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SUMMARY OF NASA AERODYNAMIC AND HEAT TRANSFER STUDIES IN TURBINE VANES AND BLADES

by Thomas P. Moffitt, Francis S. Stepka, and Harold E. Rohlik
Lewis Research Center
Cleveland, Ohio 44135

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SUMMARY OF NASA AERODYNAMIC AND HEAT TRANSFER STUDIES IN TURBINE VANES AND BLADES

by Thomas P. Moffitt, Francis S. Stepka, and Harold E. Rohlik
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

Aerodynamic effects of trailing edge geometry, hole size, angle, spacing, and shape have been studied in two- and three-dimensional cascades and in a warm turbine test series. Heat transfer studies have been carried out in various two- and three-dimensional test facilities in order to provide corresponding heat transfer data. Results are shown in terms of cooling effectiveness and aerodynamic efficiency for various coolant fractions, coolant-primary temperature ratios, and cooling configurations.

CYCLE AND MISSION STUDIES for advanced commercial transport and military aircraft show that turbine inlet temperatures ranging from 1600 to more than 1900 K are needed. Gas temperatures in that range mandate sophisticated turbine cooling methods to protect the vanes, blades, and endwalls. Internal convection must be combined with film cooling or a thermal barrier coating in order to utilize the coolant efficiently and provide satisfactory turbine life. Also, the effects of cooling on turbine aerodynamic and overall engine performance must be quantified so that rigorous engine studies can be employed to select optimum temperatures and pressures for the various aircraft and their missions. Tradeoffs among such factors as fuel consumption, thrust-to-weight ratio, cost and engine life may then be systematically considered.

NASA and the Department of Defense have funded a number of programs ranging from component evaluations through engine development. These have included analytical and experimental studies: in-house and under contract with various universities and engine companies. Many of these have been reported in general publications while some of the information has been proprietary and/or classified.

The material reported herein was selected from a large number of NASA reports as well as recent unpublished information. The experimental studies were conducted in two- and three-dimensional turbine vane cascades, a flat-plate heat transfer tunnel, a flow visualization tunnel, a warm core turbine test facility, and a research turbojet turbine. Aerodynamic performance, cooling effectiveness, and visualized coolant injection into the gas stream were determined for a number of hole sizes, hole distributions, and injection angles with ranges of coolant flow and efflux energy.

This paper highlights the research program results, relates heat transfer to aerodynamic performance, and describes current programs and their objectives. A comprehensive reference list is included.

SYMBOLS

A	Area
c_p	Specific heat at constant pressure
D, d	Hole diameter
$\overline{e}$	Kinetic energy loss coefficient
K	Thermal conductivity
M	Blowing ratio = $\rho_c V_c / \rho_\infty V_\infty$
Nu_d	Nusselt number = hd/K
P_g	Total pressure at turbine inlet
Re_d	Reynolds number = $\rho dV/\mu$
St	Stanton number = $h/\rho_\infty V_\infty c_p$
T	Temperature
T_g	Total temperature at turbine inlet
V	Velocity
x	Distance in streamwise direction
y	Coolant fraction, ratio of coolant-to-primary air mass flows
α	Coolant hole injection angle relative to local tangent plane, see fig. 1
β	Coolant hole injection angle relative to streamwise flow, see fig. 1
η	Stage or vane thermodynamic efficiency, output power (or total kinetic energy) divided by total ideal power (or total ideal kinetic energy) of all flows involved, primary air plus coolant.
η_{FILM}	Film cooling effectiveness = $(T_{aw} - T_\infty)/(T_c - T_\infty)$
η_p	Vane primary efficiency, total kinetic energy output divided by ideal kinetic energy of primary air only.
μ	Molecular viscosity
ρ	Density

φ Temperature difference ratio = $(T_g - T_m)/(T_g - T_c)$

Subscripts:

