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SUMMARY

A method is presented for rapidly determining the approximate average midspan metal temperatures and coolant flow requirements of turbine airfoils cooled by compressor exit bleed air. Presented also are the potential reductions in airfoil metal temperatures, coolant flows, or increases in average turbine inlet gas temperatures that are possible by utilization of various airfoil cooling designs, cooling methods, and cooling systems other than the direct use of compressor exit bleed air. These additional cooling systems were direct cooling with engine fuels (Jet A or liquid methane) and the use of heat exchangers to reduce the compressor exit bleed air temperature. The cooling fluids that were used in the heat exchangers were: engine fuel, by-pass air in the fan exit duct of a turbofan engine, and ram air. The analysis was conducted for engines of aircraft cruising at Mach 3 at an altitude of 75 000 feet (22.8 km). Turbine inlet gas temperatures from 2270° to 3170° F (1515° to 2020° K) were investigated.

The results of the analysis indicate that, with compressor exit bleed air as the coolant, only moderate reductions in airfoil metal temperatures can be expected from improvements in currently known advanced convection cooled designs. This is due to the limitations imposed by the high temperature (1200° F; 920° K) of the compressor exit bleed air at the assumed engine and flight conditions. Reduction of the temperature of this cooling air by means of a heat exchanger can provide a greater potential for reducing airfoil metal temperatures or coolant flow-requirements than either improvements in the airfoil cooling design (higher thermal effectiveness) or changes in the cooling method. In addition, the results indicate that the use of the cooling systems and methods described can provide the potential for attaining turbine inlet gas temperatures as high as 3500° F (2200° K).

INTRODUCTION

The purposes of this report are to (1) provide a method for rapidly predicting metal temperatures or coolant flow rates for cooled turbine airfoils, (2) provide a basis for the comparison of the potential reductions in average airfoil midspan metal temperatures that might be expected with improvements in cooling designs, (3) define the limits in average turbine inlet gas temperature that could be attainable when compressor exit bleed air is used to cool the turbine airfoils to present day allowable metal temperatures, and (4) show the potential reductions in average airfoil midspan metal temperature that are attainable with cooling systems other than the use of compressor exit bleed air for direct cooling of the airfoils. These systems include reducing the temperature of the compressor exit bleed cooling air by the use of heat exchangers and using the engine fuel for direct cooling.

To achieve the performance required of advance aircraft, turbine inlet gas temperatures have increased markedly in recent years. Because these higher gas temperatures would result in turbine blade and vane metal temperatures that are beyond the capabilities of current materials, some means of cooling these parts must be used. The usual means is to bleed air from the compressor exit and route it through internal passages in the blades and vanes. Unfortunately, as the speeds of the aircraft increase, the temperature of this compressor exit air increases, thereby imposing a severe limitation on the potential for reducing the airfoil metal temperatures or increasing the inlet gas temperatures.

The approach often used for determining the cooling air requirements for given engine operating conditions and allowable turbine airfoil metal temperatures is to perform a detailed heat balance on each turbine airfoil row after detail designs of the airfoil cooling configurations have been made. This approach is not only time-consuming and tedious but is probably not necessary for preliminary evaluations of turbine cooling requirements. The analysis presented in this report, without resorting to detailed designs of cooled airfoils, provides a simplified and rapid method for determining the approximate requirements and limitations for cooling turbines of high speed aircraft. In many cases, this approximate method can eliminate effort that might be expended on airfoil design and refinement when actually little potential cooling improvement may exist.

The conditions for the analysis presented in this report are for an engine designed for a Mach 3 cruise aircraft operating at an altitude of 75 000 feet (22.8 km) with average turbine inlet gas temperatures (to the first stage vanes) of 2270° to 3170° F (1515° to 2020° K). Cooling of the turbine airfoils was accomplished by direct use of compressor exit bleed air or engine fuel. Heat exchangers for reducing the compressor exit bleed air temperatures were also considered. The secondary heat exchanger cooling fluids were fuel (ASTM Jet A and liquid methane), fan exit duct air (for the case of a turbofan engine), and ram air.

ANALYTICAL PROCEDURE

Rather than become concerned with the details and complexity of individual blade designs and determination of local metal temperatures, the analysis utilized an approach which was both simplified and direct. It consisted of determining the heat sink requirements for cooling each turbine row as a function of average midspan gas and metal temperatures. These heat sink requirements were then equated to the heat sink capacities available in various cooling systems when using blade and vane cooling designs having various values of thermal effectiveness. The resulting heat balances for each turbine airfoil row defined the average midspan blade metal temperatures at specific coolant flows for several average turbine inlet gas temperatures.

The following sections describe the conditions of the analysis, the method of determining the heat sink requirements to cool the turbine, the various cooling systems that were considered, and the method of determining the available heat sink capacities in these systems.

Turbine and Engine Conditions

The turbine and engine conditions selected for the analysis were obtained from reference 1. Some of the conditions which were utilized in the analysis herein and which describe the base design point of the turbine and engine at cruise conditions are tabulated:

Mach number	3
Flight altitude, ft; km	75 000; 22.8
Compressor pressure ratio	3.67
Compressor inlet total temperature, ^{o}F; ^{o}K	637; 610
Compressor discharge total temperature, ^{o}F; ^{o}K	1200; 920
Compressor inlet total pressure, psi; MN/m^2	15.2; 0.105
Compressor inlet air flow, lb/sec; kg/sec	182.6; 82.7
Ratio of compressor exit bleed air to compressor inlet flow	0.095
Average turbine inlet gas temperature, ^{o}F; ^{o}K	2270; 1515
Number of turbine stages	2

The first stage vane and blade geometries, and the average inlet and exit gas pressures and temperatures, which are found in reference 1, were sufficient information to determine directly the heat sink capacities required to cool these parts. However, the determination of heat sinks capacities for the second stage turbine first required that conditions in and out of this stage be determined from engine data in reference 1. With the compressor airflow and temperature rise known, the compressor work was equated to turbine work to determine the temperature drop across the turbine. The turbine flow used was the arithmetic average of the turbine inlet and exit flows. The turbine inlet flow was the compressor inlet flow less the 9.5 percent bleed flow that was used for cooling. This bleed flow was assumed to rejoin the gas as it flowed through the turbine so that the turbine exit flow was equal to the compressor inlet flow. An assumption of an approximately even work split between the two stages made it possible to determine the conditions for the second stage turbine vanes and blades.

Turbine Cooling Heat Sink Requirements

The determination of the heat sink requirements for cooling each row of the turbine first made use of the following equation:

$$Q_R = \bar{h}_g S_g \left(\bar{T}_g - \bar{T}_m\right) \tag{1}$$

(All symbols are defined in appendix A.) However, to present the data using a more meaningful reference metal temperature than the average for the entire airfoil $\bar{T}_m$, the average temperature at the midspan of the airfoil T_m was used. This location was selected because it generally is the critical region of cooled airfoils. Along with this change in the reference metal temperature, the average spanwise gas temperature distribution from reference 1 was used and an assumption was made that the spanwise gas and airfoil metal temperature profiles are such that the difference between the average midspan gas and airfoil metal temperatures $(T_g - T_m)$ is equal to the difference between the average gas and airfoil metal temperatures $(\bar{T}_g - \bar{T}_m)$. This assumption then transforms equation (1) into

$$Q_R = \bar{h}_g S_g (T_g - T_m) \tag{2}$$

For the base case of an average turbine inlet gas temperature $\bar{T}_g$ of 2270° F (1515° K) the average midspan gas temperatures T_g relative to each of the four airfoil rows were:

First stage vane, °F; °K	2350; 1555
First stage blade, °F; °K	2080; 1400
Second stage vane, °F; °K	2000; 1360
Second stage blade, °F; °K	1800; 1250

These temperatures are a constant 80° F (44° K) higher than the average inlet gas temperature relative to each of the airfoil rows. This assumed difference between T_g and $\bar{T}_g$ was based on data in reference 1.

Because the average midspan gas temperature was used in the determination of the heat sink requirements of the stator vanes, the analysis does not account for local hot spots in the gas that may exist circumferentially ahead of the vanes. However, the approximate coolant flow that would be required to account for an assumed hot spot (but would overcool the vanes at other circumferential locations) can be obtained by multiplying the vane coolant flow requirements as obtained in this report by the ratio $(T_{g,M} - T_m)/(T_g - T_m)$. This factor is the ratio of the difference between the hot spot midspan gas temperature and the desired midspan metal temperature to the temperature difference in equation (2).

In order to determine the heat sink requirements for cooling the turbine at higher turbine inlet gas temperatures, the relative gas temperatures tabulated above are increased directly by the amount of the assumed increase in average turbine inlet gas temperature. This simplifying assumption implies that the turbine work and turbine size remain the same. Also, the effects of changes in gas properties and the effects of changes in cooling bleed requirements about the base engine condition are neglected. As a result the gas temperature drop across the turbine remains constant and the higher exit gas temperature and pressure are utilized to increase engine thrust.

The gas-to-blade heat transfer coefficients $\bar{h}_g$ for each vane and blade row was determined by using average state conditions across each row and the following equation from reference 2:

$$\bar{h}_g = 0.037\ \mathrm{Re}^{0.8} \mathrm{Pr}_g^{1/3} \frac{k_g}{x} \tag{3}$$

where

$$\mathrm{Re} = \frac{(\rho_g V_g)x}{\mu_g}$$

The term x is the average surface distance from the leading edge to the trailing edge. The term $(\rho_g V_g)$ is determined from the continuity equation:

$$\rho_g V_g = P_g \sqrt{\frac{\gamma g}{\bar{R} T_g}} \left[\frac{M_g}{\left(1 + \frac{\gamma - 1}{2} M_g^2\right)^{(\gamma+1)/2(\gamma-1)}} \right] \tag{4}$$

The term M_g is the only unknown on the right side of this equation and is determined from the average relative total to static gas pressure ratio across each row. The average gas-to-blade heat transfer coefficient $\bar{h}_g$ can then be determined from equation (3).

The heat sink requirement to cool each row to a desired or allowable metal temperature can then be determined by the use of equation (2). The heat sink requirement to cool the airfoils of the entire turbine is simply the sum of the individual row requirements.

Turbine Cooling Systems

The five systems considered for cooling the turbine are shown schematically in figure 1. These systems either used the compressor exit air (fig. 1(a)), or engine fuel (ASTM Jet A or liquid methane (LCH_4), fig. 1(b)), to directly cool the turbine, or made use of compressor exit air after its temperature had been reduced. The temperature reduction was accomplished by means of heat exchangers which utilized the heat sink capacities of a secondary coolant. The secondary coolants considered were engine fuel (fig. 1(c)), fan exit air in the case of a turbofan engine (fig. 1(d)), and engine ram air (fig. 1(e)).

The primary method for cooling the turbine airfoils was convection cooling. Other methods of cooling, such as film and transpiration, however, were considered.

Heat Sink Capacities Available

The following sections describe the methods used to determine the available heat sink capacities for each of the cooling systems considered.

