gas-temperature reduction. The exhaust-gas temperature is also reduced because part of the engine air flow (the cooling air for the turbine rotors and stators) is unavailable for turbine work, although work is done on this air in compressing it. As a result, specific turbine work is increased and there is a higher gas-temperature drop across the turbine. The gross thrust is proportional to the square root of the exhaust-gas temperature; consequently, a reduction in gas temperature results in a reduction in thrust.

(2) Higher specific turbine work for a given turbine-inlet temperature causes an increased pressure drop across the turbine. Because of this higher pressure drop less pressure is available at the exhaust nozzle for creating jet thrust.

(3) The cooling air for the turbine rotor has to be accelerated to turbine tip speed as it flows out through the turbine blades and is discharged at the blade tips, so that the turbine must do additional work on the rotor cooling air. Although this additional work is small, an additional loss in jet thrust is incurred, as explained in items (1) and (2). 3516

The effect of air-cooling on specific fuel consumption is generally smaller than the effect on thrust. Since part of the compressor air is bled off for cooling, less fuel per pound of compressor-inlet air is burned for a cooled engine than for an uncooled engine for a constant turbine-inlet temperature. Thus, the thrust specific fuel consumption is affected to a smaller degree than the specific thrust by air-cooling, since fuel consumption as well as engine thrust is decreased.

In the following discussion, the effects of turbine rotor cooling on turbojet-engine performance are first shown for ranges of turbine-inlet temperature, compressor pressure ratio, and compressor and turbine efficiency. The effects of both turbine rotor and stator cooling on engine performance are then compared at specific values of turbine-inlet temperature, compressor pressure ratio, and component efficiencies to show effects of flight speed and cooling-air pressure losses.

Effects of turbine cooling are shown for coolant flows up to 9 percent of the compressor flow for both rotors and stators. This range should cover requirements for most turbine blades for the gas-temperature range covered in this report. The quantity of coolant required for operation at a specified turbine-inlet temperature can vary greatly with the turbine blade coolant-passage configuration, blade size, and blade stress. For reference, it is expected that good design practice can result in the following range of coolant-flow requirements:

Turbine-inlet temperature, $^{\circ}R$	Total turbine coolant flow, percent of compressor flow
2000	1 to 4 (noncritical blades)
2500	3 to 6
3000	6 to 15

The rotor and stator blades generally require about the same amount of cooling air up to turbine-inlet temperatures on the order of 2500° R. At higher turbine-inlet temperatures, it is expected that the stators may require more flow than the rotors, based on unpublished calculations.

<u>Effects of turbine-inlet temperature and compressor pressure ratio</u>. - In figure 3 relative specific thrust and relative thrust specific fuel consumption are plotted against the percent of compressor air bled for turbine rotor cooling. At zero coolant flow the relative values plotted on the curve ordinates are always unity; in this way the percentage changes due to cooling are easily obtained. The calculations were made for a non-afterburning turbojet engine operating at a flight Mach number of 2 in the stratosphere. Parameters on the curve cover ranges of turbine-inlet temperature and compressor pressure ratio. These curves cannot be used for comparing the thrust or fuel consumption that results from operation at different temperature levels or different compressor pressure ratios. Such information is obtainable from figure 2.

It will be observed in figure 3 that the percentage changes in specific thrust and specific fuel consumption for each percent of coolant flow bled from the compressor decrease as the turbine-inlet temperature increases, particularly as the temperature increases from 2000° to 2500° R. At a turbine-inlet temperature of 2000° R, the thrust reduction due to turbine rotor cooling is considerably higher for a compressor pressure ratio of 10 than for 4, and the percentage increase in specific fuel consumption is more than doubled at the higher pressure ratio. At higher turbine-inlet temperature, compressor pressure ratio has a small effect on performance variations due to cooling.

At a turbine-inlet temperature of 2500° R, which appears to be a reasonable goal for the first round of engines incorporating turbine cooling, the engine specific thrust decreases approximately 1 percent for every percent of turbine rotor cooling air. The specific fuel consumption increases only about 1 percent for every 6 percent of compressor air bled for cooling.

In order to illustrate the combined effect of increased turbine-inlet temperature and compressor air bled for turbine rotor cooling on the performance of a nonafterburning turbojet engine, a map of relative specific thrust plotted against relative specific fuel consumption is presented in figure 4 for ranges of turbine-inlet temperature, compressor pressure ratio, and percent compressor flow used for turbine rotor cooling. All values of specific thrust and specific fuel consumption are relative to the values obtained at a compressor pressure ratio of 4 at a turbine-inlet temperature of 2000° R with no cooling air. From this plot, it is obvious that, even for high quantities of coolant flow, the thrust output of turbojet engines can be substantially increased by using air-cooling to permit operation at higher turbine-inlet temperatures. Generally, the effects of air-cooling on specific fuel consumption are very small at the higher turbine-inlet temperatures; however, efforts should be made to minimize coolant flow in the effort to obtain higher specific thrust.

Effects of compressor and turbine efficiency level. - Both the thrust and fuel consumption of an engine are improved by use of high component efficiencies in the compressor and turbine. The question arises, however, as to whether efficiency level has an effect on variations in thrust and specific fuel consumption due to turbine cooling. Figure 5 shows that for turbine rotor cooling the efficiency level does not significantly affect these variations for the complete range of turbine-inlet temperature shown. It will be noted, however, that the effect of efficiency level is somewhat more pronounced at a turbine-inlet temperature of 2000° R than at the higher temperatures.

Compromises between turbine efficiency and coolant flow. - Some types of turbine blades may be difficult to cool because of difficulty in locating adequate coolant-passage area in some portions of the blade. Added area in these regions may possibly reduce coolant-flow requirements but alter the blade profile with an attendant loss in turbine efficiency. The use of transpiration cooling can also reduce coolant-flow requirements, but bleeding cooling air into the boundary layer may decrease the aerodynamic efficiency of the blades. It may be necessary, therefore, to compromise turbine aerodynamic efficiency in order to provide a blade that will cool with a smaller quantity of cooling air. The extent to which this compromise should be accomplished is indicated in figure 6 for a turbine-inlet temperature of 2500° R, a compressor pressure ratio of 6, and a flight Mach number of 2 in the stratosphere. It should be noted that figures 5 and 6 are used to show two completely different effects. Figure 5 shows the percentage variation in engine performance due to cooling at two different efficiency levels, but the figure does not show how efficiency affects performance. Figure 6, on the other hand, shows how both turbine efficiency and coolant flow affect engine performance.

The loss in thrust for each percent of cooling air is approximately the same as for 3 points decrease in turbine efficiency (fig. 6). This indicates that, with respect to thrust, the turbine efficiency could possibly be compromised in order to permit fabrication of blades that would require smaller quantities of coolant flow. With respect to fuel consumption, however, figure 6 indicates that the turbine efficiency cannot be compromised, because the increase in specific fuel consumption for each percent of cooling air is approximately the same as for a decrease of 1/2 point in turbine efficiency. It is very doubtful whether alterations to the blade that would cause such small changes in efficiency could result in a type of blade where the coolant flow could be reduced by the quantities indicated in figure 6 for no change in specific fuel consumption. Study of this figure indicates, therefore, that turbine blade-cooling research must be directed toward methods of cooling blades efficiently without altering the aerodynamic performance in an adverse manner.

Comparison of engine performance variations due to turbine rotor and stator cooling. - In cooling the turbine stator, no additional pumping work to accelerate the air to the turbine tip speed is required; therefore, the effects on engine performance due to stator cooling are somewhat smaller than for rotor cooling. These effects are shown in figure 7(a) for a non-afterburning turbojet engine with a turbine-inlet temperature of 2500° R, a compressor pressure ratio of 6, and a flight Mach number of 2 in the stratosphere. It is shown that stator cooling affects engine performance only slightly less than rotor cooling. For either case, each percent of
3516 cooling air results in about 1 percent loss in thrust and only a very slight increase in specific fuel consumption. For most cases the total effect of cooling both the rotor and stator can be closely approximated by adding the individual effects on engine performance.

Effect of flight Mach number. - The effects of turbine cooling on engine performance are compared in figure 7(b) for flight Mach numbers of zero and 2. Altitude varies simultaneously with flight speed, but its effect is of small magnitude, so that the conclusions drawn from the study are unaffected by the altitude. The thrust reduction resulting from turbine cooling at sea-level static conditions (Mach number of zero) is about four-fifths that for a Mach number of 2 in the stratosphere. At low flight speeds the specific fuel consumption can actually improve as the result of turbine cooling. The reason for improved performance at low flight speeds can be explained from the following considerations: Turbine cooling has approximately the same effect on gross thrust at high or low flight speeds. The net engine thrust is determined from the gross thrust minus the inlet momentum of the air taken into the engine. At high flight speeds this inlet momentum may be in excess of 50 percent of the gross thrust. Because of the inlet momentum, small percentage changes in gross thrust can result in much larger percentage changes in net thrust. Consequently, larger percentage net thrust reductions due to cooling occur at high flight speeds. At a flight Mach number of 2, the specific fuel consumption increases slightly with increasing coolant flow. Smaller percentage thrust reductions due to cooling at lower flight speeds cause specific fuel consumption to decrease with increasing coolant flow as shown at a flight Mach number of zero.

Effects of cooling-air pressure losses. - The higher the pressure losses in the cooling-air passages, the higher will be the pressure at which the compressor must be bled. To illustrate the effect of these pressure losses on engine performance a comparison is made in figure 7(c) for two compressor bleed points - with air bled from the discharge of the compressor and from a point where the compressor bleed pressure ratio is half the total compressor pressure ratio. The figure shows that a reduction of pressure losses results in a small improvement in thrust specific fuel consumption. This improvement is enough that, even at high flight speeds (flight Mach number of 2), cooling can result in improved fuel consumption. It will also be observed that the engine thrust improves somewhat when the compressor is bled at the lower pressure ratio that may be possible with low cooling-air pressure losses. The compressor bleed pressure ratio has a small effect on thrust reduction due to turbine cooling, because the work required for compressing the cooling air provides only a portion of the thrust reduction due to cooling. Generally, the primary factor causing thrust reduction is the reduction of exhaust-gas temperature due to dilution by the cooling air rather than the effects of larger pressure and temperature ratios across the turbine as the result of increased turbine work due to cooling-air compression. An additional advantage of bleeding air at lower pressure ratios, of course, is the fact that the cooling air has a lower temperature, and turbine cooling can thus be accomplished more effectively.

3516

Comparison of turbine cooling and overboard bleed. - A prevalent practice with turbojet engines is to bleed air from the compressor for cabin cooling, accessory drives, electronic equipment cooling, and so forth. Air used for these purposes cannot be ducted back into the exhaust gases to obtain jet thrust. Therefore, reductions in thrust due to overboard bleed are larger than those due to turbine cooling, as illustrated in figure 7(c). The reduction in thrust due to overboard bleed results in large increases in specific fuel consumption, a percentage increase of as much as twice the percentage of air bled from the compressor. At zero flight speed the increase in thrust specific fuel consumption due to overboard bleed is only about one-fifth of that shown on figure 7(c) (resulting from elimination of the inlet momentum loss), but the fuel consumption is still considerably higher than for turbine cooling. These results show that, for a given quantity of bleed, the common use of overboard bleed is very costly in engine performance compared with the use of turbine cooling.

AFTERBURNING TURBOJET ENGINES

Performance with No Cooling Air

In a manner similar to that for the nonafterburning turbojet engine, the engine performance for the afterburning engine is first shown for a range of turbine-inlet temperatures and compressor pressure ratios for no cooling air. Performance is then shown with cooling air where the cooled engine performance is relative to the uncooled performance. The over-all effect of increasing turbine-inlet temperature and using air-cooling can be obtained by multiplying the relative performance values shown for engines with no cooling air (fig. 8) by the relative performance of engines utilizing turbine cooling (figs. 9 and 10).

The relative performance of afterburning engines compared with non-afterburning engines is shown in figure 8 for ranges of turbine-inlet temperature and compressor pressure ratio. All performance values are relative to a nonafterburning engine with a compressor pressure ratio of 4 and a turbine-inlet temperature of 2000° R (the same as shown in fig. 2). Afterburning engines produce much higher values of specific thrust than nonafterburning engines because of the higher exhaust-gas temperatures. This thrust is obtained at high cost in specific fuel consumption. Increasing the turbine-inlet temperature has the combined effect of increasing the thrust and decreasing the thrust specific fuel consumption of the afterburning turbojet engine. These performance improvements accompanying increased turbine-inlet temperatures are caused by (1) lower pressure ratios across the turbine and thus higher pressures in the exhaust nozzle resulting from increased temperatures ahead of the turbine, and (2) more efficient burning of the fuel because a larger percentage of the fuel is burned in the primary combustor, which has a higher combustion efficiency than the afterburner.