∞ Free stream

aw Adiabatic wall

c Coolant

cr Flow conditions at Mach 1

g Gas stream

id Ideal or isentropic process

m Wall metal

0 Basic conditions of solid vane or blade

p Primary air

AERODYNAMIC STUDIES

This portion of the paper will give some of the pertinent results of aerodynamic studies conducted in-house at Lewis and under contract. The bulk of the testing was done in 2D and 3D cascades and at temperature ratios (primary air-to-coolant inlet) of unity (refs. 1 through 15). In addition to simplicity and cost it was felt that the first-order basic losses could be investigated cold and in simple cascades, and their characteristics at actual engine conditions predicted. Initial results of this concept of predicting engine conditions were studied in reference 16 using cold data from reference 17. The major results will be shown in this paper. In addition to cascade studies, a major effort under contract with General Electric at Evendale, Ohio was conducted using a single-stage turbine tested at actual engine temperature ratios with two sets of hardware resulting from different cooling designs (refs. 18 through 21). The performance of the solid blade turbine was determined in-house using a 1/2 size cold air model (ref. 22). Tip clearance studies were also made of this turbine and reported in reference 23.

CASCADE TESTS - Results of basic studies from 2D cold cascade tests will be highlighted. For all such tests using fixed geometry holes, coolant flow rate was varied by changing the inlet cavity pressure of the coolant relative to the primary air. Special emphasis will be placed on the condition when the inlet total pressure of the coolant is equal to the inlet total pressure of the primary air, because it represents a realistic condition relative to the ideal energies between the coolant and primary air. It will be referred to as "cavity pressure ratio of unity" when describing the figures.

Coolant Hole Angle Orientation - One of the major variables in both aerodynamic and heat transfer studies is the angle orientation of the coolant holes in the streamwise and spanwise directions. Figure 1 shows the angle defintions used throughout this paper. As indicated, α is the angle the hole makes relative to the tangent plane at the local surface, and β is the angle of the hole relative to the streamwise direction of the mainstream flow. Also shown are the nomenclature of the three streamwise angles investigated; in-line ($\beta = 0^{o}$), spanwise ($\beta = 90^{o}$), and compound (β between 0^{o} and 90^{o}).

Single and Multirow Coolant Ejection - The core vane model used for these studies is shown in figure 2. The photo at the top shows six rows of holes each on the pressure and suction surface of the vane. The diameter and pitch (spacing between holes in the same row) of the holes are 0.076 centimeter (0.030 in.) and 0.114 centimeter (0.045 in.), respectively (ref. 3). The tests were repeated in reference 4 using the same number of holes with half the diameter, or 0.038 cm (0.015 in.) and the same pitch. The results from the single-row ejection studies (refs. 3 and 4) indicated that only a small amount of the coolant energy ejected from the back portion of the suction surface (rows 10, 11, and 12) contributed to vane exit total energy compared to coolant ejected from other locations on the vane surface. For example, when the vane cavity pressure ratio was unity (equal ideal energies for primary air and coolant), only 10 to 50 percent of the ideal kinetic energy of the coolant ejected from the back portion of the suction surface was realized at the vane exit. On the other hand, coolant ejected from the pressure surface and forward portion of the suction surface (rows 1 through 9) realized up to 80 percent of its ideal kinetic energy at the vane exit.

In addition to single-row tests, multirow tests were conducted with various row combinations open on the pressure surface, suction surface, and both. Figure 3 shows the results when all 12 rows were open for both the multirow tests (solid curve) and predicted multirow results by adding single-row data (dotted curve). The results are shown as fractional change in primary efficiency relative to the uncooled solid vane. This term is a convenient measurement of the change in output kinetic energy caused by the coolant and is commonly used. The results shown in figure 3 are typical of all of the multirow tests and indicate that the coolant ejected from a given row does not influence the loss characteristics of coolant ejected from any other row. Although this vane had only 12 total rows of holes, the same observation was made for a full-film cooled vane with 45 rows of smaller holes designed for an actual core turbine (ref. 24). This vane was tested with various portions of the vane coolant holes sealed off.