Compressor exit air. - The heat sink capacity in the compressor exit bleed air available for cooling an individual turbine airfoil row was obtained from the equation

$$Q_A = W_c C_{p_c} (T_{co} - T_{ci}) \tag{5}$$

For the purposes of this analysis this equation was modified to include the following: (1) the thermal effectiveness of an airfoil configuration,

$$\eta_t = \frac{(T_{co} - T_{ci})}{\left(\bar{T}_m - T_{ci}\right)} \tag{6}$$

(2) the effect of the difference between the average metal temperature at the midspan

of the airfoil and the average for the entire airfoil (which was assumed to be 80° F (44° K) to be compatible with the difference between the average and midspan gas temperatures in ref. 1), and (3) the ratio of the compressor exit bleed flow used for cooling to the flow entering the compressor W_c/W_E. With these modifications incorporated, equation (5) becomes

$$Q_A = \frac{W_c}{W_E} W_E C_{p_c} \eta_t \left[T_m - (T_{ci} + 80^\circ \text{ F})\right] \tag{7}$$

For the base conditions of the engine used in this analysis, the compressor inlet air flow was 182.6 pounds per second (82.7 kg/sec), as noted previously. Although some heating of the compressor exit air would be expected to occur during its passage to the airfoil bases, for the purposes of illustrating the maximum heat sink potential in the compressor bleed air, an assumption was made that the air was so ducted that the relative supply temperature of the air to the airfoils was equal to the compressor discharge temperature of 1200° F (920° K). The coolant flow ratios considered ranged from 0 to as high as 0.08 for individual airfoil rows. Thermal effectiveness values of 0.5, 0.75, and 1.0 were considered. The first value represents an average effectiveness that is attainable with currently available convection-cooled airfoils, the second value represents advanced designs, and the last figure represents the highest value attainable with a convection-cooled design. Appendix B illustrates the validity of these values by comparing them with like values determined for the blade designs in references 3 to 5.

Thus for a given compressor inlet flow W_E, and with the cooling air temperature to the airfoils assumed to be equal to the compressor exit temperature, the available heat sink capacities can be evaluated in terms of the coolant flow ratio, blade thermal effectiveness, and average midspan metal temperature. Equating the heat sink capacities available (eq. (7)) to the heat sink capacities required (eq. (2)), the airfoil metal temperatures for each turbine row can be determined for the inlet gas temperature relative to the row, and the coolant flow ratio and airfoil thermal effectiveness of the row.

Because additional heat sink capacity is available in the cooling air when it is used as a buffer layer in the form of a film or a transpired layer on the outside surface of the vanes or blades, consideration was given to these cooling methods. The simplified analyses used herein could not readily account for the effects of the large range of film cooling configurations that can be utilized. The benefits that could be expected with film cooling would, however, range from that attainable with an ineffective convection-cooled configuration to the cooling potential of a transpiration-cooled airfoil.

The metal temperatures attainable with a transpiration-cooled airfoil are more readily determined. The following two equations obtained from reference 3 were used:

$$\frac{d}{dx}\left(w_{bl} C_{p_{bl}} t_{aw}\right) = C_{p_c} t_{co} \frac{dw_c}{dx} + C_{p_g} t_g \frac{dw_g}{dx} \tag{8}$$

$$h_g(t_{aw} - t_m) = C_{p_c} \frac{dw_c}{dx}(t_{co} - T_{ci}) \tag{9}$$

These equations apply to an increment of the airfoil in the direction of gas flow and for a unit length of its span. Equation (8) is an enthalpy balance in the boundary layer on the external surface of the airfoil, and equation (9) is a heat balance through the porous wall. The same property values were used for the air and gas, and the values were assumed constant in the equations as in reference 3. On combining equations (8) and (9), and by assuming a cooling air flow rate in the boundary layer of w_c proportional to $x^{0.8}$, such as to provide a uniform wall temperature, reference 3 shows that

$$\frac{\overline{T}_m - T_{ci}}{\overline{T}_g - T_{ci}} = \frac{1 - 0.14\,\psi}{1 + \psi} \tag{10}$$

where

$$\psi = \frac{C_p \dfrac{dw_c}{dx}}{h_g} \tag{11}$$

Assuming turbulent gas flow, and using gas conditions and airfoil dimensions for the first stage airfoils, equation (11) becomes

$$\psi = 48.5\left(\frac{W_c}{W_E}\right) \tag{12}$$

On replacing the average gas and airfoil metal temperature with the average mid-span values to make them compatible with the assumptions made earlier, equation (10) becomes

$$\frac{T_m - (T_{ci} + 80^{\circ}\ F)}{T_g - (T_{ci} + 80^{\circ}\ F)} = \frac{1 - 0.14\,\psi}{1 + \psi} \tag{13}$$

Engine fuels. - A potential heat sink for direct cooling of the turbine is the engine fuel. The application of liquid methane fuel to high-speed aircraft has been suggested and benefits of its higher heat of combustion are described and evaluated in references 6 and 7. This fuel also provides the engine and aircraft with a potentially useful coolant because of its low temperature state in liquid form, and because of its large heat sink capacity prior to being burned. Currently used ASTM Jet A fuel, although not possessing a heat sink capacity as large as liquid methane, nevertheless, is a potentially useful coolant.

Direct cooling of rotating blades, however, may not be practical because of the large number of problems that may be present. Some of these are fuel leakage through seals between stationary and rotating parts, and centrifuging of decomposition products to the tips of the blades where removal would be difficult. Problems may also exist in the detail designs of the blades or vanes to maintain metal temperatures hundreds of degrees higher than that of the fuel being used for cooling. These problems would require considerable study and are beyond the scope of this investigation.

The simplified analysis used herein to illustrate the potentials for reducing metal temperatures or for increasing turbine inlet gas temperature with direct fuel cooling made use of a balance between the heat sink capacity available in the fuel and the heat sink capacity required to cool the turbine. The conditions assumed and the procedure used for determining the heat sink capacities in Jet A and LCH_4 fuel are described in the following paragraphs.

The fuels were assumed to be supplied from the aircraft tanks at 5 psia (0.0344 MN/m^2) and 100° F (311° K) for Jet A, and 5 psia and -280° F (100° K) for methane. The pressures of both fuels were assumed to be increased by engine-driven pumps such that after cooling the turbine the fuel pressures were 300 psia (2.07 MN/m^2). The temperatures reached after cooling the turbine were assumed to be either 700° F (645° K) or 1000° F (810° K) for Jet A, and 1200° F (920° K) for methane. Jet A cannot be normally heated to temperatures in the range of 700° to 1000° F (645° to 810° K) without encountering fouling, but the results of some recent unpublished experiments at NASA-Lewis indicate that some blends of Jet A with minimum oxygen content may be safely heated to at least 700° F (645° K). Furthermore, experimental tests on a highly paraffinic jet fuel, F-71, reported in reference 8 indicate that this fuel maintained satisfactory thermal stability without excessive deposits at temperatures of 1000° F (810° K). Based on the data on cracking rates of fuels in reference 6, no problems with the thermal stability or with excessive deposits are expected when methane is heated to 1200° F (920° K).

The quantity of fuel available for use in cooling the turbine was assumed to be the fuel flow of the primary burner. The fuel-air ratio for the burner was determined from the following heat balance across the burner,

$$f\eta_B H_f = (1 + f)h_2 - h_1 - fh_f \tag{14}$$

The heat content of the fuel before burning h_f was determined from the end state properties of the fuel after it performed its cooling function. Enthalpy data were obtained from reference 9 for a fuel with a 45° API and a characteristic factor of 12.5 to represent Jet A. Enthalpy data for methane were obtained from reference 10. The heat of combustion H_f, and the other terms of equation (14), were determined directly or by the extrapolation of the data of reference 11 for fuels with hydrogen carbon ratios of 0.17 and 0.33 for Jet A and CH_4, respectively, and for turbine inlet temperatures of 2270° F (1515° K) to 3170° F (2020° K). The net available heat sink capacity in each fuel was determined by the use of the following equation:

$$Q_A = (W_E f h_f) - 130 \tag{15}$$

The term Q_A represents the total heat sink capacity in the fuel if it is heated to the assumed temperatures in the process of cooling the airfoils. The constant, 130 Btu per second (0.137 MW) is that portion of the total heat sink capacity of the fuel that was assumed to be required for aircraft and engine cooling other than for turbine cooling.

Although the use of afterburner fuel was not considered in this analysis, the effect of its use can easily be determined. The fuel heat sink capacity based on the primary burner fuel flow would simply be increased by the ratio of the total fuel flow (including that of the afterburner) to that of the primary burner.

Reduced compressor bleed air temperatures. - The engine fuel, ram air, or fan exit air are potential heat sinks that can be added to the heat sink capacity already available in compressor exit bleed air. This increased capacity can be obtained by using these secondary fluids in heat exchangers to reduce the temperature of the compressor bleed air before it is used to cool the turbine. These heat exchangers will hereinafter be referred to as the fuel, ram air, or fan exit air heat exchangers.

A heat balance across a heat exchanger gives,

$$\Delta T_c = \frac{Q_o}{\left(\frac{W_c}{W_E}\right) W_E C_{p_c}} \tag{16}$$

This equation indicates that the temperature reduction of the compressor air ΔT_c is directly proportional to the heat sink capacity in the secondary cooling fluid of the heat exchanger Q_o and inversely proportional to the ratio of compressor bleed flow to compressor inlet flow. When equation (16) is substituted into a form of equation (7), which defines the available heat sink capacity in the compressor exit bleed air, the following can be obtained:

$$\Delta Q_A = \eta_t Q_o \tag{17}$$

This equation indicates that ΔQ_A, the additional heat sink capacity available from the secondary fluids for cooling the turbine, is proportional to the airfoil thermal effectiveness and the heat sink capacity in the secondary cooling fluid of the heat exchanger. The total heat sink capacity available to cool the turbine, therefore, is the sum of that available in the compressor bleed air at its exit temperature (eq. (7)), and that made available by the secondary fluid in the heat exchanger (eq. (17)).

The following sections describe the methods and assumptions used in determining the heat sink capacities in the secondary fluids; $Q_{o,f}$ (fuel), $Q_{o,d}$ (fan exit air), $Q_{o,r}$ (ram air), and the additional available heat sink capacities ΔQ_A provided by these fluids.

Fuel heat exchanger: The total heat sink capacity in the fuel was determined from the equation

$$Q_{o,f} = W_E \left[1 - \left(\frac{W_c}{W_E}\right)\right] f h_f - 130 \tag{18}$$

The terms used in this equation are determined in the same manner as described in the prior section Engine fuels. The enthalpy of the fuel h_f was based on the same fuel supply condition as in the prior section and with the same final temperatures of the Jet A fuel after being used as the coolant, that is, 700° and 1000° F (645° and 810° K). The final temperature of the methane was assumed to be 1000° F (810° K) instead of the 1200° F (920° K) used in the prior section. The last term in equation (18), as previously stated, represents, in Btu/sec, the heat sink capacity of the fuel that is used for aircraft and engine cooling other than that for turbine cooling.

The heat sink capacity in the fuel $Q_{o,f}$ is then used in equation (17) to obtain the additional available heat sink capacity for cooling the turbine $\Delta Q_{A,f}$. This additional capacity was determined for an arbitrarily assumed cooling air-to-engine flow ratio of 0.10 for airfoil thermal effectiveness values of 0.5 and 0.75. This additional capacity was added to the capacity of the compressor exit air to determine a total heat sink capacity available. This total heat sink capacity was then balanced against that required for various airfoil midspan metal and turbine inlet gas temperatures to determine allowable operating limits.