3516

Performance Variations due to Turbine Cooling

Air-cooling affects the performance of the afterburning turbojet engine in a manner similar to that discussed previously for the nonafterburning engine, with one important exception; namely, that reduction of the combustion-gas temperature downstream of the turbine by dilution and increased turbine work does not affect the thrust, because the gas temperature is raised to a predetermined level by afterburning. The only effect on thrust is due to decreased exhaust-nozzle pressure resulting from increased turbine work. For these reasons, turbine cooling will have a smaller effect on the thrust of afterburning engines than of nonafterburning engines. The reduction of the turbine-exit temperature by cooling necessitates the use of more fuel than when no cooling is used, in order to raise the exhaust-gas temperature to a specified afterburner-outlet temperature. As a result, the increase in specific fuel consumption due to turbine cooling will usually be higher for afterburning engines than for nonafterburning engines. Some of the effects are illustrated by comparing figure 9 for afterburning engines with figure 3 for nonafterburning engines. The thrust reduction due to turbine rotor cooling for afterburning engines is less than half that for nonafterburning engines. At turbine-inlet temperatures in excess of 2500° R, turbine cooling results in larger percentage increases in thrust specific fuel consumption for afterburning engines than for nonafterburning engines; but at 2000° R, larger increases in specific fuel consumption can occur for nonafterburning engines at high compressor pressure ratios. Generally, the effect of compressor pressure ratio on performance variations due to turbine cooling is very similar for both afterburning and nonafterburning engines. Reductions in thrust and increases in specific fuel consumption are somewhat higher at a compressor pressure ratio of 10 than at a pressure ratio of 4.

There is such a similarity between the turbine-cooling effects for afterburning and nonafterburning engines that the effect of most variations in engine operation and component efficiency require no further discussion. In figure 10, however, are shown comparisons of effects of bleeding at two points on the compressor air for turbine rotor cooling, stator cooling, and overboard bleed. As discussed for the nonafterburning turbojet engine, the compressor bleed point is determined by the pressure losses in the cooling air. Comparison of figures 7(c) and 10 shows that trends for afterburning and nonafterburning engines are the same, but the magnitude of performance changes due to cooling are different. For a given quantity of compressor bleed, the performance variations due to turbine cooling are considerably smaller than for overboard bleed, for the same reasons discussed for the nonafterburning turbojet engine.

Figure 10 shows that a reduction in cooling-air pressure losses, which will permit bleeding the compressor at a lower pressure ratio, results in improvements in specific fuel consumption, but the improvements do not result in a decrease in specific fuel consumption with increasing coolant flow as was shown previously for the nonafterburning engine (fig. 7(c)). Even though turbine cooling has the effect of increasing the specific fuel consumption for afterburning engines, increases in turbine-inlet temperature decrease the specific fuel consumption (fig. 8), so that the combined effect of increasing turbine-inlet temperature through the use of turbine cooling is an improvement in both specific thrust and specific fuel consumption.

TURBOPROP ENGINES

Performance with No Cooling Air

Figure 11 for turboprop-engine performance, which is similar to figures 2 and 8 for turbojet engines, shows the effect of turbine-inlet temperature with no cooling-air bleed for compressor pressure ratios of 8, 12, and 16. All performance values are relative to an engine with a compressor pressure ratio of 8 and a turbine-inlet temperature of 2000° R, resulting in a corrected specific horsepower of 100.05 horsepower per second per pound and a corrected brake specific fuel consumption of 0.557 pound per horsepower-hour. (Thrust horsepower can be obtained by assigning a propeller efficiency.) The compressor pressure ratios are believed to cover the desirable range for this type engine. Slight improvements in over-all cycle efficiency are obtainable at pressure ratios in excess of 16, but engine design and operation problems can become quite severe at high pressure ratios.

The use of high turbine-inlet temperatures is particularly desirable for turboprop or other shaft-power turbine engines, because the power increases and the specific fuel consumption decreases as turbine-inlet temperature is increased. As will be discussed later, this trend continues even when cooling losses are included.

Performance Variations due to Turbine Cooling

Bleeding air from the compressor for turbine cooling in the turboprop engine results in effects on engine performance that are somewhat different from those for the turbojet engine. The effects are as follows:

(1) For a constant compressor pressure ratio, the specific turbine work of the turboprop engine is essentially determined by the turbine-inlet temperature, because the exhaust gases are expanded across the turbine to as low a pressure as practical. By bleeding part of the compressor air for cooling purposes, less mass flow is available at the turbine. At constant turbine-inlet temperature and constant compressor pressure ratio (constant turbine specific work), the turbine mass-flow reduction causes a gross turbine power reduction that is directly proportional to the decrease in turbine mass flow.

(2) Concurrent with and in addition to the effect in item (1), a greater proportion of the gross turbine power is required for compressing air, because part of the air being compressed is not used as turbine mass flow, and additional work is done on the turbine rotor air in accelerating it to turbine tip speed. The net turbine power is therefore further decreased.

(3) Cooled turbine stages remove heat from the combustion gases and reject it to the cooling air. The air used for cooling the turbine rotor is usually discharged at the blade tip and mixes with the combustion gases. The heat rejected to the cooling air is thus returned to the combustion gases, but the lower temperature of the cooling air relative to the combustion gases results in a reduction in gas temperature. For multistage turbines, the effects of heat removal and combustion-gas dilution are to decrease the gas temperature for the following stages. Since the specific turbine work is directly proportional to the stage turbine-inlet temperature for a given turbine pressure ratio, the reduced gas temperature results in reduced specific work for the stage. Since in subsequent turbine stages some work can probably be obtained from the cooling air that mixes with the combustion gases, the loss in turbine power is less than if the air had been bled from the compressor and thrown away.

The brake specific fuel consumption of the turboprop engine is not affected by item (1), because the fuel-flow rate (for constant turbine-inlet temperature) decreases directly as the turbine mass flow decreases, but the brake specific fuel consumption is increased because of power losses mentioned in items (2) and (3).

Effects of turbine-inlet temperature and compressor pressure ratio. - The effects of air-cooling the turbine rotors of turboprop engines are shown for ranges of turbine-inlet temperature and compressor pressure ratio in figures 12 and 13 in a manner similar to that shown previously for the nonafterburning turbojet engine. The level of the compressor pressure ratio has a negligible effect on the variations in specific thrust and specific fuel consumption that occur due to turbine cooling. Comparison of figures 3 and 12 shows that turbine-inlet temperature has the same general effect for turbojet and turboprop engines, but the effects of air-cooling result in about a 50 percent greater power reduction for the turboprop engine than for the turbojet engine. (Power refers to thrust for the turbojet and horsepower for the turboprop.) The percentage increases in specific fuel consumption due to turbine cooling are even greater for the turboprop engine. 3516

Figure 13 shows a map type of performance plot for the turboprop engine for ranges for turbine-inlet temperature, compressor pressure ratio, and percent compressor air bled for turbine rotor cooling. It will be observed that using cooling to permit higher turbine-inlet temperatures for the turboprop engine has even greater advantages than for the turbojet engine (fig. 4), because the power output can be greatly increased and at the same time the brake specific fuel consumption can be decreased.

Effects of compressor and turbine efficiency level. - Comparison of figures 5 and 14 for turbojet and turboprop engines, respectively, shows that the levels of compressor and turbine efficiency have approximately the same effect on engine performance variations resulting from turbine cooling for both types of engines. The principal difference between the two types of engines is the magnitude of the power reductions and specific-fuel-consumption increases that result from turbine cooling. The larger variations occur with the turboprop engine.

Compromises between turbine efficiency and coolant flow. - The question of compromising turbine aerodynamics to permit use of a blade that may be easier to cool in the turboprop engine can be at least partially answered by reference to figure 15. The loss in power output of the engine for each percent of compressor bleed for turbine rotor cooling is approximately the same as for each point decrease in turbine efficiency. The increase in specific fuel consumption for each percent of turbine rotor coolant flow is approximately the same as for each 0.4 point in turbine efficiency. It is doubtful, therefore, whether gains in engine performance would be possible by relaxing on the aerodynamic design to permit a better cooling design.

Figure 15 also shows the importance of high turbine efficiency for turboprop engines. A decrease of 5 points in turbine efficiency can result in from 10- to 17-percent increase in fuel consumption. For a turbojet engine (fig. 6), a decrease of 5 points in turbine efficiency results in only about a 2-percent increase in fuel consumption.

Comparison of engine performance variations due to turbine rotor and stator cooling. - Stator cooling has a smaller effect on engine performance than rotor cooling for turboprop as well as turbojet engines because of the expenditure of less turbine work on the cooling air. The variations in turboprop-engine performance due to rotor and stator cooling are shown in figure 16(a) for a turbine-inlet temperature 2500° R and a compressor pressure ratio of 12 at sea-level static conditions. Comparison of this figure with figure 7(a) for the turbojet engine shows that the difference in engine performance due to rotor and stator cooling is somewhat higher for the turboprop engine, particularly with regard to specific fuel consumption. For rotor cooling, the specific fuel consumption increases almost 1 percent for each percent of cooling air. For stator cooling, the specific fuel consumption increases slightly over 1/2 percent for each percent of cooling air.

Effect of flight Mach number. - The total power for a turboprop engine is the sum of the propeller power and the thrust power obtained from the exhaust jet. The jet-thrust power is a function of the flight speed, and there is an optimum jet thrust at each flight speed that results in maximum total power and minimum specific fuel consumption. At a flight Mach number of zero, there is no jet-thrust power, and maximum power from the engine is obtained by taking a maximum expansion across the turbine. When jet thrust is utilized, however, the air used for turbine cooling can be expanded through the exhaust nozzle to regain some of the energy in the air. A comparison is made in figure 16(b) of the sea-level performance of air-cooled turboprop engines at a flight Mach number of zero with no jet thrust and for a flight Mach number of 0.9 with optimum jet thrust. The variations in specific power and specific fuel consumption due to turbine cooling are decreased when jet thrust is utilized, but the improvements are of small magnitude. It can be concluded, therefore, that the use of jet thrust does not have a large effect on percentage engine performance variations that occur as the result of cooling either the turbine rotor or stator.

Effects of cooling-air pressure losses. - The effects of cooling-air pressure losses, which in turn affect the compressor bleed point, are shown in figure 16(c) for the turboprop engine. The trends shown are similar to but larger than those for the afterburning and nonafterburning turbojet engines (figs. 7(c) and 10). Lower pressure losses result in improved power and brake specific fuel consumption. The increase in specific fuel consumption due to stator cooling with bleed at a pressure ratio half that of the total compressor pressure ratio is less than half that resulting when the compressor is bled from the discharge. The effect of reducing the bleed pressure for turbine rotor cooling is appreciable but somewhat less spectacular than for turbine stator cooling.

Comparison of turbine cooling and overboard bleed. - The effects on turboprop-engine performance of bleeding air for turbine cooling and for overboard bleed are also shown in figure 16(c). The difference between the engine performance variations due to turbine cooling and due to overboard bleed are not nearly so great in the turboprop engine as in the

turbojet engine. The relative magnitude of these performance variations is dependent on the turbine work that can be obtained from the cooling air in turbine stages subsequent to the cooled stages. In this particular analysis, it was assumed that not all the turbine stages in the multistage turbine were cooled. After the combustion gases were reduced to 2100° R by turbine heat extraction due to work and heat rejection to the cooling air, the remaining turbine stages were assumed to be uncooled. In addition, it was assumed that the turbine cooling air was available for doing work in the uncooled stages. Because of this work in the uncooled stages, the power loss due to turbine cooling is less than for overboard bleed 3516
(fig. 16(c)).

If it is assumed that some work can be obtained from the cooling air in some of the cooled stages, better cooled engine performance would be shown. If, however, it is assumed that no work is obtained from the cooling air, the performance of the cooled turbine would be worse than for overboard bleed for turbine rotor cooling, and the performance for stator cooling and overboard bleed would be approximately the same. Because the comparison between the performance variations for cooled turbines and overboard bleed is dependent to such a great extent on the assumptions of the analysis, the only conclusion that can be drawn is that both types of bleed have approximately the same effect on engine performance.

SUMMARY OF RESULTS

The results of this analytical investigation on the effects of bleeding air from the compressors of turbojet and turboprop engines for turbine cooling or other purposes for a range of turbine-inlet temperatures up to 3000° R can be summarized as follows:

1. For nonafterburning turbojet engines at a turbine-inlet temperature of 2500° R and a flight Mach number of 2 in the stratosphere, each percent of air bled from the compressor for turbine-cooling purposes results in approximately 1-percent decrease in thrust and only a slight increase in specific fuel consumption. For afterburning engines at constant afterburning temperature, the thrust reduction is cut in half, but the percentage increase in specific fuel consumption may be as much as half the percentage of cooling air used for turbine cooling. For turboprop engines at 2500° R turbine-inlet temperature and sea-level static conditions, turbine cooling results in larger percentage power reductions and specific-fuel-consumption increases than for turbojet engines.

2. For both turbojet and turboprop engines, the percentages of power (thrust or horsepower) reduction and specific-fuel-consumption increase are smaller at high than at low turbine-inlet temperatures for each percent of cooling air bled from the compressor. For nonafterburning turbojet engines, turbine cooling can result in improved specific fuel consumption under some conditions of operation.

3. Interstage compressor bleed results in slightly lower power reductions and specific-fuel-consumption increases than compressor-discharge bleed.

4. With respect to engine performance, it does not appear feasible to sacrifice turbine aerodynamic performance in order to produce turbine blades that would cool with smaller quantities of cooling air.

5. For both afterburning and nonafterburning turbojet engines at high flight speeds, reductions in thrust and increases in specific fuel consumption resulting from bleeding a given quantity of compressor air overboard for such purposes as cabin cooling and accessory drives are very much greater than for the case where the same amount of air is used for turbine cooling. For turboprop engines, overboard bleed air and turbine-cooling air have approximately the same effect on engine performance.