Full-Film Cooled Vane - The full-film cooled vane with 45 rows of coolant holes referred to above was also tested with varying hole orientation (ref. 24). Cooling loss derivatives were obtained for various locations on the vane surface as a function of hole angle orientation and the results shown as figure 4. For all of the tests shown, the holes were inclined 35^{o} to the local surface ($\alpha = 35^{o}$). The loss derivatives shown represent the percent penalty in "thermodynamic" efficiency per percent coolant when the cavity pressure ratio was unity for each case. The appropriate cross-hatchings on the vane sketch and the bar graphs indicate the three areas of the vane selected for comparison.

Figure 4 indicates that coolant flow losses from the pressure surface and the forward portion of the suction surface are fairly insensitive to hole orientation angle. They vary from about 0.2 percent penalty for each percent coolant flow from the forward part of the suction surface to about 0.7 penalty from the pressure surface. However, coolant flow from the back portion of the suction surface both produces high losses and is very sensitive to hole angle. For example, each percent coolant flow from in-line holes in this region results in a loss in efficiency of about 0.9 (utilizes about 10 percent of available coolant kinetic energy). This loss increases to about 2.1 percent reduction for each percent coolant for spanwise holes. Compound angle holes are only somewhat worse than in-line holes, or about 1.1 percent loss per percent coolant. This suggests a compromise between aerodynamic and heat transfer considerations. Although best overall from aerodynamics, in-line holes can result in ineffective film cooling (ref. 60). Compound angle holes, on the other hand, can result in coolant vortices, effective film coverage and only a nominal penalty difference in losses. Particularly for the back portion of the suction surface, then, compound angle holes may be a desirable compromise. For any other portion of the vane, figure 4 indicates that hole alignment can probably be dictated from heat transfer considerations.

Varying Primary-to-Coolant Temperature Ratio - As indicated previously, it was felt that first order basic losses could be studied cold and the result of actual engine temperature ratio (primary-to-coolant) predicted from cold data. Two vanes were tested in-house at varying temperature ratio (ref. 17) and were used as the basis of such a prediction method in reference 16. A sketch of the vanes is shown in figure 5. The upper vane had a combination of convection cooling with trailing edge ejection and a single film-cooling slot on both the pressure and suction surface. The vane on the bottom was cooled by transpiration and had a wire mesh shell welded to an internal strut. The cold test results are shown in terms of efficiency in figure 6 at an indicated temperature ratio of one (solid circles). As indicated on the right side of the figure, the coolant fraction was varied by increasing the cavity pressure ratio from 1.0 to 1.5. As seen on the ordinate, this resulted in higher coolant flows and lower thermodynamic efficiency for both vanes. This cold data was then used in reference 16 to predict performance at higher temperature ratios. The basic assumption used in the analysis is that the aerodynamic losses (total pressure losses) due to boundary layer growth are sufficiently modeled if the coolant-to-primary momentum ratios are maintained constant. This occurs when the coolant-to-primary inlet total pressure ratio (or cavity pressure ratio) is maintained constant as shown by the solid lines of figure 6. The experimental data at higher temperature ratios are shown for comparison in figure 6 as open circles. Very good agreement was obtained for the convection-film vane and fair agreement for the transpiration vane. Although such 2D test results tend to validate the prediction of profile losses at actual engine temperature ratios, preliminary data from annular tests with endwall cooling and secondary flows are not as definitive. More testing in this area is needed.

Effect of Ceramic Coating on Vane Efficiency - There is current interest in the use of thermal barrier coatings at intermediate temperature levels from cost, life, and performance considerations. Figure 7 shows a ceramic coated vane used to evaluate the effect of the coating on vane loss (ref. 11). In the as-sprayed condition, the surface roughness averaged 8.9 micrometers. Light polishing with a solid piece of aluminum oxide reduced the roughness to an average of 2.8 micrometers. Exit surveys were made behind the vane both in the rough and smooth condition in a cold 2D cascade. The resulting loss coefficients as a function of exit velocity ratio are shown in figure 8. At design exit velocity ratio of about

0.8, the additional loss for the rough coating over the uncoated vane is very large, about 4 points. Light polishing eliminated most of this additional loss. Figure 8 shows the polished vane to have a loss about 0.7 point greater than the uncoated vane, and is directly attributable to the 38 percent thicker trailing edge of the polished vane.