Fan exit air heat exchanger: Another means of providing additional heat sink capacity for turbine cooling is to use a heat exchanger in the fan exit duct of a turbofan engine to reduce the compressor bleed air temperature before this bleed air is used to cool the turbine. For the purposes of this analysis and to illustrate the benefits of this cooling method, an assumption was made that engine conditions used previously could be applied to a turbofan. The overall pressure ratio and exit temperature of the fan and compressor were also assumed to be the same as for the base engine; 3.67 and 1200° F (920° K), respectively. It was also assumed that the gas temperature and pressure drop through the first two turbine stages of a turbofan engine would be the same as for the base engine and that additional turbine stages would be added to supply the energy required to drive the by-pass fan. Cooling of only the first two stages was considered. In addition to the engine conditions previously assumed, the following conditions which are representative of a turbofan engine for supersonic flight were used:

By-pass duct-to-compressor inlet flow ratio, (W_d/W_E)	1.30
Fan pressure ratio	1.50
Fan efficiency	0.90

The fan exit air temperature (fan duct heat exchanger inlet temperature) was determined to be 785° F (693° K) from the known flight and altitude conditions, fan pressure ratio, and fan efficiency.

The heat sink capacity in this air was calculated from

$$Q_{o,d} = \left(\frac{W_d}{W_E}\right) W_E C_{p_c} \Delta T_d \tag{19}$$

Values of temperature rise in the duct air ΔT_d of 10°, 20°, and 30° F (6°, 11°, and 17° K) were assumed. The portion of the heat sink capacity of this secondary cooling fluid that was available for cooling the turbine $\Delta Q_{A,d}$ was determined using equation (17) with values of η_t of 0.5 and 0.75. This additional heat sink capacity was added to the capacity available from a fixed ratio of compressor bleed-to-total compressor air flow of 0.10. The total available heat sink capacity was then balanced against that required for various airfoil midspan metal temperatures and turbine inlet gas temperatures to determine allowable operating limits.

Ram air heat exchanger: The temperature of the ram air at the assumed flight conditions is 637° F (610° K) compared to a compressor exit bleed air temperature of 1200° F (920° K). This temperature difference and the rate at which this air would pass through a heat exchanger can be used to reduce the temperature of the compressor exit air. For the purpose of illustrating the maximum reductions in turbine airfoil metal temperatures attainable using this system, the ram air temperature at the heat exchanger exit was assumed to be a relatively high value of 1000° F (810° K). However, it was assumed this temperature would not create a critical environment for such items as the engine cowl or the controls and lines surrounding the engine.

The heat sink capacity in the ram air was obtained from the following equation as a function of the temperature rise across the ram air side of the heat exchanger and the ratio of ram-to-compressor air flow

$$Q_{o,r} = \frac{W_r}{W_E} W_E C_{p_r} \Delta T_r \tag{20}$$

The portion of this heat sink capacity that is available for cooling the turbine $\Delta Q_{A,r}$ was determined from equation (17) for appropriate values of thermal effectiveness. The available heat sink capacity in ram air for ratios of W_r/W_E of 0.05, 0.10, and 0.15 and for airfoils with effectiveness of 0.5 and 0.75 was added to that in 10 percent of the compressor exit air to provide the total heat sink capacity available for turbine cooling. The total available heat sink capacity was then balanced against that required for various airfoil midspan metal temperatures and turbine inlet gas temperatures to determine allowable operating limits.

RESULTS AND DISCUSSION

The analysis described has been applied to a typical turbine for high speed (Mach 3) aircraft to obtain the heat sink requirements to cool the turbine to various metal temperatures and turbine inlet gas temperatures. The ability of satisfying these requirements with various cooling systems was evaluated. These systems included (fig. 1) the use of compressor exit air or engine fuel for direct cooling of the turbine airfoils and the use of fuel, fan exit duct air, or ram air in a heat exchanger to reduce the compressor bleed air temperature. The results are presented in figures 2 to 8, and are discussed in the following sections.

Heat Sink Requirements to Cool Turbines

The heat sink capacity required to cool each of the turbine airfoil rows is dependent on the average gas-to-blade heat transfer coefficient, the surface area of the airfoil row, and the difference between the effective gas and metal temperatures. For the analysis presented herein these temperatures were assumed to be the relative average midspan gas temperature and the average midspan metal temperature. The interrelation of the factors involved results in the heat sink requirements (heat transferred from the gas stream to the airfoils) decreasing linearly with increasing metal temperature for a given gas temperature. This is shown in figure 2 (solid lines) for each row of the two stage turbine assumed. The figure shows the heat sink requirements for reducing the average midspan metal temperatures to levels of 1500° to 2000° F (1090° to 1365° K) for average turbine inlet gas temperatures of 2270°, 2570°, and 2870° F (1515°, 1680°, and 1850° K). It was assumed that vane temperatures of 1800° F (1255° K) and blade temperatures of 1700° F (1200° K) are the allowable limits for a life of 1000 hours with a material such as IN100, based on the criteria established in reference 1. The heat sink requirements for an average turbine inlet gas temperature of 2270° F (1515° K) for these assumptions would be about 590 Btu/sec (0.623 MW) for the first stage vanes, 400 Btu/sec (0.423 MW) for the first stage blades, 220 Btu/sec (0.232 MW) for the second stage vanes, and 110 Btu/sec (0.116 MW) for the second stage blades. The heat sink requirement for the entire turbine is the sum of these numbers; 1320 Btu/sec (1.394 MW). The heat sink requirements for the entire turbine at various blade metal and turbine inlet temperatures are shown in figure 3 (solid lines) for the assumed conditions of the allowable vane temperatures being 100° F (56° K) higher than that of the blades.

Heat Sink Capacities Available

Compressor exit bleed air. - The following is a discussion of the compressor exit bleed air flow required to cool individual rows of turbine airfoils, and to cool the entire turbine. The results of an analysis to illustrate the effect on metal temperature of changing the turbine airfoil thermal effectiveness are also discussed.

Coolant flow requirements: Under steady state conditions, the heat transferred from the gas to the airfoils must be equal to the heat transferred from the airfoils to the cooling air. The heat transferred from the gas to the airfoils, designated in this report as the required heat sink capacity, is shown as solid lines in figure 2 for each turbine row over a range of average midspan metal temperatures and average turbine inlet gas temperatures. The heat transferred from the airfoil to the cooling air, designated in this report as the heat sink capacity available for cooling, is shown as dashed lines in figure 2 for various ratios of compressor exit bleed air to compressor flow over a range of average midspan metal temperatures. The cooling air was assumed to be bled from the compressor exit and to have a temperature of 1200° F (920° K). The airfoils were assumed to be representative of average cooling designs having a thermal effectiveness of 0.5.

The intersection of a dashed line representing the available heat sink capacity at a given coolant flow ratio with a solid line representing the required heat sink capacity at given average turbine inlet gas temperatures results in a heat balance that defines the airfoil metal temperature at these conditions.

A typical example of the use of figure 2 can be shown by assuming a turbine inlet gas temperature of 2270° F (1515° K) and an allowable average midspan blade and vane metal temperature of 1700° and 1800° F (1200° and 1255° K), respectively. For these assumed conditions, the ratio of coolant-to-compressor flow required for the first stage vanes is about 0.045, for the first stage blades 0.038, for the second stage vanes 0.017, and for the second stage blades 0.01.

The summation of the flow requirements for the individual turbine airfoil rows provides the requirements for the entire turbine. For the above assumed conditions, the total coolant flow ratio required would be 0.11. The required coolant flow ratios for individual rows or for the entire turbine at any other condition can be determined in a similar manner.

Airfoil thermal effectiveness: To determine the effect of airfoil thermal effectiveness on available heat sink capacities, thermal effectiveness values of 0.5, 0.75, and 1.0 were assumed in this part of the analysis. The validity of using these values of thermal effectiveness to represent different airfoil cooling designs is presented in appendix B. The cooling design features of airfoils that cover this range of effectiveness values can be obtained from references 3 to 5. The relation of the thermal effectiveness with other factors affecting heat transfer in turbine airfoils is discussed and presented in appendix C.

The heat sink capacities required to cool the entire turbine and the heat sink capacities available in 10 and 20 percent of compressor exit air when using airfoils of various thermal effectiveness are shown in figure 3. This figure is based on the assumption that the average midspan vane metal temperature is a constant 100° F (55° K) higher than that of the blade. From this plot, the benefits of increasing airfoil thermal effectiveness from 0.5 to 1.0, and the potential limits of cooling with compressor exit bleed air up to coolant ratios of 0.20 can be determined.

It can be seen from the figure that at an average turbine inlet gas temperature of 2270° F (1515° K), 10 percent of the compressor flow can cool a two-stage turbine with airfoils having a 0.5 thermal effectiveness to average midspan metal temperatures of 1710° F (1205° K) for the blades and 1810° F (1260° K) for the vanes. These levels of temperature are representative of allowable limits for present day blade and vane materials. At the same cooling airflow ratio, increasing the average turbine inlet gas temperature to 2570° F (1680° K) would result in the blades operating at approximately 1900° F (1310° K), which is considerably above the allowable limit for present day materials.

Increasing the thermal effectiveness from 0.5 to 0.75 at an average turbine inlet gas temperature of 2270° F (1515° K) when using 0.10 of the compressor exit air for cooling would reduce the average midspan metal temperature throughout the turbine by about 75° F (42° K). If the airfoil cooling design was improved to a perfect convection-cooled airfoil ($\eta_t = 1$), although not practically possible, reductions in airfoil metal temperatures of about 60° F (33° K) below that of the 0.75 thermal effectiveness airfoils would result. The potential reductions in airfoil metal temperatures with improvement in airfoil thermal effectiveness become slightly larger at increased turbine inlet gas temperatures. Also, the airfoil metal temperature reduction accompanying an improvement in thermal effectiveness from 0.5 for an average airfoil design to 0.75 for an advanced design is greater than that which would accompany an improvement from 0.75 to 1.0. At an inlet gas temperature of 2570° F (1680° K), for example, an airfoil metal temperature reduction of about 100° F (56° K) can be obtained by increasing thermal effectiveness from 0.5 to 0.75 while only an 80° F (44° K) temperature reduction is obtained by increasing thermal effectiveness from 0.75 to 1.0.

Considering the difficulty involved in increasing the thermal effectiveness of airfoils above a value of 0.75 and the relatively small reduction in airfoil metal temperatures associated with such an increase in thermal effectiveness, other means of reducing metal temperatures must be sought. The curves of figure 3 indicate that if turbine inlet gas temperatures of 2570° F (1680° K) or higher are to be used, even the advanced convection cooled airfoil designs would require about 15 percent of compressor flow for cooling the airfoils alone. Additional flow would be required to cool tip shrouds, supports, and the base regions of the vanes and blades. At a turbine inlet gas temperature of 2870° F (1850° K), the required coolant flow ratio would be over 20 percent of the compressor flow for advanced airfoil cooling designs. Engine performance penalties associated with such high coolant flows may be too large to accept. Therefore, either secondary heat sinks must be utilized to reduce the temperature of the compressor bleed air and increase its available heat sink capacity, or other coolants, such as the engine fuel must be used. Both of these approaches are discussed in the following sections.

Engine fuel. - Direct cooling of all four rows of the turbine using either ASTM Jet A or liquid methane is the first of the systems considered. In the process of cooling the airfoils, the Jet A was assumed to be heated to 700° or 1000° F (645° or 810° K) and the liquid methane was assumed to be heated to 1200° F (920° K). Considerations relative to heating the fuels to these temperatures and the conditions and method used for determining the heat sink capacities available in the fuels were discussed in the section ANALYTICAL PROCEDURE.

No use was made of the thermal effectiveness parameter in describing the cooling design of fuel-cooled airfoils as was done with the air-cooled airfoils. The reasons are that the coolant (fuel) flow path through the turbine airfoil rows is not defined, the inlet fuel temperature to each of the rows probably will not be the same, and for fuel pressures below the critical pressure a phase change will occur in the flow path during which the value of the thermal effectiveness term will not be a valid indicator of the airfoil cooling performance. As a consequence, the results of the analysis of direct cooling with either fuel are presented in terms of the airfoil metal temperatures that satisfy the equality of the heat sink required with the heat sink available for a range of average turbine inlet gas temperatures and given fuel exit temperatures.