6. Generally, the performance variations resulting from compressor air bleed for turbine cooling are not prohibitive. The net effect of using compressor bleed air for turbine cooling to permit operation at higher turbine-inlet temperatures is improved power and specific fuel consumption for turboprop engines and afterburning turbojet engines, compared with the performance attainable at turbine-inlet temperatures that are feasible without turbine cooling. For nonafterburning turbojet engines, very large increases in thrust are obtained by increasing temperature; the resulting increases in specific fuel consumption are primarily the result of increased temperature, and not air-cooling.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 9, 1954

APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
b	blade length, ft
c	blade chord, ft
D	diameter, ft
g	acceleration due to gravity, ft/sec^2
H	gas-to-blade heat-transfer coefficient, Btu/(sec)(sq ft)($^\circ$R)
l_o	blade outside perimeter, ft
M	Mach number
N	number of blades
p	pressure, lb/sq ft
Q	heat flow, Btu/sec
R	gas constant, ft-lb/(lb)($^\circ$R)
T	temperature, $^\circ$R
V	velocity, ft/sec
w	flow rate, lb/sec
γ	ratio of specific heats
η	component efficiency
ρ	density, lb/cu ft
σ	solidity, $cN/\pi D_m$

Subscripts:

B	burner

b	blade
C	compressor
e	effective
F	flow
G	gearbox
g	gas
m	mean
n	nozzle
opt	optimum
p	propeller
s	surface
T	turbine
∞	polytropic
0, 1, 2, 3, 4, 5, 6, 7, 8	stations in engine, see fig. 1

Superscript:

'	stagnation conditions

APPENDIX B

METHOD OF CALCULATING HEAT REMOVAL BY COOLED TURBINE BLADES

In order to keep the calculations on a general basis, a relation between turbine flow area and blade heat-transfer surface area was calculated in the following manner: It was assumed that the blade solidity $\sigma = 1.3$ at the mean turbine diameter and the turbine stage-exit axial gas Mach number $M = 0.6$. Then 3516

$$\text{Gas-flow area, } A_F = \pi D_m b \tag{B1}$$

$$\text{Number of blades, } N = \frac{\pi D_m \sigma}{c} = \frac{1.3\pi D_m}{c} \tag{B2}$$

$$\text{Rotor blade surface area, } A_s = l_o bN \approx 2cbN \approx 2.6b\pi D_m \tag{B3}$$

Combining equations (B1) and (B3),

$$A_s = 2.6A_F \tag{B4}$$

The gas-flow rate is

$$w = \rho A_F V = MpA_F \sqrt{\frac{\gamma g}{RT}} \tag{B5}$$

For a Mach number of 0.6 and a mean value of γ of 1.31, $T = 0.948T'$ and $p = 0.796p'$. Then substituting numerical values into equation (B5),

$$w = \frac{0.436A_F p'}{\sqrt{T'}} \tag{B6}$$

Combining equations (B4) and (B6),

$$\frac{A_s}{w} = \frac{5.96\sqrt{T'}}{p'} \tag{B7}$$

where the temperature T' and the pressure p' are at the stage exit. The heat per pound of gas rejected to each stage of the cooled turbine rotor blades is

$$\frac{Q}{w} = H\frac{A_s}{w}(T_{g,e} - T_b) = \frac{5.96H\sqrt{T'}}{p'}(T_{g,e} - T_b) \tag{B8}$$

where the gas-to-blade heat-transfer coefficient H is calculated using reference 11 and the same assumptions listed herein with the additional stipulation that the turbine tip diameter is 30 inches and the blade aspect ratio is 2.0. The effect of heat rejection is small; consequently, the specification of geometry for the heat-rejection calculations does not significantly affect the generality of the analysis on engine performance. The number of stages requiring cooling was determined by assuming a gas-temperature drop of 450° R per stage. It was assumed that the rotor and stator surface areas were the same and that gas-to-blade heat-transfer coefficients were 10 percent higher for the stator than for the rotor. The effective gas temperatures for the stators were assumed to be 50° R less than the gas total temperature. For the rotor blades, the effective gas temperature was assumed to be 250° R less than the total temperature. For turbine-inlet temperature of 2000° R, the cooled rotor blade temperature was assumed to be 1500° R. For turbine-inlet temperatures of 2500° and 3000° R, the cooled rotor blade was assigned a temperature of 1700° R. For both cases, the stator blade temperatures were 200° R higher than the rotor blade temperature.

Heat-rejection rates calculated in this manner are considerably smaller than indicated in references 1 and 2; consequently, the effects of heat removal on engine performance due to cooling are considerably smaller. In references 1 and 2 the heat-rejection rates are based on an analysis presented in reference 12, where an analogy is drawn between skin friction and heat transfer. In that analysis it was assumed that all inefficiencies in the turbine were caused by skin friction. Reference 13 points out that the principal losses occurring in turbine blade rows are (1) profile loss resulting from skin friction, (2) secondary-flow loss, (3) tip-clearance loss, and (4) annulus loss resulting from friction in the inner and outer turbine shrouds. In addition, reference 13 states that, for most blade profiles that have been used in turbines, the total losses have been many times the loss which would be obtained from consideration of skin friction alone. The only loss that could be used for drawing an analogy between friction and heat transfer to the turbine blades would be the profile loss. Since frictional losses are much smaller than assumed in reference 12, the heat-rejection rates must also be smaller. Calculations indicate that the heat-rejection rates will be on the order of one-fourth to one-third of the values used in references 1 and 2.

REFERENCES

1. Rohsenow, W. M.: The Effect of Turbine Blade Cooling on the Efficiency of a Simple Gas Turbine Power Plant. Tech. Rep. No. 4, Gas Turbine Lab., M.I.T., Jan. 1953. (ONR Contract N5ori-7862, NR-091-158, Proj. 6888.)

2. Burke, J. C.: Turbine Blade Cooling Project Progress Report. Dept. Mech Eng., M.I.T., May 15, 1954. (ONR Contract N5-ori-07862, D.I.C. Proj. 6888.)

3. Schramm, Wilson B., Nachtigall, Alfred J., and Arne, Vernon L.: Preliminary Analysis of Effects of Air Cooling Turbine Blades on Turbojet-Engine Performance. NACA RM E50E22, 1950.

4. Arne, Vernon L., and Nachtigall, Alfred J.: Calculated Effects of Turbine Rotor-Blade Cooling-Air Flow, Altitude, and Compressor Bleed Point on Performance of a Turbojet Engine. NACA RM E51E24, 1951. 3516

5. Hubbartt, James E., Rossbach, Richard J., and Schramm, Wilson B.: Analysis of Factors Affecting Selection and Design of Air-Cooled Single-Stage Turbines for Turbojet Engines. III - Engine Design-Point Performance. NACA RM E54F16a, 1954.

6. Esgar, Jack B., and Ziemer, Robert R.: Methods for Rapid Graphical Evaluation of Cooled or Uncooled Turbojet and Turboprop Engine or Component Performance (Effects of Variable Specific Heat Included). NACA TN 3335, 1954.

7. Pinkel, Benjamin, and Karp, Irving M.: A Thermodynamic Study of the Turbine-Propeller Engine. NACA Rep. 1114, 1953. (Supersedes NACA TN 2653.)

8. Schey, Oscar W.: The Advantages of High Inlet Temperature for Gas Turbines and Effectiveness of Various Methods of Cooling the Blades. Paper No. 48-A-105, A.S.M.E., 1948.

9. Gabriel, David S., Krebs, Richard P., Wilcox, E. Clinton, and Koutz, Stanley L.: Analysis of the Turbojet Engine for Propulsion of Supersonic Fighter Airplanes. NACA RM E52F17, 1953.

10. Krebs, Richard P., and Wilcox, E. Clinton: Analysis of the Turbojet Engine for Propulsion of Supersonic Bombers. NACA RM E54A21, 1954.

11. Brown, W. Byron, and Donoughe, Patrick L.: Extension of Boundary-Layer Heat-Transfer Theory to Cooled Turbine Blades. NACA RM E50F02, 1950.

12. Hawthorne, W. R., and Walker, Antonia B.: The Effect of Blade Cooling on Stage Efficiency of a Gas Turbine. Rep. No. 6574-2, Gas Turbine Lab., M.I.T., Mar. 15, 1949. (ONR Contract N5ori-78, Task Order 21, NR-220-010, Proj. DIC 6574.)

13. Ainley, D. G., and Mathieson, G. C. R.: An Examination of the Flow and Pressure Losses in Blade Rows of Axial Flow Turbines. Rep. No. R.86, British N.G.T.E., Mar. 1951.

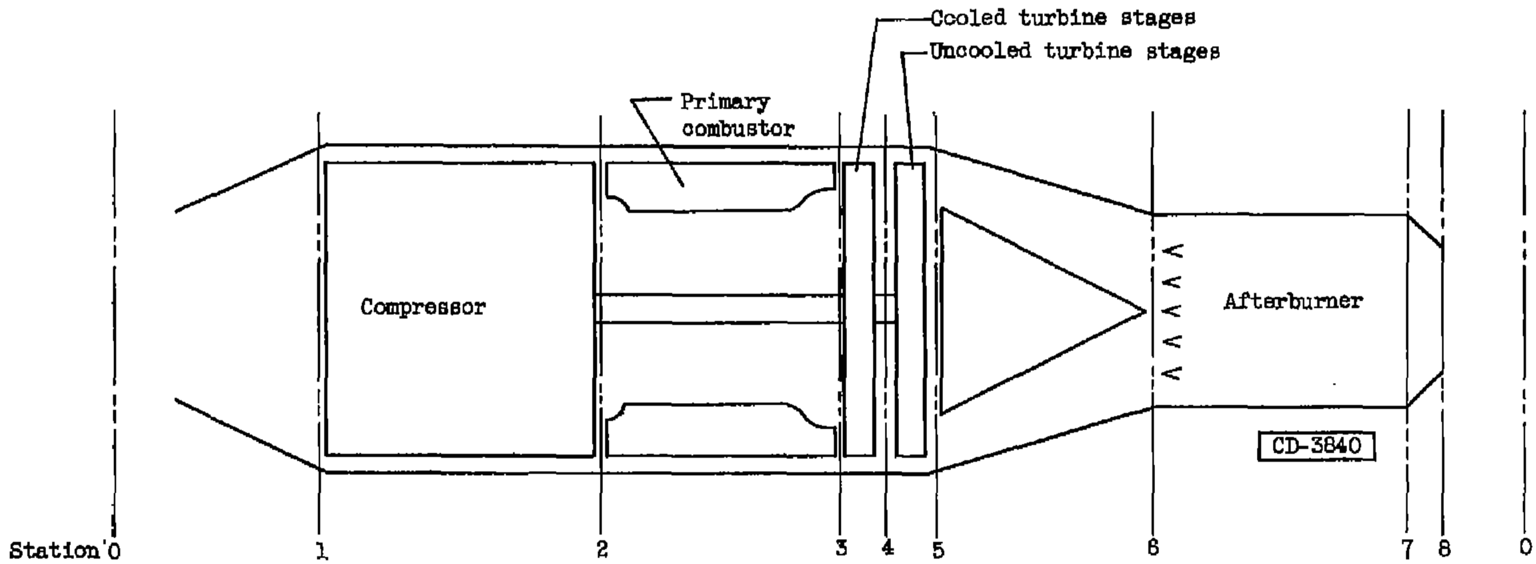


Figure 1. - Schematic sketch of engine showing calculation stations.

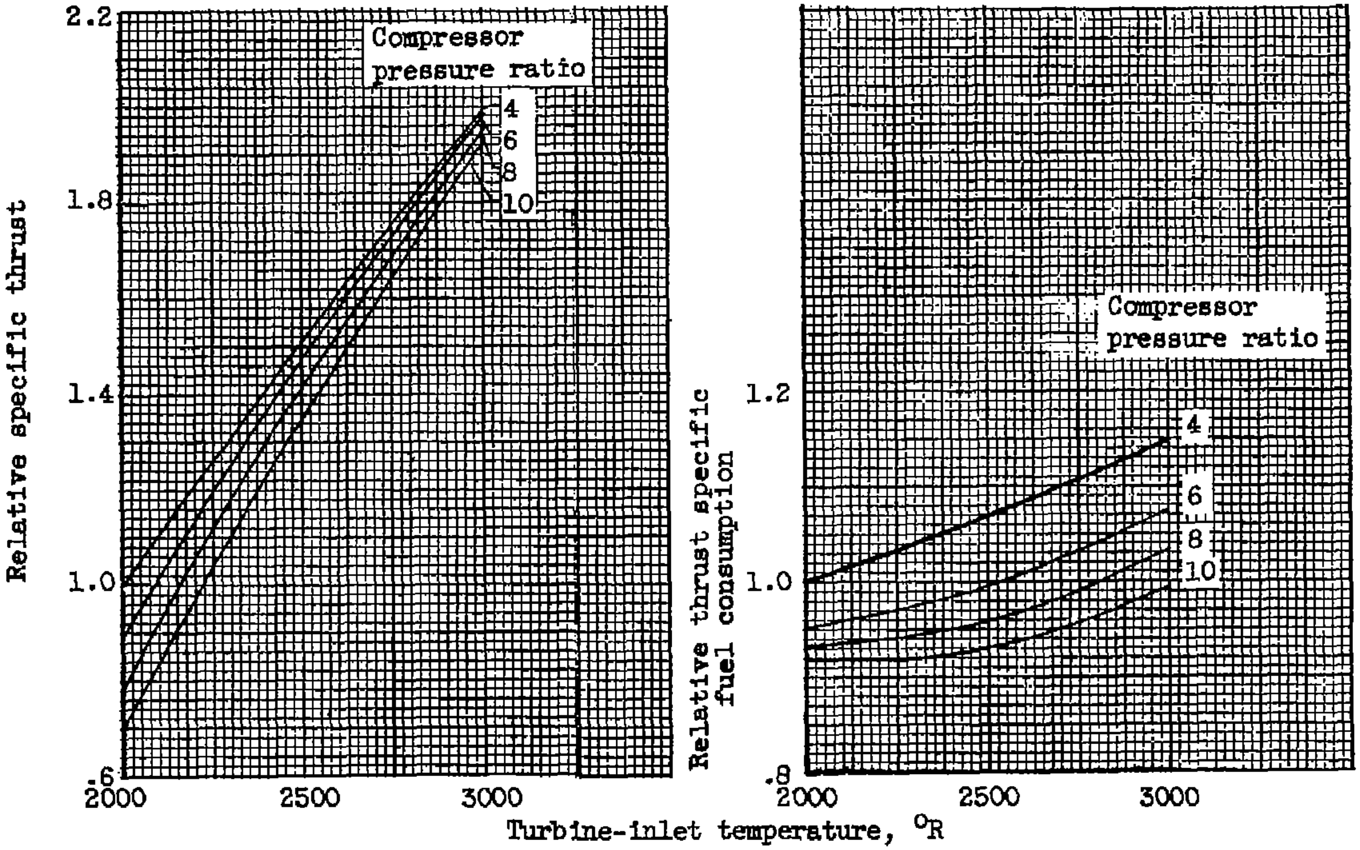


Figure 2. - Effect of turbine-inlet temperature and compressor pressure ratio on performance of nonafterburning turbojet engine with no cooling-air bleed. Flight Mach number, 2 in stratosphere.