ROTATING STAGE TESTS - An experimental investigation of a 50.8-cm (20-in.) diameter core turbine was conducted under contract by the General Electric Company in Evendale, Ohio (refs. 18 through 21). Three turbines were tested; a solid vane-solid blade combination, and two cooled versions using the same vane and blade profiles as the solid turbine but with different coolant hole designs. Stage tests were made at primary-to-coolant inlet temperature ratios up to those encountered by hot engine conditions. In addition to overall performance ofthe cooled turbines, vane and blade loss derivatives were obtained by testing a combination of solid and cooled vanes and blades and will be described.

Description of Turbines - A photograph comparing the three blade versions tested is shown as figure 9. Cooled turbine #1 had in-line holes aligned about 35^{o} to the local surface. As seen in figure 9, turbine #2 had much fewer, bigger holes. Turbine #2 had a total of about 1/3 as many holes as turbine #1. The design temperature level was high enough to require full-film cooling combined with internal impingement cooling. A sketch of the flow path showing the five individually controlled cooling circuits is shown as figure 10. The vane and blade passage heights were both 3.81 cm (1.5 in.), and the vane and blade axial chords were 3.81 cm (1.5 in.) and 3.43 cm (1.35 in.), respectively. The turbine test facility is shown in figure 11. Power was absorbed and speed controlled by a waterbrake on the far side of the large exhaust collector shown in the photograph. The turbines were both tested at a reduced (from engine conditions) inlet gas temperature of

778 K (1400° R). Coolant air was temperature controlled to set actual engine primary-to-coolant inlet temperature ratio.

Test Results - The overall turbine test results are tabulated in figure 12. The solid efficiency of 0.88 was calculated by NASA from the solid half-size turbine results from reference 23 corrected for scaling effects and clearance. A significant penalty in thermodynamic efficiency is noted from the figure. The overall loss derivative is the penalty in efficiency expressed in percent decrease caused by the coolant per percent coolant required. Each percent of coolant required resulted in a penalty of about 0.5 and 0.4 percent in efficiency for turbines #1 and #2, respectively.

In addition to the overall loss, an individual loss study was made of turbine #2 to separately determine the derivative losses of the vanes and blades. The combination of solid and cooled vanes and blades tested are tabulated in figure 13. For each combination shown, coolant was ejected from the vane inner and outer walls, and from the static shroud over the rotor blades (see fig. 10). The results of these tests are shown in figure 14. Of the total of 17 percent coolant used to cool turbine #2, figure 14 indicates that the blades and vanes used 8 and 4.3 percent, respectively. The loss derivatives are shown to be 0.56 and 0.32 percent reduction in stage efficiency for each percent coolant required by the blades and vanes, respectively. Using the coolant flows shown, this represents a four-point penalty in efficiency caused by blade coolant and about a one-point penalty caused by the vane coolant. Unfortunately, the turbine was not tested with solid vane endwalls to determine their individual loss derivatives. Unpublished in-house data collected from 3D tests indicate that vane endwall loss derivatives can be as high or higher than vane losses. More testing of vane and endwall cooling is required to define these losses for this turbine configuration.

Although transpiration cooling of a surface using a porous wall consisting of a multitude of closely spaced minutely small holes is more effective (ref. 26) than full coverage film cooling that utilizes arrays of discrete holes in the surface, primary emphasis by NASA and industry has been on the latter. The reasons for this are that the relatively low structural strengths of the porous walls, potential oxidation of the materials, and the plugging of the very small holes have precluded, as yet, their use in operational cooled turbines. Although NASA has conducted and supported research on transpiration-cooling, particularly in areas related to the fabrication, oxidation, and flow characteristics of wall materials (refs. 27 to 32), the studies to be discussed in this paper are primarily concerned with local and full coverage film cooling using discrete hole arrays in (a) flat plates and (b) cascades and engines.