ASTM Jet A: The total heat sink capacity available in the ASTM Jet A fuel flow to the primary burner over a range of average turbine inlet gas temperatures from 2270° to 3170° F (1515° to 2020° K) is shown in figure 4 as dashed lines. These lines designated Q_1 and Q_2 represent heating of the fuel to 700° F (645° K) and 1000° F (810° K), respectively, in the process of cooling the turbine airfoils. However, because of practical considerations, these heat sink capacities may not be available for the airfoils alone. Heat pickup, for example, will occur in ducting the fuel among the airfoils. Therefore, curves representing $0.75\ Q_1$ and $0.75\ Q_2$ are shown in the figure. Also shown in the figure, as solid lines, are the heat sink capacities required to cool the turbine over a range of average midspan blade metal temperatures for several values of average turbine inlet gas temperature. The lines for the three lower turbine inlet gas temperatures are repeated from figure 3.

As shown in figure 4, the fuel theoretically has a heat sink capacity to cool the turbine to current allowable metal temperatures at an average turbine inlet gas temperature of slightly less than 2270° F (1515° K) if the fuel is permitted to be heated to 700° F (645° K). If the fuel can be heated to 1000° F (810° K) the average turbine inlet gas temperature can be increased to about 2650° F (1730° K). Even if only 75 percent of the total available heat sink capacity can be used for cooling the airfoils, average turbine inlet gas temperatures as high as 2370° F (1570° K) are possible when the fuel is heated to 1000° F (810° K). If the maximum allowable fuel temperature is 700° F (645° K) and only 75 percent of the available heat capacity can be used, blade and vane metal temperatures of 1800° and 1900° F (1255° and 1310° K), respectively, will result at a turbine inlet gas temperature of 2270° F (1515° K). These metal temperatures are about 100° F (55° K) above allowable values for a 1000-hour life with current materials.

Liquid methane: The total heat sink capacity available in the liquid methane (LCH_4) fuel supplied to the primary burner over a range of turbine inlet gas temperatures is shown in figure 4 as the dashed line designated Q_3. The fuel is assumed heated to 1200° F (920° K) in the process of cooling the airfoils. The figure indicates that even at an average turbine inlet gas temperature of 3170° F (2020° K), liquid methane has a total heat sink capacity that is theoretically sufficient to cool the blades to metal temperatures of 1370° F (1015° K). This blade metal temperature is considerably below the maximum allowable temperature for a 1000-hour blade life, and represents a theoretical potential of increased blade life or reliability. Even if only 75 percent of the heat sink capacity in the LCH_4 were available to cool the airfoils because of heat pickup in ducting the fuel among the airfoil rows (curve designated $0.75\ Q_3$), allowable average midspan metal temperatures of the blades of 1700° F (1200° K) or less could be obtained at average turbine inlet gas temperatures as high as 3000° F (1920° K).

Reduced compressor exit air temperatures. - One of the ways of extending the cooling capacity of compressor bleed air is to reduce its temperature by means of a heat exchanger before using the air to cool the turbine. The potential reductions in the first stage blade midspan metal temperature are shown in figure 5. The conditions selected were those of an average turbine inlet gas temperature of 2270° F (1515° K) and a cooling air-to-compressor flow ratio of 0.025. This flow ratio was selected as representative of that required to provide an average midspan blade metal temperature of 1700° F (1200° K) when using 1200° F (920° K) cooling air in an advanced convection cooled blade design ($\eta_t = 0.75$). Curves are also shown on figure 5 for blades with thermal effectivenss values of 0.5 and 1.0, and for a transpiration cooled blade.

The figure indicates that reductions in the cooling air temperature below that of the compressor exit temperature can provide a greater potential to reduce vane and blade metal temperatures or permit higher turbine inlet gas temperatures than can be obtained by changes in airfoil cooling design or cooling method. From the figure, it can be seen, for example, that at a 1200° F (920° K) cooling air temperature, changing the cooling method from an advanced convection cooled design ($\eta_t = 0.75$) to a transpiration cooled design, results in a reduction of the average midspan metal temperature of about 120° F (67° K). The lower allowable metal temperatures associated with porous materials used for transpiration cooling may negate most of this temperature reduction. However, by reducing the cooling air temperature by 400° F (222° K) from 1200° to 800° F (920° to 700° K), a metal temperature reduction of 190° F (105° K) can be obtained without changing the blade cooling method. The lower air temperature can also be used to reduce the metal temperature of the transpiration cooled design by 250° F (139° K) or to permit lower coolant flows or higher turbine inlet gas temperatures.

Figure 5 also shows that in order to use an average convection cooled blade design ($\eta_t = 0.5$) at the engine and coolant flow conditions assumed, a reduction of the compressor exit bleed temperature from 1200° to 1000° F (920° to 810° K) would be required to attain the assumed allowable blade metal temperature of 1700° F (1200° K).

Further reductions in the cooling air temperatures, with accompanying additional reductions in blade metal temperatures, would provide the potential for increased blade life and reliability. However, lower temperature cooling air will require careful attention to the cooling configurations or designs so as to effectively utilize the additional heat sink capability without increasing structural problems such as those resulting from larger thermal gradients.

The following sections describe the results of the analysis made of the use of the heat sink capacities in the engine fuel, bypass fan exit air (turbofan engine), or ram air to reduce the compressor exit air temperatures. These secondary cooling fluids were used in a heat exchanger with an arbitrarily assumed constant quantity of cooling air (10 percent) being bled from the exit of the compressor. The potential reductions in airfoil metal temperatures or the increases in average turbine inlet gas temperatures made possible by the use of these secondary fluids are presented graphically in figures 6 to 8.

Fuel heat exchanger: The assumption of an allowable fuel temperature out of the heat exchanger fixes the heat sink capacity available in the fuel. The extent to which this heat sink capacity can be used effectively in reducing airfoil metal temperatures is dependent on the thermal effectiveness of the airfoils as discussed previously in the section ANALYTICAL PROCEDURE.

The potential for turbine cooling with a combination of 10 percent compressor exit bleed air and a fuel-air heat exchanger is illustrated in figures 6(a) and (b) for airfoils with thermal effectiveness values of 0.5 and 0.75, respectively. The dashed curves represent the heat sink capacities available over a range of average blade metal temperatures with (1) 10 percent of the compressor exit air, (2) 10 percent of the compressor exit air plus Jet A fuel heated to 700° F (645° K), (3) 10 percent of the compressor exit air plus Jet A fuel heated to 1000° F (810° K), and (4) 10 percent of the compressor exit air plus LCH_4 fuel heated to 1000° F (810° K). The solid lines in the figures show the heat sink capacity required for average turbine inlet gas temperatures between 2270° and 3170° F (1515° and 2020° K) over a range of blade metal temperatures. These data are taken from figure 4. The metal temperatures that satisfy the heat balance for the turbine airfoils are obtained from the intersections of the curves of the heat sink capacity required and available.

A comparison of figures 6(a) and (b) indicates the large benefits in lower metal temperatures or increases in turbine inlet gas temperatures that can be attained by increasing airfoil thermal effectiveness from 0.50 to 0.75. This is evident by the large difference in the spread of the curves of available heat sink capacity for the two values of airfoil thermal effectiveness.

The figures show that for an allowable blade metal temperature of 1700° F (1200° K), the available heat sink capacities in 10 percent of the compressor exit air alone (line 1) will permit average turbine inlet gas temperatures of about 2270° F (1515° K) and 2400° F (1590° K) for airfoils with thermal effectiveness values of 0.50 and 0.75, respectively. When the available heat sink capacity in Jet A fuel heated to 700° F (645° K) is added to that available in 10 percent of the compressor exit air (line 2), the allowable turbine inlet gas temperatures can be increased to about 2400° F (1590° K) and 2670° F (1740° K) for the airfoils with thermal effectiveness values of 0.5 and 0.75, respectively. Permitting the Jet A to be heated to 1000° F (810° K) at the heat exchanger exit (line 3) will permit operation at average turbine inlet gas temperatures of 2540° F (1670° K) and 3000° F (1920° K) for airfoils with thermal effectiveness values of 0.50 and 0.75, respectively.

The additional heat sink capacity provided by Jet A fuel can be used in other ways. At an average turbine inlet gas temperature of 2570° F (1680° K), figures 6(a) and (b) indicate that the additional heat sink capacity provided by Jet A heated to 1000° F (810° K) (line 3) can provide reductions in blade metal temperatures of 190° F (106° K) for airfoils with 0.5 thermal effectiveness and 230° F (128° K) for airfoils with 0.75 effectiveness compared to the airfoil metal temperatures resulting from the use of 10 percent of the compressor exit air alone. Other results of the analysis (not shown in fig. 6) indicate that the additional heat sink capacity provided by Jet A heated to 1000° F (810° K) is approximately the same as that provided by increasing the compressor bleed from 10 to 20 percent for the 0.5 thermal effectiveness airfoils. Thus for given turbine inlet gas and blade metal temperatures, the heat capacity in Jet A can be used to reduce compressor exit air bleed requirements by one-half.

The use of liquid methane-fuel as the secondary cooling fluid in the heat exchanger can provide even greater heat sink capacity than the use of Jet A fuel. This can be seen in figures 6(a) and (b). When the additional available heat sink capacity of LCH_4 fuel heated to 1000° F (810° K) is added to that available in 10 percent of compressor exit air, current allowable blade metal temperature of 1700° F (1200° K) can be maintained at average turbine inlet gas temperatures as high as 2720° F (1770° K) with 0.5 thermal effectiveness airfoils, and about 3500° F (2200° K) with 0.75 thermal effectiveness airfoils. This combined heat sink capacity can also be used to reduce airfoil metal temperatures or the required coolant flow rate. At an average turbine inlet gas temperature of 2570° F (1680° K), for example, the combined heat sink capacity can reduce the average midspan blade metal temperatures by as much as 260° F (145° K) and 325° F (180° K) for airfoils with 0.5 and 0.75 thermal effectiveness values, respectively, compared to the metal temperatures obtained using only the heat sink capacity available in 10 percent of the compressor exit air.

The effect of the reduction of compressor exit bleed air temperatures on the cooling air flow distribution to the individual rows of airfoils needs some discussion. An initial distribution of cooling air flow to individual airfoil rows with 0.5 thermal effectiveness for an arbitrarily selected operating condition is shown in table I. These conditions are (1) average turbine inlet gas temperature of 2570° F (1680° K), (2) average midspan blade metal temperature of 1900° F (1310° K) with the vanes temperature assumed to be a constant 100° F (55° K) higher, and (3) a total coolant to the turbine of 10 percent of compressor air. The conditions and the data in table I were obtained from figures 2 and 6(a).

It is apparent from figure 6(a) that combining the heat sink capacities of Jet A fuel heated to 1000° F (810° K) and 10 percent compressor exit bleed air will reduce blade metal temperature to 1720° F (1210° K) (with a corresponding vane temperature of 1820° F (1266° K)). Although this figure indicates that these lower metal temperatures can be obtained with the increased available heat sink capacity in the coolant flow to the entire turbine, a reapportionment of the flow to the individual rows is necessary to satisfy the heat sink requirements of each row. This can be seen in table I. The heat sink capacity required to obtain these reduced metal temperatures in the individual airfoil rows is tabulated against the heat sink capacity available when the reduced cooling air temperature is applied to the initially assumed apportionment of flow to each airfoil row. It is apparent from the table that the first stage turbine vanes and blades will be overcooled relative to those in the second stage because of the larger gain in available heat sink capacity associated with the higher first stage cooling air flows. The coolant air flow to the second stage airfoils must be increased and the flow to the first stage airfoils decreased to satisfy the heat balance for the individual rows.