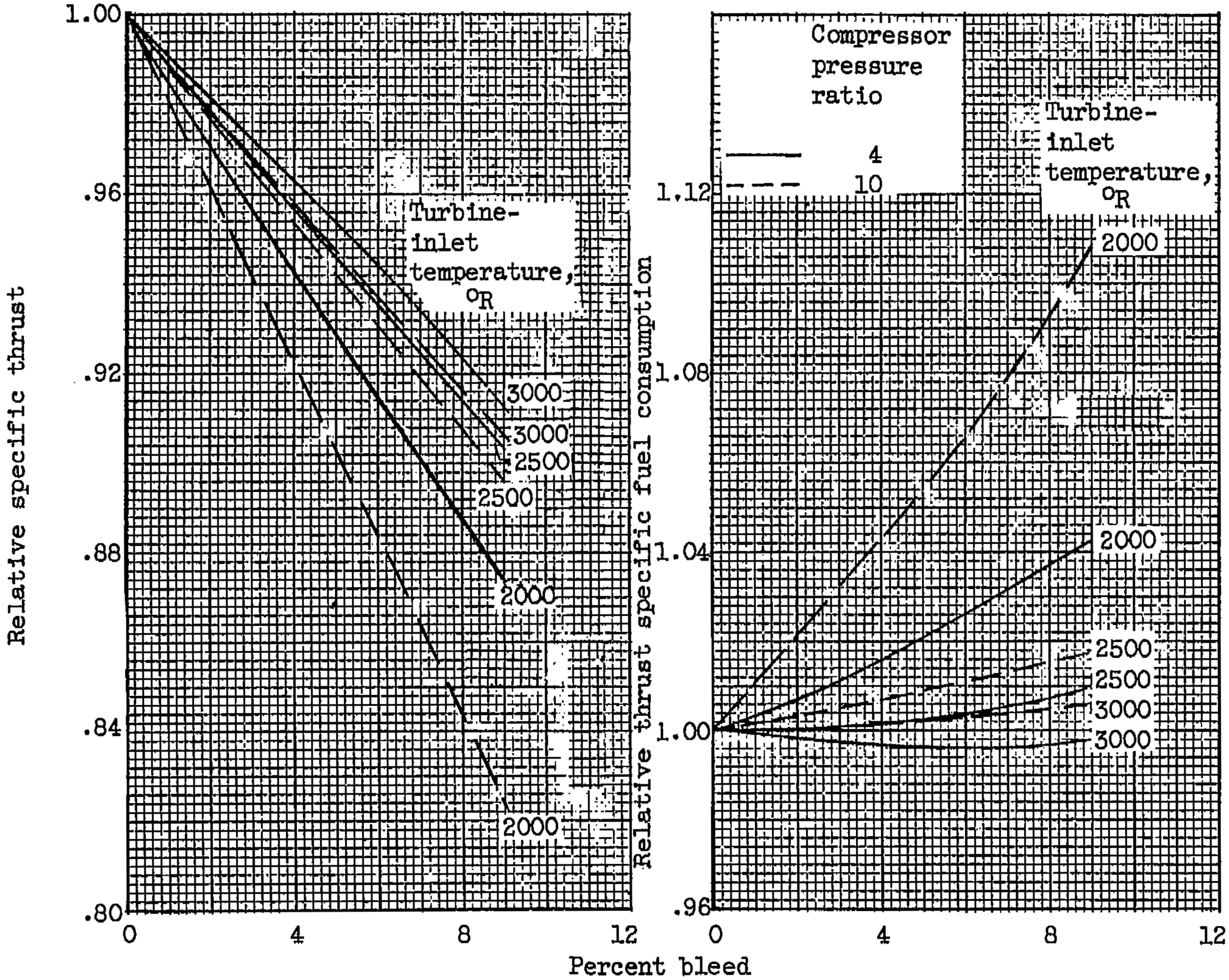


Figure 3. - Effect of turbine rotor cooling on performance of nonafterburning turbojet engine over ranges of turbine-inlet temperature and compressor pressure ratio. Flight Mach number of 2 in stratosphere.

3516

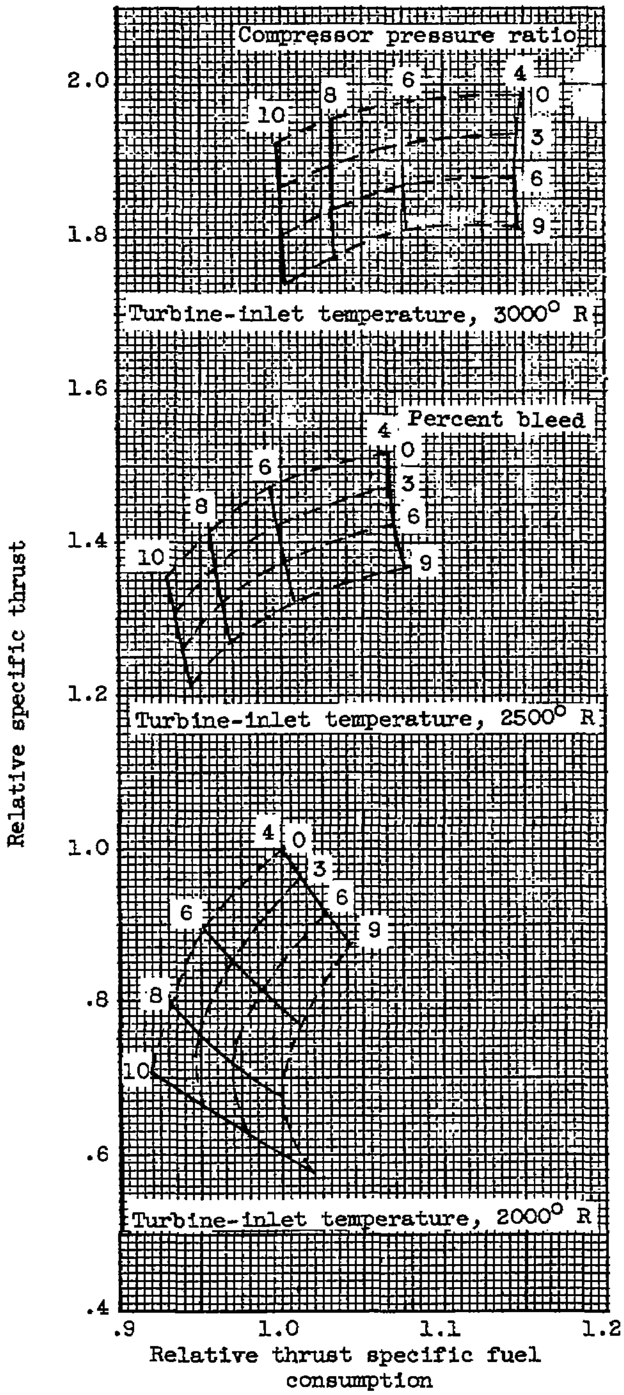


Figure 4. - Performance of nonafterburning turbojet engine over ranges of turbine-inlet temperature, compressor pressure ratio, and compressor bleed for turbine rotor cooling. Flight Mach number of 2 in stratosphere.

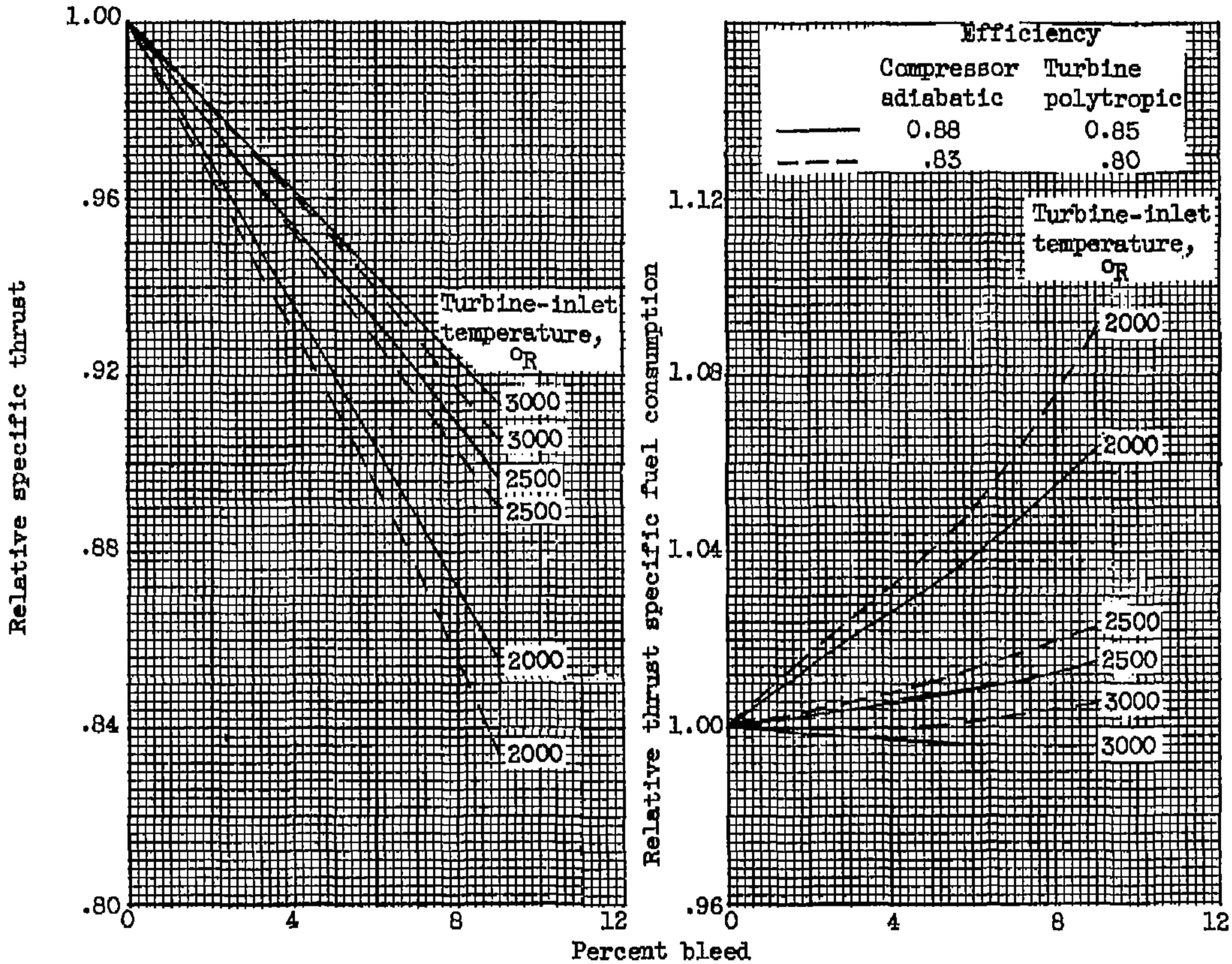


Figure 5. - Effect of compressor and turbine efficiency on variations in performance of nonafterburning turbojet engine due to turbine rotor cooling over range of turbine-inlet temperature. Compressor pressure ratio, 6; flight Mach number, 2 in stratosphere.