Although potentially large benefits have been indicated herein with the use of fuel-air heat exchangers, the ultimate benefit or cost to an engine and aircraft is dependent on a large number of factors. Some of the more obvious are: the ability of the fuel to withstand high temperature levels for extended periods of time without significant deposits in the heat exchanger that would effect its heat transfer or pressure drop characteristics; the weight and size of the heat exchanger plus penalties that the requirement for low pressure drop on the cooling air side of the heat exchanger may impose; and problems that off-design and transient operation may impose on the system.

Fan exit air heat exchanger: The potential reductions in turbine airfoil metal temperatures or increases in average turbine inlet gas temperature attainable with the combined heat sink capacity of 10 percent of the compressor exit air and the entire fan bypass exit flow are shown in figure 7. Increases in the fan exit duct air temperatures of 10°, 20°, and 30° F (6°, 11°, and 17° K) were considered. The data shown in the figure are for turbine airfoils with a 0.5 thermal effectiveness. The dashed lines in the figure represent the available heat sink capacity. The solid lines (taken from fig. 3) represent the heat sink capacity required to cool the turbine over a range of airfoil metal temperatures for average turbine inlet gas temperatures of 2270°, 2570°, and 2870° F (1515°, 1680°, and 1850° K).

It can be seen from figure 7 that for a given blade metal temperature, the additional heat sink capacity provided by a 20° F (11° K) rise in fan exit duct air temperature will permit an increase of about 160° F (89° K) in the average turbine inlet gas temperature above the temperature that can be attained when using compressor exit bleed air alone. It is also evident from this figure that, at any given turbine inlet gas temperature, a 20° F (11° K) rise in fan exit duct air temperature can provide approximately a 100° F (55° K) reduction in average midspan blade metal temperature.

Other results of the analysis (not given on fig. 7) show that the heat sink capacity available in the fan exit air can be used to appreciably reduce the compressor exit bleed air required for turbine cooling. For example, an airfoil with a thermal effectiveness of 0.50 operating at a turbine inlet gas temperature of 2570° F (1680° K) would require about 20 percent of the compressor airflow to cool the blades to a metal temperature of 1725° F (1215° K). By utilizing the heat sink capacity associated with a 30° F (17° K) temperature rise in the fan exit air, the required coolant flow can be reduced to 12 percent of the compressor airflow.

The influence that airfoil thermal effectiveness has on the fan exit air heat exchanger system was also evaluated in the analysis. For example, if the thermal effectiveness of the airfoils is increased from 0.50 to 0.75, the heat sink capacity available from a 20° F (11° K) rise in fan exit duct air temperature can, for a given blade metal temperature, provide increases in the average turbine inlet gas temperature from 160° F (89° K) to about 240° F (133° K), respectively, above the gas temperature level that is permissible when using the heat sink capacity in compressor exit air alone. Also, a 20° F (11° K) rise in the fan exit duct air temperature can provide blade metal temperature reductions of about 130° F (72° K) at a given average turbine inlet gas temperature.

Although large reductions in turbine metal temperature are theoretically possible with a fan exit duct heat exchanger, a detailed study must be made of various factors affecting the practicality and net benefits of this system as applied to an engine and aircraft. Some of these factors were mentioned in the prior section. Additional factors are the effect of the heat exchanger on the off-design performance and dynamic response characteristics of the engine, and the increase in fan pressure ratio that would be required to compensate for the pressure drop through the heat exchanger.

Ram air heat exchanger: The potential reductions in turbine blade metal temperatures or increases in turbine inlet gas temperatures available from a ram air-compressor exit bleed air heat exchanger system are shown in figure 8 for airfoils with a thermal effectiveness of 0.50. The heat sink capacity available, presented in this figure as dashed lines, is the combined heat sink capacity of 10 percent of the compressor airflow and three ratios of ram airflow-to-compressor airflow (0.05, 0.10, and 0.15). At all three ram airflow ratios, the ram air temperature at the exit of the heat exchanger is assumed to be 1000° F (810° K). The solid lines in figure 8 represent the heat sink capacities required for the various average turbine inlet gas temperatures over a range of average blade midspan metal temperatures. These lines first shown in figure 3 are repeated in this figure.

It is apparent from figure 8 that, at any given blade metal temperature, the additional available heat sink capacity provided by an increment of ram airflow equal to 10 percent of compressor flow will permit increases in turbine inlet gas temperatures of about 190° F (106° K). At any given turbine inlet gas temperature, reductions in blade metal temperatures of about 120° F (67° K) can be obtained for this same increment of ram airflow. Other results of the analysis (not shown on fig. 8) indicate that for airfoils with a thermal effectiveness value of 0.75, increases in turbine inlet gas temperature of about 290° F (160° K) or decreases in blade metal temperatures of about 150° F (83° K) would be possible for an increment of ram airflow equal to 10 percent of the compressor airflow.

Any benefits resulting from the application of this cooling system to an engine and aircraft entails considerations of problems similar to those stated in the previous two sections.

SUMMARY OF RESULTS

The following is a summary of the results obtained from an analysis made of the requirements and capabilities of cooling systems for turbines of typical Mach 3 cruise engines where the temperature of the compressor bleed air available for turbine cooling was 1200° F (920° K).

1. A simplified method was developed for determining the approximate coolant flows required to cool individual rows of airfoils or the entire turbine over a range of turbine inlet gas temperatures for various values of airfoil thermal effectiveness.

2. Average midspan turbine blade metal temperature of about 1635° to 1710° F (1160° to 1205° K) (with vane temperatures 100° F (58° K) higher) can be expected at an average turbine inlet gas temperature of 2270° F (1515° K) when 10 percent of the compressor exit air is used for cooling. The lower metal temperatures would be attainable if all the airfoils in the turbine, both vanes and blades, had thermal effectiveness values of 0.75, which is representative of advanced cooling designs. The higher metal temperatures would result if the airfoils had thermal effectiveness of 0.50, which is representative of average current cooling designs. Metal temperatures of about 1700° F (1200° K) for blades and about 1800° F (1255° K) for vanes are representative of allowable temperature levels for 1000-hour life with current materials such as IN100.

3. When compressor exit air at a temperature of 1200° F (920° K) is used directly as the coolant, relatively small metal temperature reductions can be expected by improving airfoil thermal effectiveness beyond current advanced convection cooled designs. When using 10 percent of the compressor air for cooling, improving the thermal effectiveness of the airfoils (vanes and blades) from 0.75 to 1.0 (i.e., achieving perfect convection-cooled airfoils) would permit reductions in average midspan metal temperatures of only about 60° F (33° K) at a turbine inlet gas temperature of 2270° F (1515° K).

4. Reduction of the cooling air temperature can provide greater potential for turbine cooling than improvements in cooling design or cooling method alone. At a turbine inlet gas temperature of 2270° F (1515° K), a cooling air temperature of 1200° F (920° K) and a cooling airflow equal to 2.5 percent of the compressor flow, the average midspan metal temperature for a first stage blade of an advanced convection-cooled design would be 1700° F (1200° K). A transpiration cooled blade under the same environment would have a metal temperature about 120° F (67° K) lower. However, by reducing the coolant temperature by 400° F (222° K), a reduction in the midspan metal temperature of about 190° F (105° K) can be obtained with the advanced convection-cooled design. This same reduction of coolant temperature would also allow a 250° F (139° K) reduction in the

metal temperature of the transpiration-cooled design. The lower air temperatures could also be used to permit operation at lower coolant flows or higher turbine inlet gas temperatures.

5. ASTM Jet A and liquid methane fuel, when used as direct coolants have the potential heat sink capacity to cool the vanes and blades of the entire turbine to current allowable metal temperatures at average turbine inlet gas temperatures of 2370° and 3000° F (1570° and 1920° K), respectively. In the process of cooling the turbines, the Jet A fuel was permitted to be heated to 1000° F (810° K) and methane to 1200° F (920° K).

6. The heat sink capacity available in the engine fuel, in the fan discharge air of a turbofan engine, or in ram air, when used in combination with a heat exchanger to reduce the temperature of the compressor bleed air, can provide substantial increases in turbine inlet gas temperature, decreases in airfoil metal temperatures, or decreases in the required coolant flow rate. For example, combining the heat sink capacity of 10 percent of compressor exit air with that of liquid methane fuel that is heated to 1000° F (810° K), in flowing through a heat exchanger, can permit operation of average convection-cooled airfoil designs to turbine inlet gas temperatures of about 2720° F (1770° K), compared to a turbine inlet gas temperature of about 2270° F (1515° K) when using the compressor exit air alone.

7. Although potentially large benefits in turbine cooling performance may be obtained by reducing compressor exit bleed air temperatures through the use of various heat exchangers systems, the net benefit or cost to an engine or aircraft is dependent on a large number of complex factors. Some of the more obvious factors are the ability of the fuel to withstand high temperature levels for extended periods of time without the formation of significant deposits which may effect the heat transfer or fuel pressure drop in the exchanger; heat exchanger weight and size penalties that the requirement for a low pressure drop on the cooling air side of the exchanger may impose; and problems that off-design and transient engine operation may impose on the heat exchanger and engine. The analysis of the effect of factors such as these was beyond the scope of this investigation.

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APPENDIX A

SYMBOLS

A	flow area
C_p	specific heat at constant pressure
$C_{1,2,\ldots,7}$	constants (used in appendix C)
d_h	airfoil internal hydraulic diameter
F	ratio, $\bar{h}_g S_g / W_c C_{p_c}$
f	fuel-air ratio
g	gravitational constant
H_f	lower heating value of fuel
$\bar{h}_c$	average internal heat transfer coefficient
$\bar{h}_e$	average effective internal heat transfer coefficient including effect of augmented surface area such as fins
h_f	enthalpy of fuel entering combustor
h_g	local gas-to-blade heat transfer coefficient
$\bar{h}_g$	average gas-to-blade heat transfer coefficient
h_1	enthalpy of air entering combustor
h_2	enthalpy of gas at combustor exit
k	thermal conductivity
M_g	average gas Mach number relative to airfoil
P_g	average gas total pressure relative to airfoil
Pr	Prandtl number
Q_A	heat sink capacity available to cool turbine
Q_o	heat sink capacity of secondary fluid in heat exchanger
Q_R	heat sink capacity required to cool turbine
R	gas constant
Re	Reynolds number, $\rho_g V_g x / \mu_g$
S	surface area of all airfoils in a row
ΔT_c	temperature reduction, cooling air
$\bar{T}_c$	average cooling air temperature
T_{ci}	average cooling air temperature, inlet to airfoil
T_{co}	average cooling air temperature, outlet from airfoil
ΔT_d	temperature rise across fan exit duct air heat exchanger

T_g average midspan gas total temperature relative to airfoil

$\bar{T}_g$ average total temperature of gas relative to airfoil

$T_{g,M}$ midspan hotspot gas total temperature ahead of the vanes

T_m average midspan airfoil metal temperature

$\bar{T}_m$ average metal temperature of entire airfoil

ΔT_r temperature rise across ram air heat exchanger

t_{aw} local adiabatic wall temperature

t_{co} local cooling air outlet temperature

t_g local gas total temperature relative to airfoil

t_m local metal temperature

V_g average gas velocity relative to airfoil

W flow rate

w flow rate per unit span length of airfoil surface

x distance from leading edge

γ ratio of specific heats

η_B combustion efficiency

η_t thermal effectiveness, $(T_{co} - T_{ci})/(\bar{T}_m - T_{ci})$

μ viscosity

ρ_g average gas density relative to airfoil

φ temperature ratio, $(\bar{T}_g - \bar{T}_m)/(\bar{T}_g - T_{ci})$

ψ parameter, $C_p(dw_c/dx)/h_g$

Subscripts:

bl boundary layer

c cooling air or cooling air side

d fan exit duct air

E compressor inlet

f fuel

g combustion gas or combustion gas side

r ram air

APPENDIX B

THERMAL EFFECTIVENESS OF COOLED TURBINE AIRFOILS

Metal temperatures of seven configurations of first stage turbine blades predicted by the detailed three-dimensional computer analysis of references 3 to 5 are shown in figure 9. Each point on the figure represents the average midspan metal temperature for the particular blade design at its design cooling air-to-compressor air flow ratio. Of the seven configurations, three were convection cooled, one was film cooled, one was transpiration cooled, and two utilized a combination of convection and film or transpiration cooling. The curves shown on figure 9 are the average midspan metal temperatures predicted by the approximate method of this report for convection cooled blades with thermal effectiveness values of 0.50, 0.75, and 1.0 and for transpiration cooled blades over a range of cooling air-to-compressor flow ratios from 0 to 0.04.

Using the data of reference 3 and equation (6), thermal effectiveness values of 0.54, 0.85, and 0.82 were determined for the three convection cooled blade designs represented in figure 9 by the circle, square, and diamond symbols, respectively. The figure indicates that the values of thermal effectiveness for two of these designs (circle and square symbols) were closely predicted using the approximate method of this report. The figure also indicates that assigning a thermal effectiveness of 0.5 to what are currently considered average convection cooled blade designs and a value of 0.75 to advanced convection cooled blade designs, as was done in the body of this report, is a reasonably good assumption.

The figure also indicates that when the equations in this report are applied to predict average midspan metal temperatures of transpiration cooled blades, good agreement was obtained with that predicted by the data of reference 4.

No simple direct method was apparent for the prediction of average midspan metal temperatures for film-cooled blade designs. This is because of the lack of adequate correlations or theories, and because of the wide variations in film cooling hole shapes, sizes, spacings, arrays, etc. that are applicable to airfoils utilizing this method of cooling. However, comparing the average midspan metal temperature predicted by the detailed analysis of reference 4 for a completely film cooled blade with the temperature predicted by the approximate method developed herein for an ideal convection cooled blade ($\eta_t = 1$) would indicate that a thermal effectiveness of 1 might be used initially to predict the performance of film-cooled blades. By similar reasoning, figure 9 shows that a thermal effectiveness of 0.75 might be used initially to predict the performance of blade designs which utilize a combination of convection and film or transpiration cooling.

APPENDIX C

RELATION OF PARAMETERS AFFECTING HEAT TRANSFER

A simple but useful relation of factors which define the heat balance, coolant flow, and cooling performance of cooled airfoils can be obtained by modifying and rearranging equations presented in the ANALYTICAL PROCEDURE section.

When the heat sink capacity required to cool the airfoil (eq. (1)) is equated to the heat sink capacity available in the cooling air (eq. (5)), the following is obtained:

$$\bar{h}_g S_g \left(\bar{T}_g - \bar{T}_m\right) = W_c C_{p_c} (T_{co} - T_{ci}) \tag{C1}$$

When the equation for airfoil thermal effectiveness

$$\eta_t = \frac{T_{co} - T_{ci}}{\bar{T}_m - T_{ci}} \tag{C2}$$

is introduced into equation (C1) and the terms rearranged, the following can be obtained:

$$\frac{\bar{T}_g - \bar{T}_m}{\bar{T}_g - T_{ci}} = \frac{1}{\left(\dfrac{\bar{h}_g S_g}{W_c C_{p_c}}\right)\dfrac{1}{\eta_t} + 1} \tag{C3}$$

The term on the left side of the equation describes the temperature performance of the airfoil relative to the temperature of the gas and cooling air, and is the temperature ratio parameter φ commonly referred to in turbine cooling literature. For constant airfoil geometry and cooling air properties, the term in the parenthesis in the denominator of the right side of the equation defines the ratio of the gas-to-blade heat transfer coefficient to the blade coolant flow rate. This term is designated herein as F. Rewriting the equation gives

$$\varphi = \frac{1}{\dfrac{F}{\eta_t} + 1} \tag{C4}$$

Thus it can be seen that the three parameters (φ, F, and η_t) will describe the airfoil heat transfer performance. A plot of the interrelation of these terms is shown in figure 10. It can be seen from this figure, that for low values of F (i.e., conditions of low gas-to-blade heat transfer coefficient or high coolant flow to the airfoil) high values of temperature ratio φ are obtained over a wide range of values of thermal effectiveness η_t. At high values of F, only low values of φ can be obtained regardless of the value of η_t.

Data points are shown in figure 10 for two different convection-cooled first stage turbine vane designs of reference 3 operating at the same engine and flight conditions of this report but are located at the circumferential hot spot downstream of the burner. These vanes have F values of 0.70 and 0.64, with corresponding φ values of 0.47 and 0.56, and η_t values of 0.63 and 0.81.

If the geometry of an airfoil is assumed to be fixed and a data point of its performance in terms of the factors in equation (C4) is available either from detail calculations of airfoil temperatures or from experimental tests, an operating line can be obtained for the airfoil over a range of operating conditions. First, the heat transferred through the airfoil wall is equated to the heat picked up by the cooling air

$$\bar{h}_e S_c \left(\bar{T}_m - \bar{T}_c \right) = W_c C_{p_c} (T_{co} - T_{ci}) \tag{C5}$$

where it is assumed that

$$\bar{T}_c = \frac{T_{co} + T_{ci}}{2}$$

Then, the assumptions are made that (1) the effective internal heat transfer coefficient $\bar{h}_e$ is proportional to $\bar{h}_c$ (the heat transfer coefficient which does not include the effect of augmented surface area), and (2) turbulent flow exists in the coolant passages. This gives

$$\bar{h}_e = C_1 \bar{h}_c = C_2 \left(\frac{W_c d_h}{A_c \mu_c} \right)^{0.8} Pr_c^{0.4} \frac{k_c}{d_h} \tag{C6}$$

which for a given airfoil geometry and constant air properties reduces to

$$\bar{h}_e = C_3 W_c^{0.8} \tag{C7}$$

Substituting equation (C7) into equation (C5) gives

$$\frac{\overline{T}_m - \overline{T}_c}{T_{co} - T_{ci}} = C_4 W_c^{0.2} \tag{C8}$$

Another required relation is obtained by solving equation (C1) for T_{co} and rewriting in terms of F:

$$T_{co} = F(\overline{T}_g - \overline{T}_m) + T_{ci} \tag{C9}$$

Substituting equation (C9) into equation (C8) and using the relation of equation (C4) gives

$$\frac{1}{\eta_t} = \frac{1}{F}\left(\frac{1}{\varphi} - 1\right) = \frac{1}{2}\left(C_5 W_c^{0.2} + 1\right) \tag{C10}$$

The constant C_5 (which describes the airfoil internal geometry) can be determined from the one known data point by using equation (C10), and the operating lines (dashed curves on fig. 10) for the airfoil can then be drawn. Because F is a function of both the gas and coolant flow conditions, examination of equation (C10) indicates that a separate operating line would be required for each heat load condition $(\overline{h}_g S_g)$, for the airfoil. Examination of the factors contained in constant C_5, show that C_5 is proportional to the size of the coolant passages and inversely proportional to the internal surface area of the coolant passages. Therefore, the smaller the value of the constant C_5, the "better" the internal coolant geometry of the blade.

The operating lines for the heat load conditions of the two vanes of reference 3 are shown as dashed curves through the data points in figure 10. The geometry constants C_5 of these vanes are 1.0 and 1.6. Also shown in figure 10 are operating lines for values of C_5 of 0.52 and 5.2 which represent vanes with "better" and "worse" internal geometry, respectively. Extreme values of C_5 were chosen to show the characteristic shape of the operating lines of airfoils with vastly different internal geometry. Even though these extreme values may not represent practical airfoil configurations, several interesting observations can be made. For example, the figure shows that (1) only airfoils with large internal surface areas and small coolant passages (designated by the smallest value of C_5 in the figure) can have high thermal effectiveness at high coolant flow rates (low values of F), (2) at high coolant flow rates these airfoils will have high values of φ (average metal temperatures approaching the cooling air supply temperature) over a wide range of thermal effectiveness, (3) at the same high coolant flow rates ($F \leq 0.1$), airfoils with large (ineffective) coolant passages (designated by the largest value of C_5 in the figure) exhibit a rapid decrease in φ without much of a change in thermal effectiveness, and (4) higher thermal effectiveness with airfoils with these large ineffective coolant passages can only be obtained with large reductions in coolant flows.

This, however, results in φ approaching zero so that little cooling of the airfoils is accomplished.

For the lower segments of the operating curves on figure 10 which represent reduced coolant flow rates, a point is reached where transition from turbulent to laminar flow will occur within the coolant flow passages. At this point, the operating curves shown on the figure will no longer be applicable, and will need to be modified to account for laminar flow. The heat transfer coefficient for laminar flow is independent of flow rate and is only dependent on the coolant passage geometry (ref. 12). Therefore,

$$h_c = C_6 \tag{C11}$$

With manipulations similar to those used to obtain equation (C10), the following equation can be obtained

$$\frac{1}{\eta_t} = \frac{1}{F}\left(\frac{1}{\varphi} - 1\right) = \frac{1}{2}\left(C_7 W_c + 1\right) \tag{C12}$$

A new constant C_7 can then be obtained from data taken in the laminar flow range for the particular airfoil being considered. The operating curves can then be extended into this flow regime.

The operating curves in figure 10 are shown to extend over an extreme range of coolant flows because no limitation on coolant supply pressure was assumed. In the practical case where the coolant supply pressure is limited, care must be exercised in determining the range of coolant flow rates (in terms of the parameter F) that will be acceptable for a given cooling configuration and its associated cooling air pressure drop characteristics.

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TABLE I. - DIFFERENCES IN REQUIRED AND AVAILABLE HEAT SINKS OF INDIVIDUAL AIRFOIL ROWS CAUSED BY COOLING AIR TEMPERATURE REDUCTION

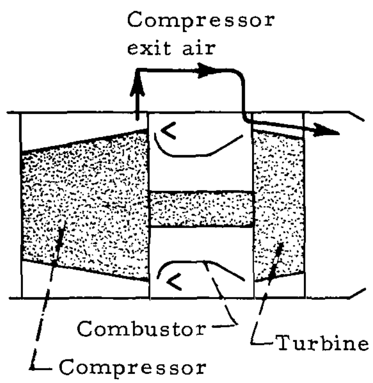
[Conditions: Turbine inlet gas temperature, 2570° F (1680° K); cooling thermal effectiveness of airfoils, 0.5; total cooling air-to-engine flow ratio, 0.10.]