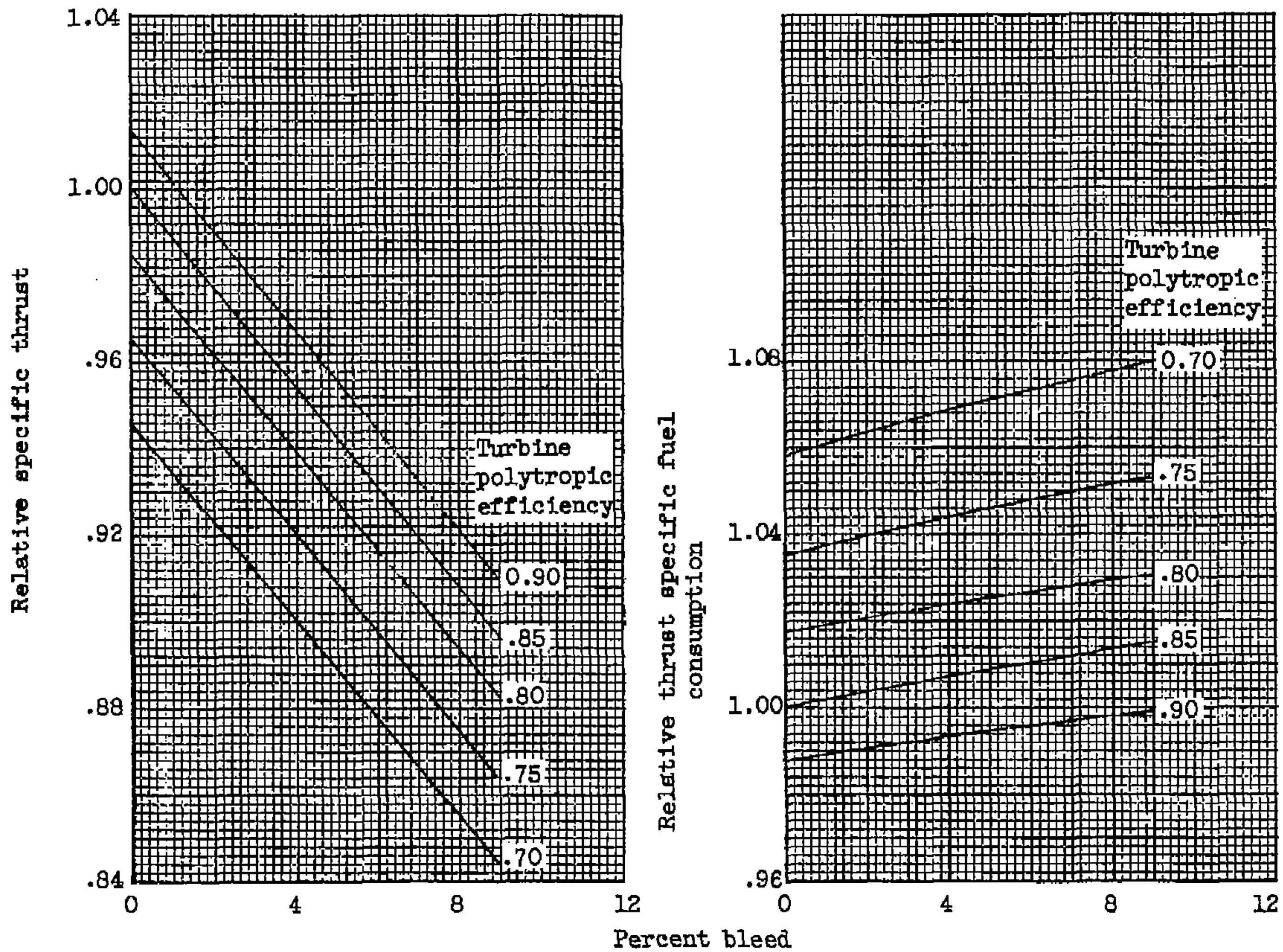
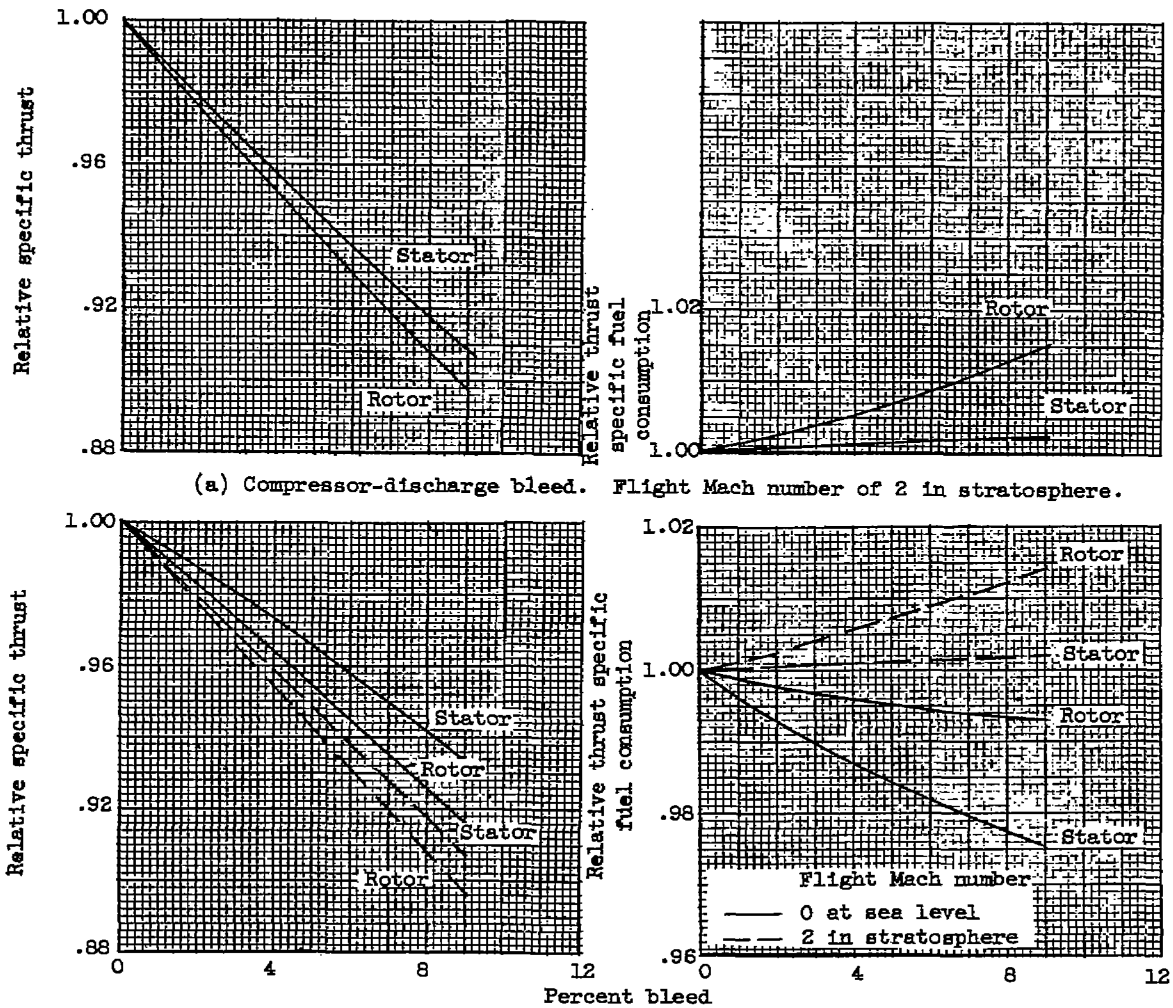
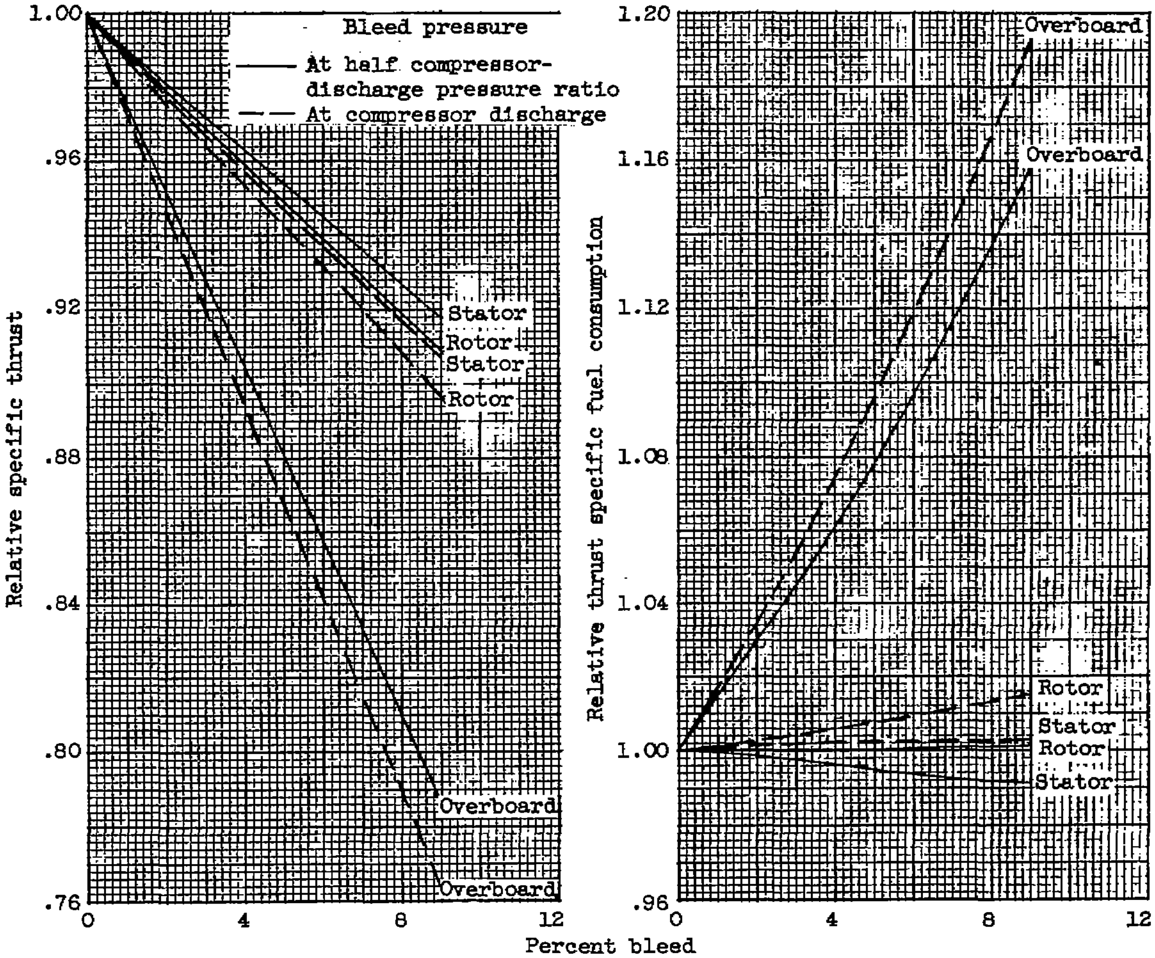


Figure 6. - Comparison of effects of turbine efficiency and compressor bleed for turbine rotor cooling on performance of nonafterburning turbojet engine. Compressor pressure ratio, 6; turbine-inlet temperature, 2500° R; flight Mach number, 2 in stratosphere.



(b) Effect of flight Mach number. Compressor-discharge bleed.

Figure 7. - Variation in performance of nonafterburning turbojet engine due to compressor bleed for turbine rotor and stator cooling. Compressor pressure ratio, 6; turbine-inlet temperature, 2500° R.



(c) Effect of cooling-air pressure losses (compressor bleed point) and overboard bleed. Flight Mach number, 2 in stratosphere.

Figure 7. - Concluded. Variation in performance of nonafterburning turbojet engine due to compressor bleed for turbine rotor and stator cooling. Compressor pressure ratio, 6; turbine-inlet temperature, 2500° R.

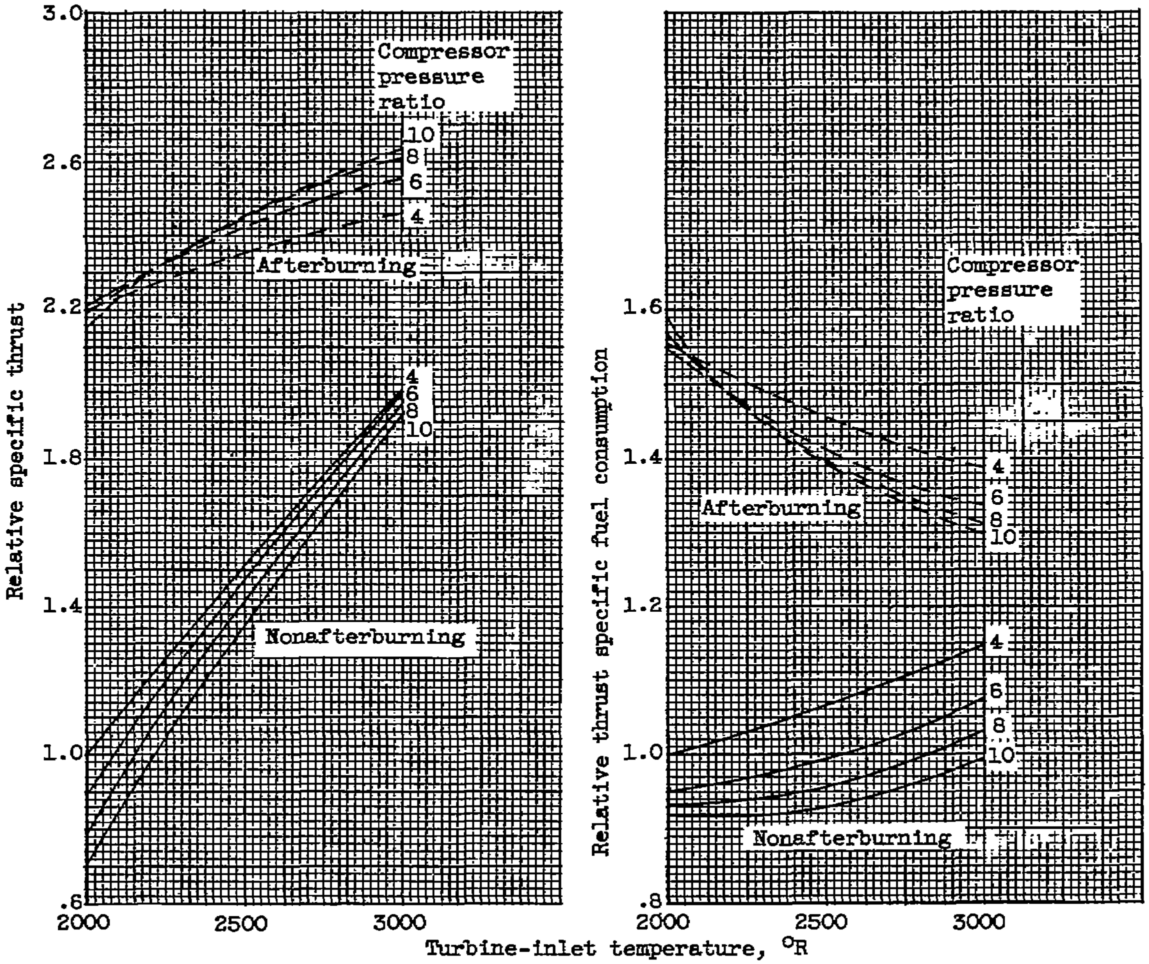


Figure 8. - Effect of turbine-inlet temperature and compressor pressure ratio on performance of afterburning and nonafterburning turbojet engines with no cooling-air bleed. Flight Mach number, 2 in stratosphere.