Turbine row	Coolant flow[a] ratio, W_c/W_E	Heat sink available[b] with reduced air temperature, Btu/sec (MW)	Heat sink required,[c] Btu/sec (MW)
First stage vane	0.037	960 (1.013)	875 (0.925)
First stage blade	.032	747 (0.790)	695 (0.734)
Second stage vane	.017	441 (0.466)	500 (0.528)
Second stage blade	.014	327 (0.346)	405 (0.428)
Totals	0.100	2475 (2.615)	2475 (2.615)

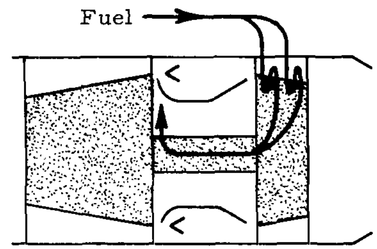
[a]Based on a total coolant flow ratio of 0.10 required to satisfy a blade metal temperature of 1900° F (1310° K) with the vane temperature assumed to be 100° F (55° K) higher and with 1200° F (920° K) cooling air supply temperature.

[b]Available with air temperature reduced by about 473° F (263° K) and at average midspan blade metal temperatures of 1720° F (1210° K) with the vanes 100° F (55° K) higher.

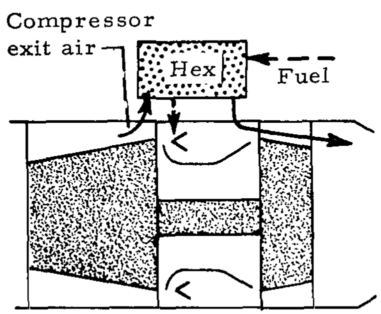
[c]Required for average midspan blade metal temperature of 1720° F (1210° K) with the vanes 100° F (55° K) higher.



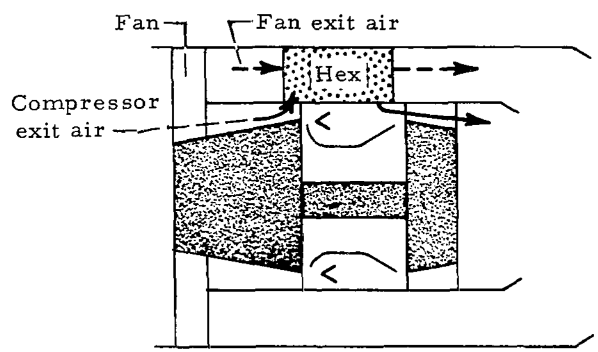
(a) Direct cooling with compressor exit bleed air.



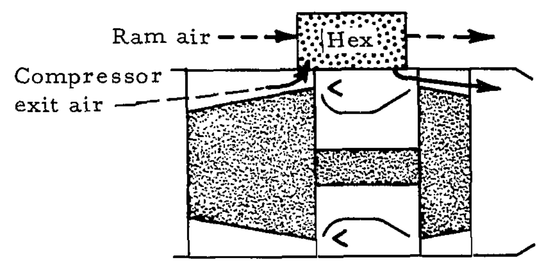
(b) Direct cooling with fuel (Jet A or LCH_4).



(c) Reduction of compressor exit air temperature with fuel (Jet A or LCH_4) heat exchanger.



(d) Reduction of compressor exit air temperature with fan exit duct air heat exchanger (turbofan engine).



(e) Reduction of compressor exit air temperature with ram air heat exchanger.

Figure 1. - Turbine cooling systems.

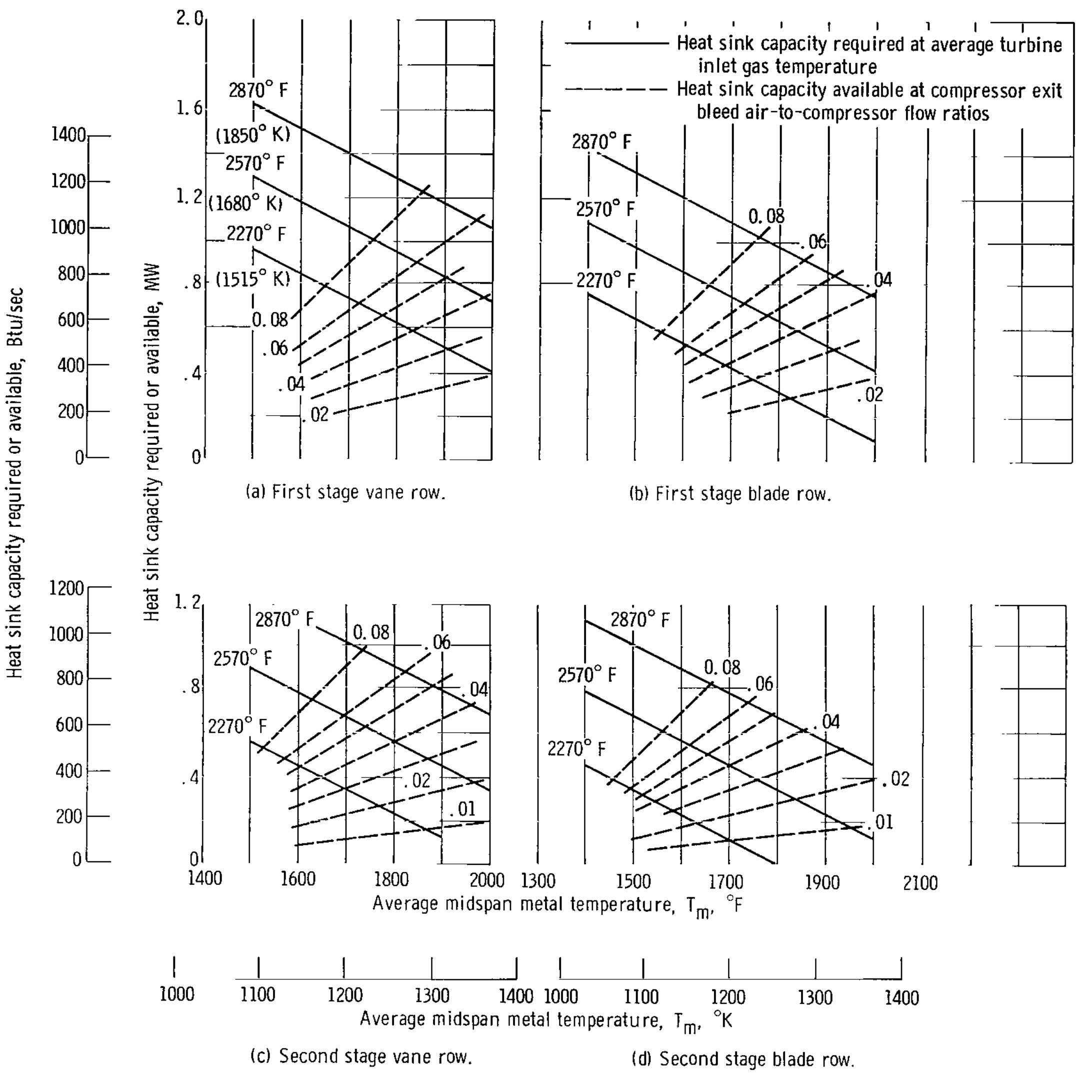


Figure 2. - Heat sink capacity required for cooling individual rows of turbine airfoils and the heat sink capacity available in compressor exit bleed air for cooling airfoils with a thermal effectiveness of 0.5.

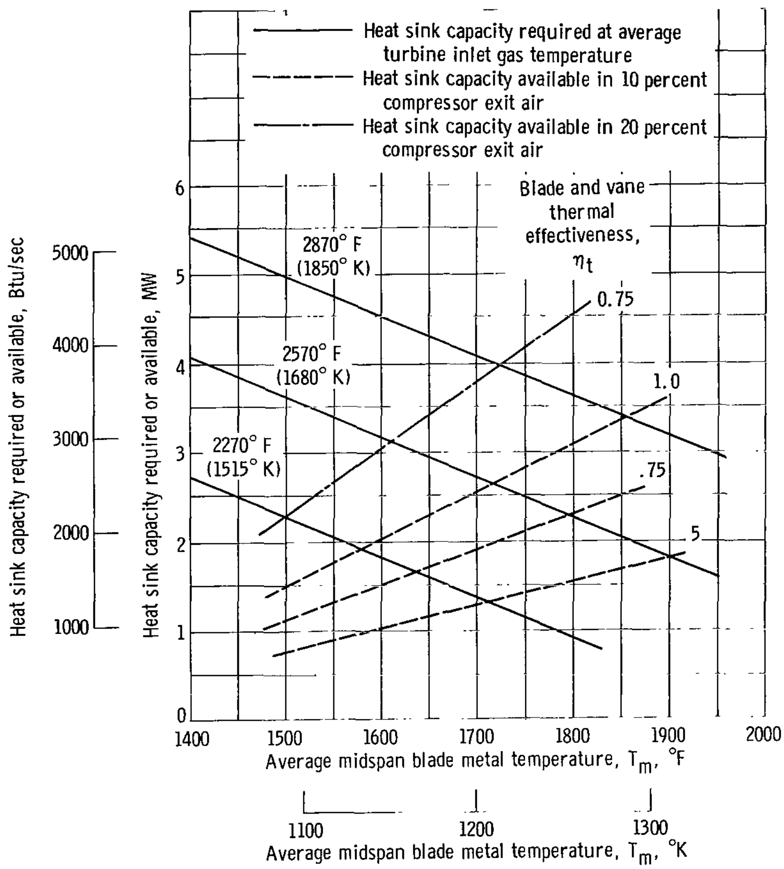


Figure 3. - Heat sink capacity required for cooling a two stage turbine and the heat sink capacity available in compressor exit bleed air. (Vane temperatures assumed to be 100° F (55° K) higher than blade temperatures).

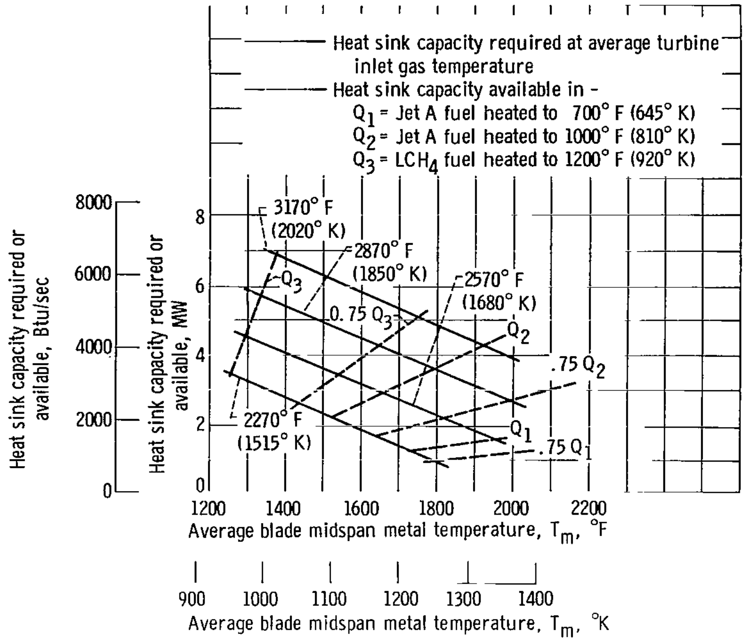


Figure 4. - Heat sink capacity required for cooling a two stage turbine and the heat sink capacity available in engine fuel (ASTM jet A and liquid methane). (Vane temperatures assumed to be 100° F (55° K) higher than blade temperatures.)