[redacted]

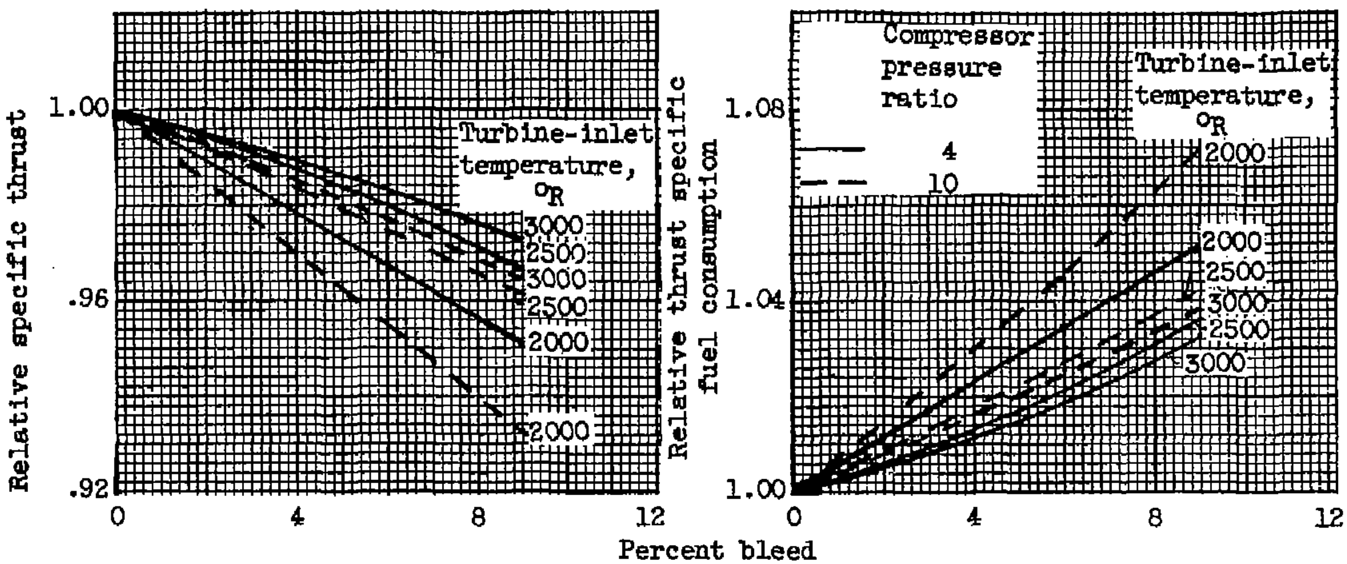


Figure 9. - Effect of turbine rotor cooling on performance of afterburning turbojet engine over ranges of turbine-inlet temperature and compressor pressure ratio. Flight Mach number, 2 in stratosphere.

[redacted]

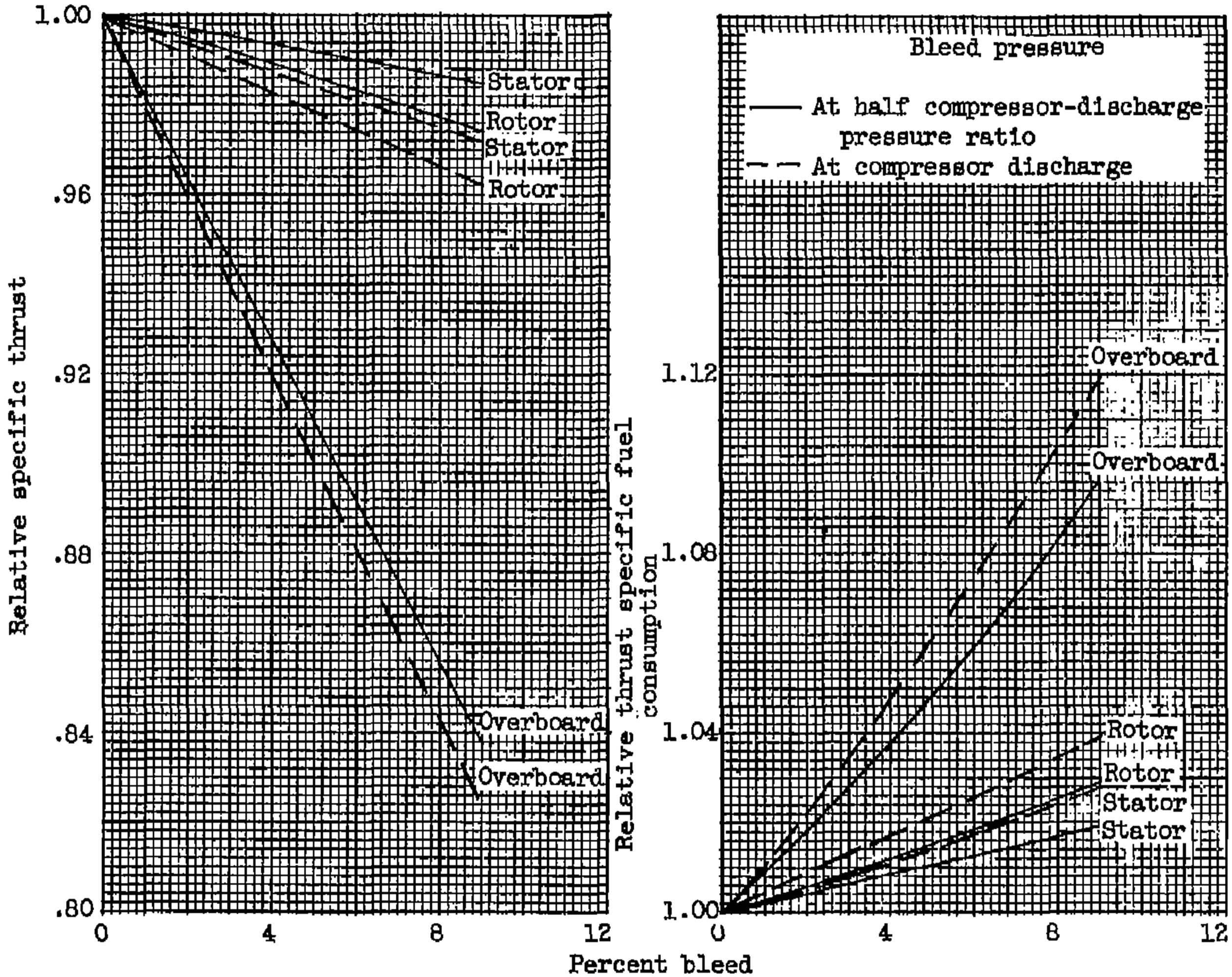


Figure 10. - Effect of cooling-air pressure losses (compressor bleed point) on variation in performance of afterburning turbojet engine due to compressor bleed for turbine rotor and stator cooling and for overboard bleed. Compressor pressure ratio, 6; turbine-inlet temperature, 2500° R; flight Mach number, 2 in stratosphere.

3516

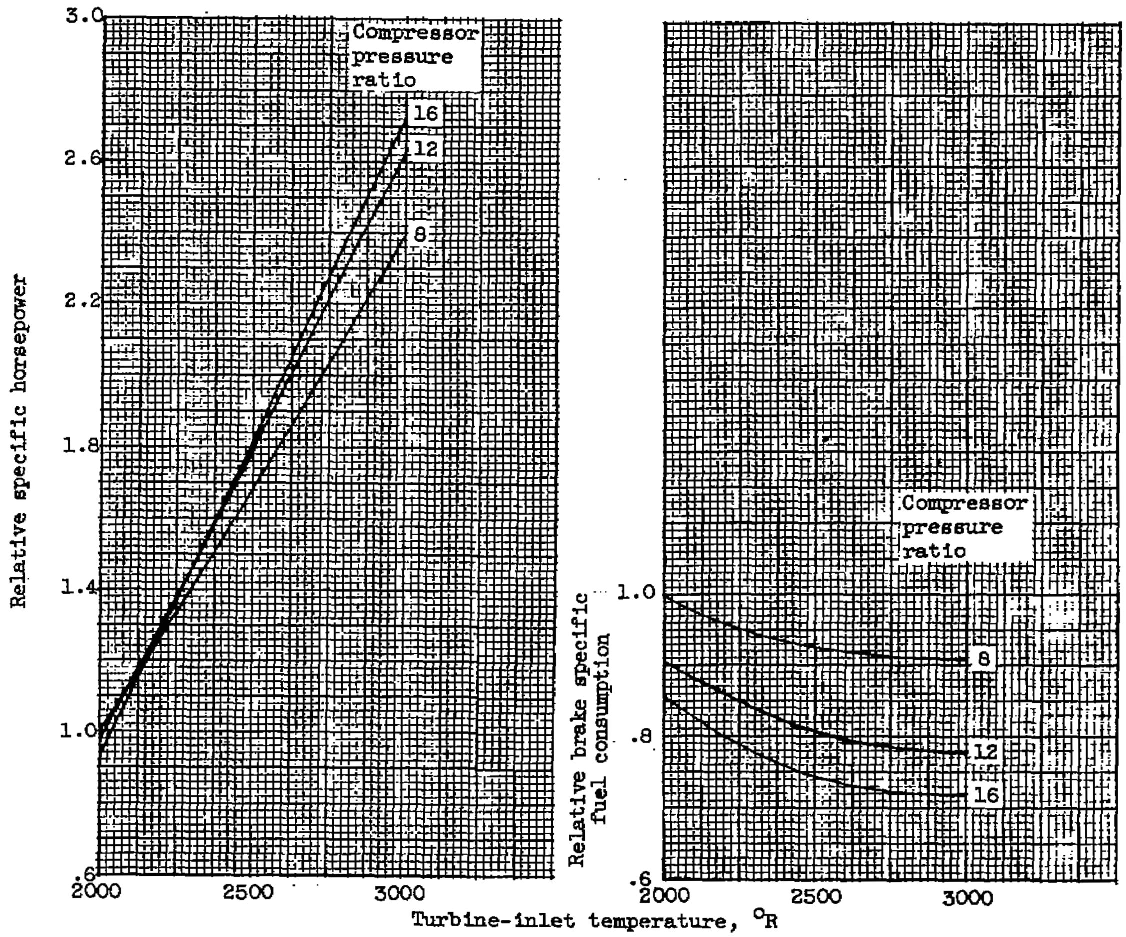


Figure 11. - Effect of turbine-inlet temperature and compressor pressure ratio on performance of turboprop engine with no cooling-air bleed. Sea-level static conditions.

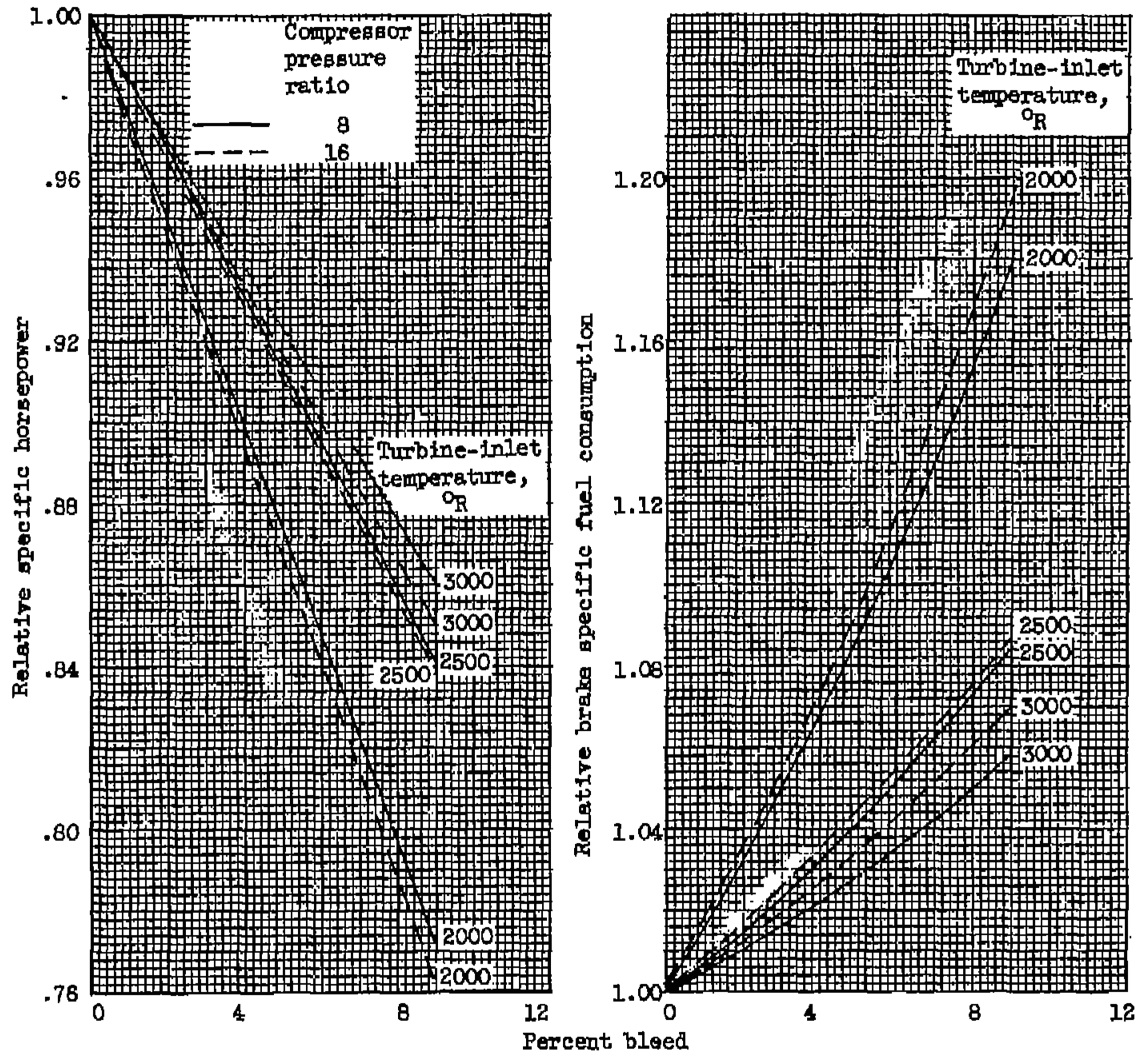


Figure 12. - Effect of turbine rotor cooling on performance of turboprop engine over ranges of turbine-inlet temperature and compressor pressure ratio. Sea-level static conditions.

3516

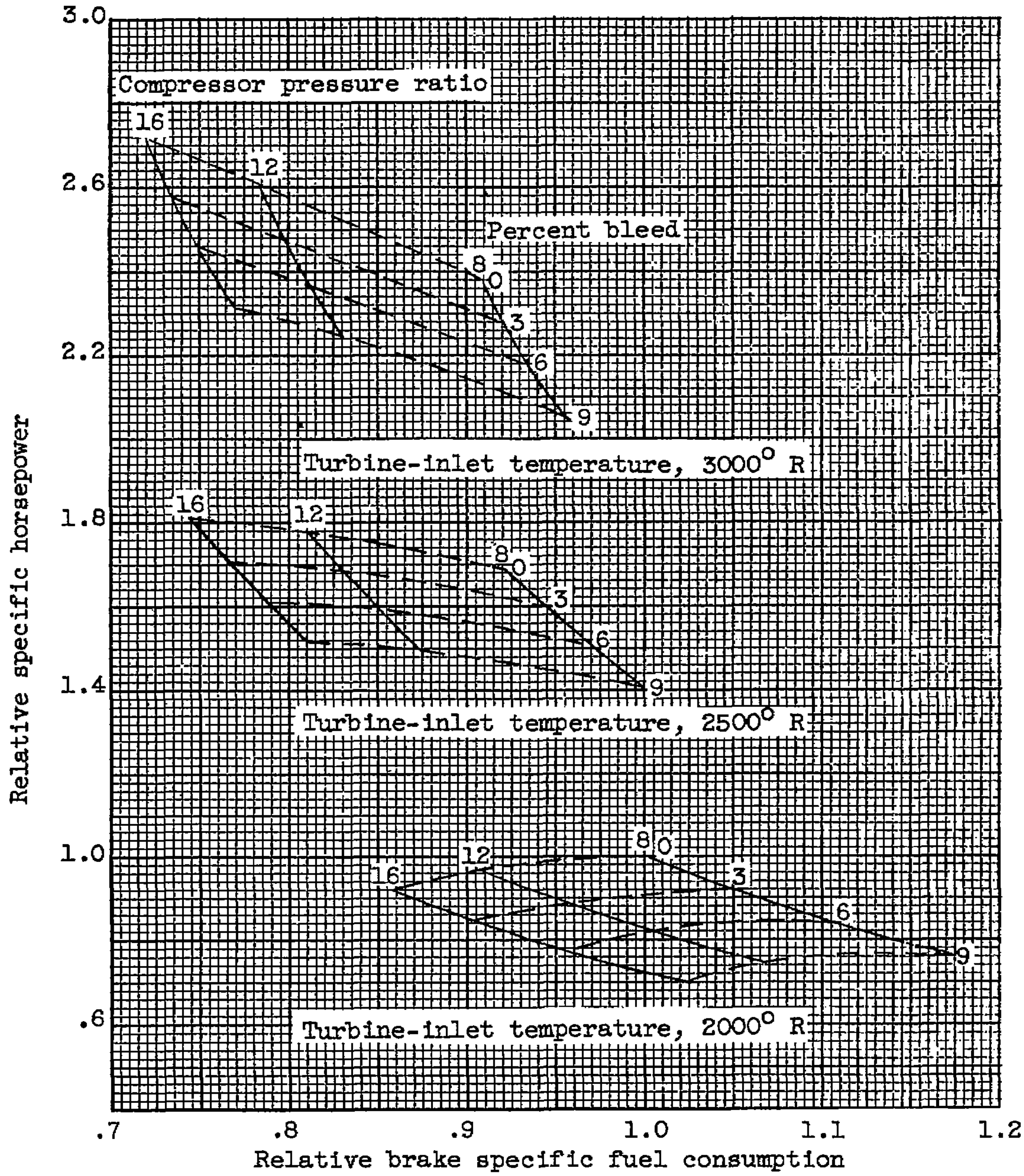


Figure 13. - Performance of turboprop engine over ranges of turbine-inlet temperature, compressor pressure ratio, and compressor bleed for turbine rotor cooling. Sea-level static conditions.

3516

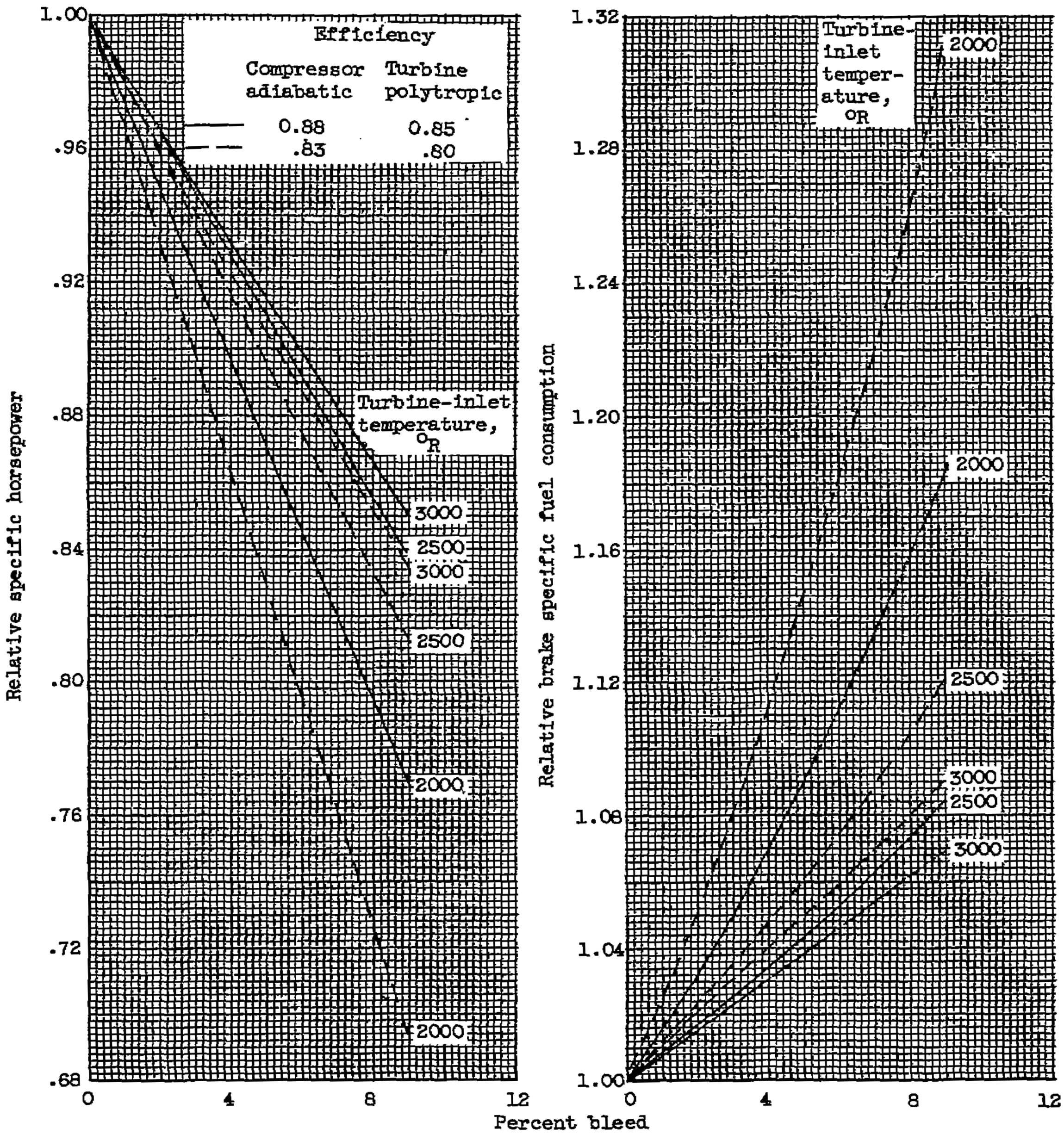


Figure 14. - Effect of compressor and turbine efficiency on variations in performance of turboprop engine due to turbine rotor cooling over range of turbine-inlet temperature. Compressor pressure ratio, 12; sea-level static conditions.