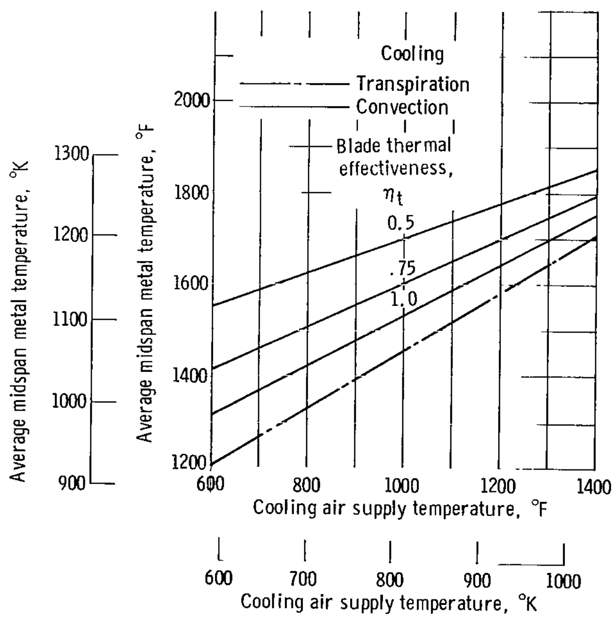
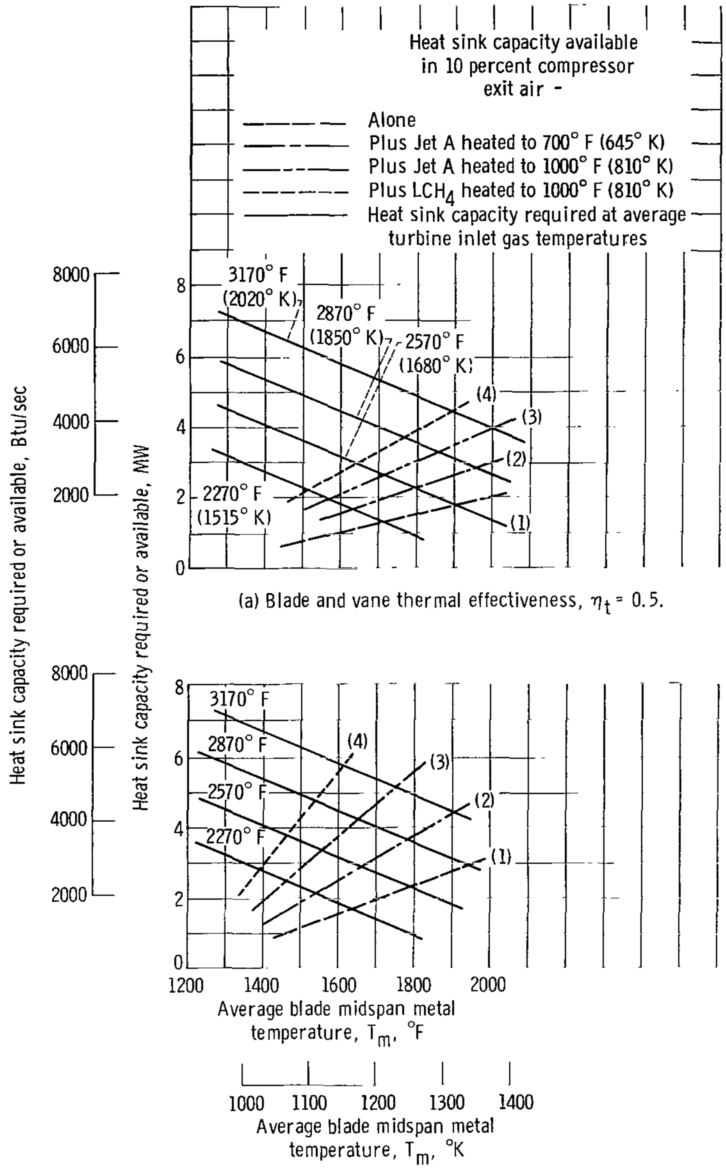


Figure 5. - Effect of cooling air temperature on first stage blade metal temperatures. (Conditions: average turbine inlet gas temperature of 2270° F (1515° K) and ratio of cooling air-to-compressor flow of 0.025.)



(b) Blade and vane thermal effectiveness, $\eta_t = 0.75$.

Figure 6. - Heat sink capacity required to cool a two stage turbine and the heat sink capacity available in 10 percent of the compressor exit air plus fuel. (Vane temperatures assumed to be 100° F (55° K) higher than blade temperatures.)

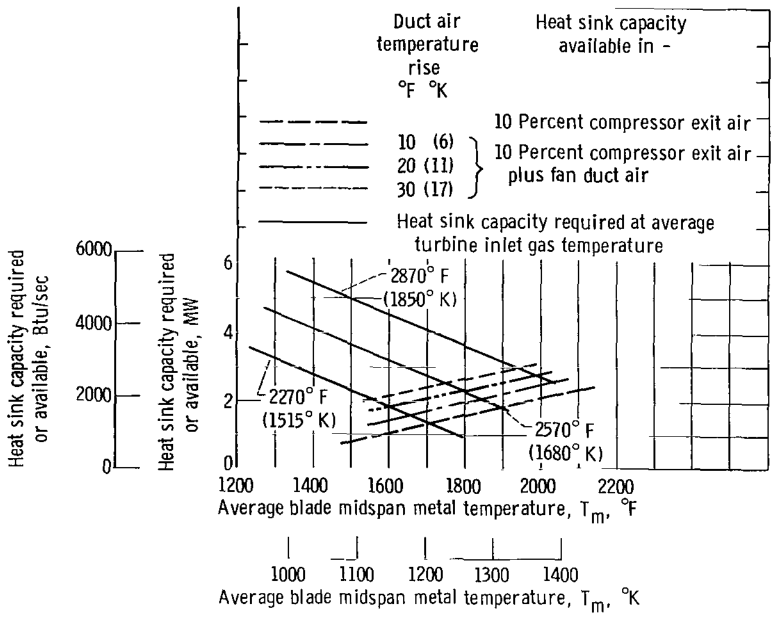


Figure 7. - Heat sink capacity required to cool a two stage turbine and the heat sink capacity available in 10 percent compressor exit air plus the fan exit, duct air of a turbofan engine. (Bypass ratio 1.3; blade and vane thermal effectiveness 0.5; vane temperatures assumed to be 100° F (55° K) higher than blade temperatures.)

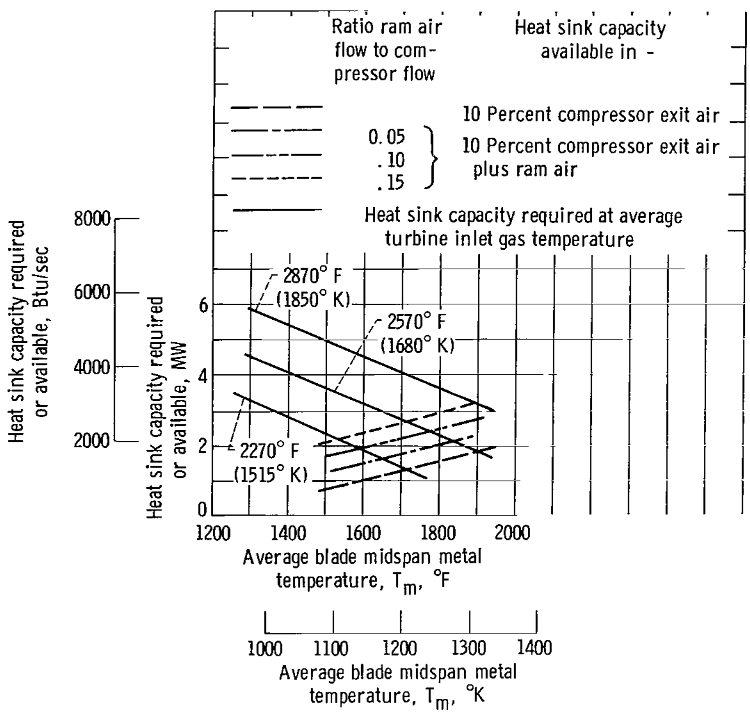


Figure 8. - Heat sink capacity required to cool a two stage turbine and the heat sink capacity available in 10 percent compressor exit air plus ram air. (Blade and vane thermal effectiveness of 0.5; vane temperatures assumed to be 100° F (55° K) higher than blade temperatures.)

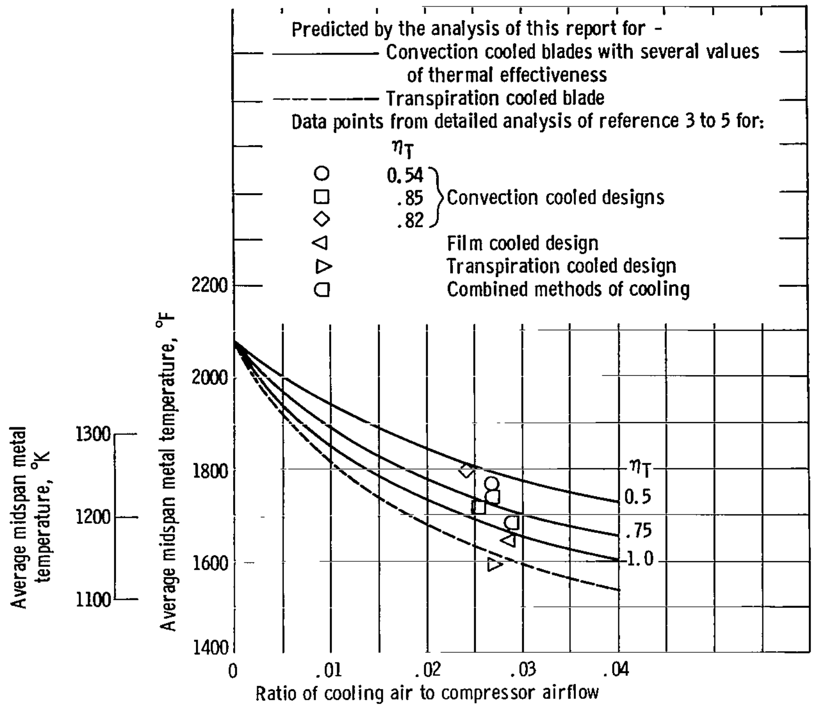


Figure 9. - Comparison of average midspan metal temperatures of first stage turbine blade predicted by detail heat transfer and flow analysis of reference 3 to 5 with temperatures predicted by the method of this report. (Average turbine inlet gas temperature 2270° F (1515° K)).

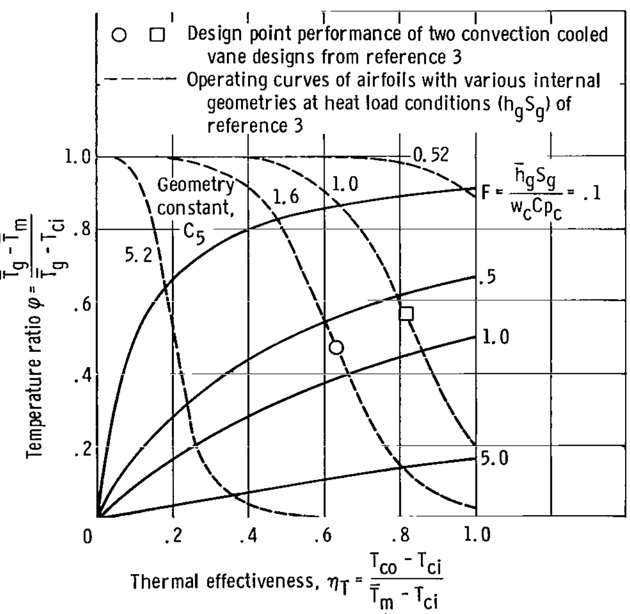


Figure 10. - Relation of parameters affecting the cooling of turbine airfoils.

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