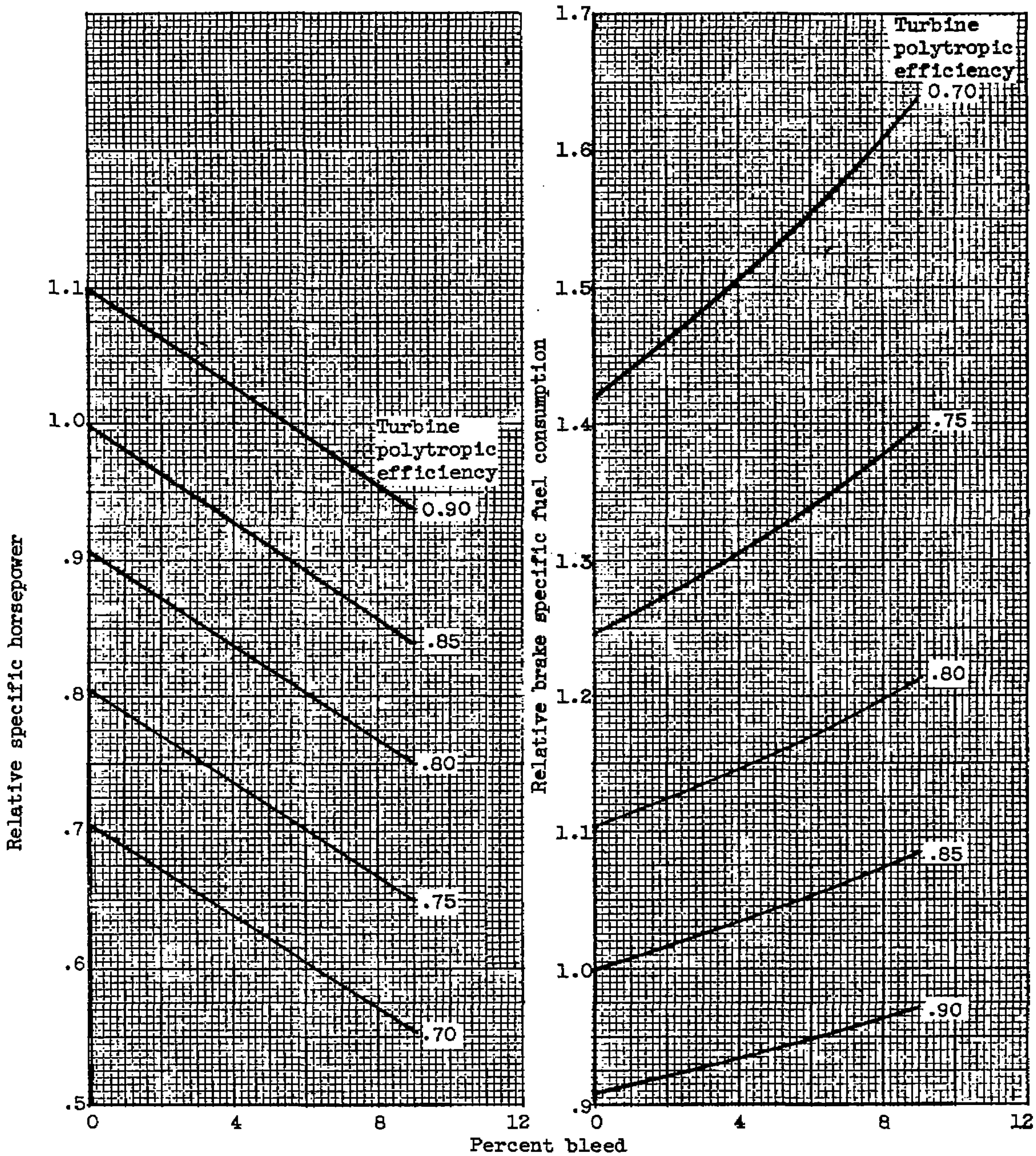
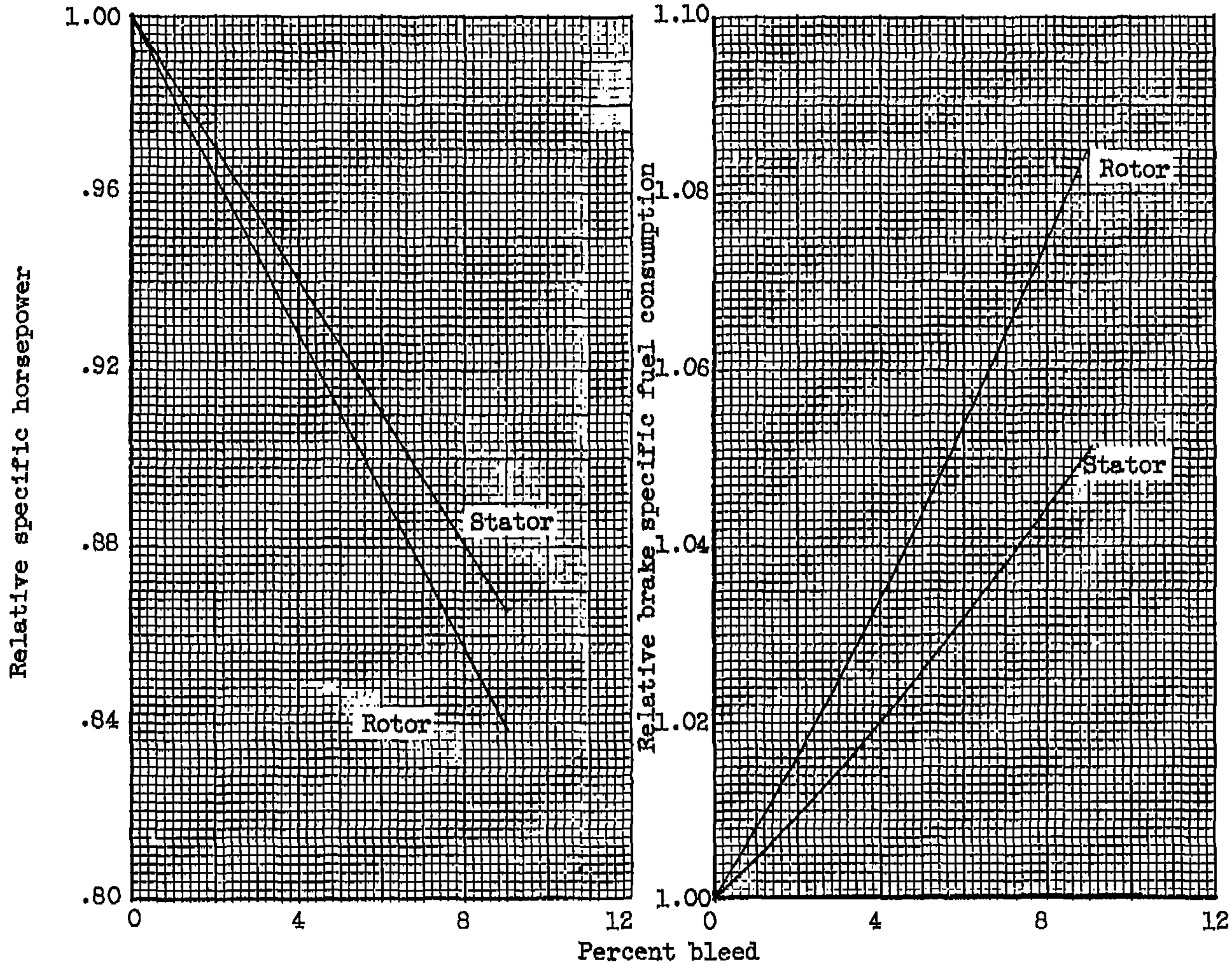
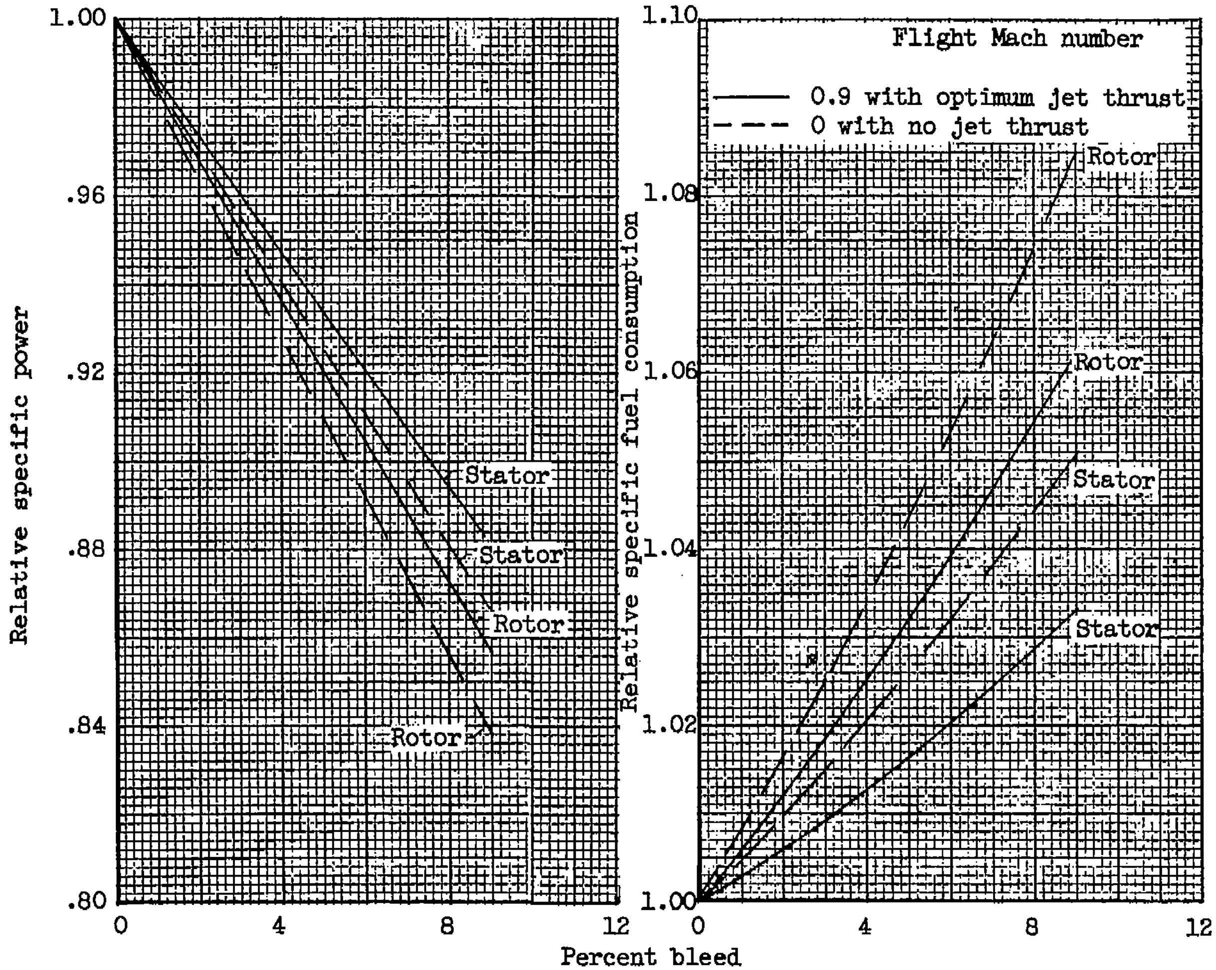


Figure 15. - Comparison of effects of turbine efficiency and compressor bleed for turbine rotor cooling on performance of turboprop engine. Compressor pressure ratio, 12; turbine-inlet temperature, 2500° R; sea-level static conditions.



(a) Compressor-discharge bleed. Sea-level static conditions.

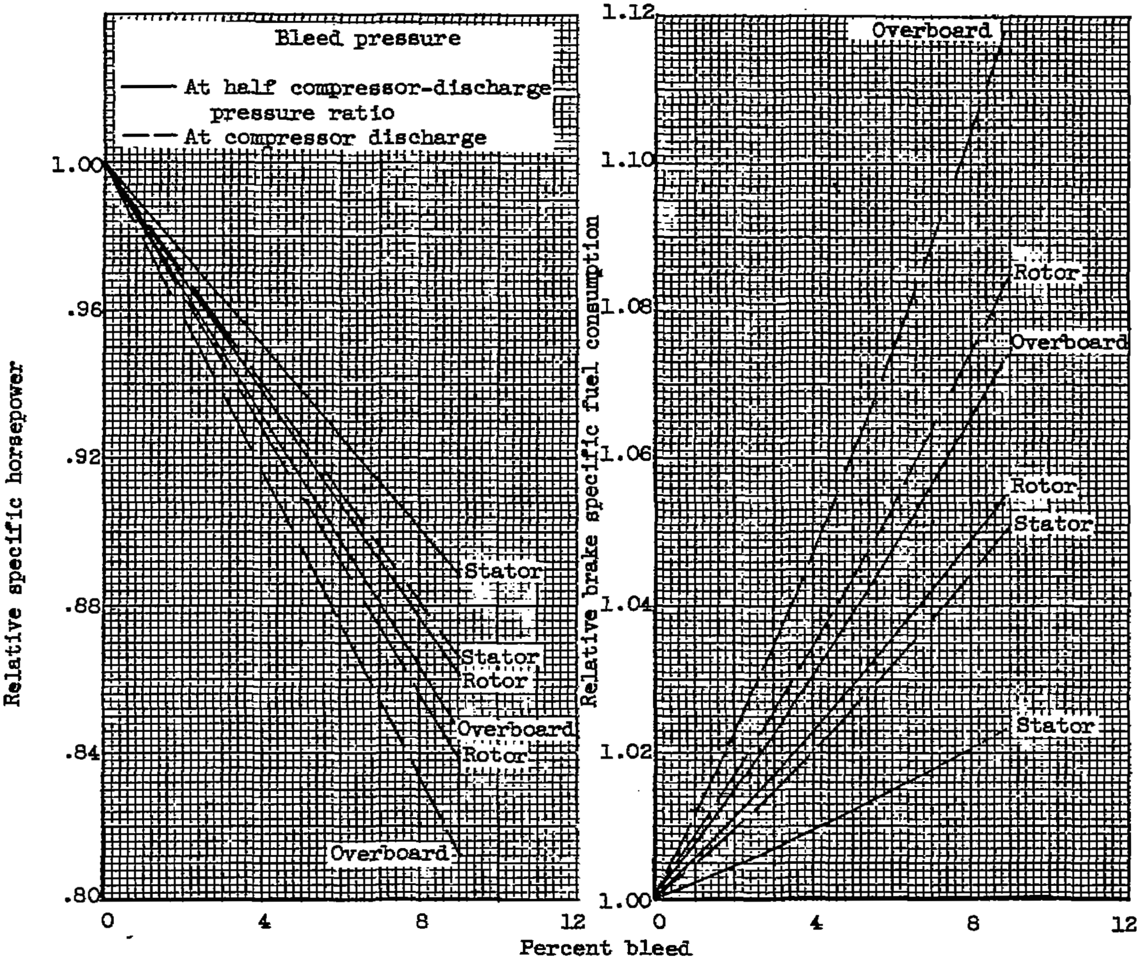
Figure 16. - Variation in performance of turboprop engine due to compressor bleed for turbine rotor and stator cooling. Compressor pressure ratio, 12; turbine-inlet temperature, 2500° R.



(b) Effect of flight Mach number. Compressor-discharge bleed.

Figure 16. - Continued. Variation in performance of turboprop engine due to compressor bleed for turbine rotor and stator cooling. Compressor pressure ratio, 12; turbine-inlet temperature, 2500° R.

3516



(c) Effect of cooling-air pressure losses (compressor bleed point) and overboard bleed. Sea-level static conditions.

Figure 16. - Concluded. Variation in performance of turboprop engine due to compressor bleed for turbine rotor and stator cooling. Compressor pressure ratio, 12; turbine-inlet temperature, 2500° R.

[redacted]

[redacted]