Other recent studies conducted by NASA in-house and under contract in areas related to turbine cooling and aerodynamics which will not be discussed further in this report but which are mentioned for completeness are: references 33 to 35 which investigated pressure loss and flow characteristics for individual coolant flow passages and for complete airfoils, reference 36 on thermodynamic and transport properties of combustion products of ASTM-A-1 fuel and air, references 37 and 38 on blade fabrication processes, references 39 and 40 on stress analysis of blades, references 41 to 43 on design of more reliable turbine disks, references 44 to 47 investigated impingement cooling primarily applicable to leading edge region, reference 48 examined the boundary conditions of porous or perforated walls, and lastly, references 49 to 54 deal with instrumentation required in turbine cooling research.

FLAT PLATE HEAT TRANSFER INVESTIGATIONS - The early research on discrete hole film cooling was conducted by the University of Minnesota under contract to NASA. This effort was concerned with obtaining an understanding of the cooling characteristics of the flow out of single holes and rows of holes normal to and inclined at several angles with the wall and the mainstream flow. The research was conducted on a flat plate in a tunnel where adiabatic wall temperatures were measured downstream of ejection holes. The results of these investigations (refs. 55 to 57) showed, as illustrated in figure 15, that for a constant blowing ratio M of 0.5, injection at a shallow angle (35°) to the wall in-line with the direction of the mainstream gave high local film cooling effectiveness, η_{FILM} which decays rather slowly with distance. Injection of the coolant in the spanwise direction (normal to the mainstream) as might be expected provided cooling of a larger surface area but at the expense of lower cooling effectiveness and a more rapid decay of cooling with distance. The injection at shallower angles in the spanwise direction, as can be seen on comparing figures 15(b) and (c), gave not only better coverage but slightly better cooling which persisted longer. It would appear from the data that injection at an angle between the streamwise and spanwise direction to be the best compromise for good cooling and coverage.

The effect of film cooling blowing ratio and interaction of adjacent holes is shown in figure 16. The figure shows the cooling effectiveness measured downstream of a single hole with that of a row of holes spaced at 3-diameters with in-line injection 35° to the wall. The effectivenesses, measured along the hole center lines, show little effect of the presence of adjacent holes until beyond 11 diameters downstream or at high blowing rates (above 1.0). More importantly, the tests in figure 16 showed that ejection from discrete holes resulted in a maximum cooling effectiveness when a blowing ratio M of about 0.5 was reached. The studies at the University of Minnesota also provided information of the developing temperature field, the velocity profiles, and the turbulence levels in the boundary layer as it developed downstream of the injected coolant (ref. 58). Results showed that the injected fluid caused intense turbulence in the boundary layer just downstream of the holes. Models for the analysis of the temperature field were also developed and are described in reference 59.

More recently, research was and currently is being conducted at Stanford University under contract to NASA to further explore discrete hole film cooling with emphasis on multiple rows of holes and the region downstream of these holes. The approach considered full coverage film cooling as a special case of transpiration cooling. Measured in the study was the average heat flux downstream of rows of holes. The heat transfer coefficients and Stanton numbers were determined from these heat fluxes. The investigation used a flat plate with 11 rows of holes in which ejection was normal to the wall or at 30° to the wall in the direction of the gas stream. Two spacing of holes were investigated, holes spaced at 5 diameters and holes spaced at 10 diameters. Results of these studies (refs. 60 to 62) indicated that the presence of the rows of film cooling holes without blowing increased the Stanton number along the plate compared to that theoretically predicted for a smooth flat plate. The results also showed that in-line injection of air at 30° to the wall at the temperature of the wall at a blowing ratio of 0.4 reduced the heat transfer coefficient but that as the blowing ratio increased to 0.7 the heat transfer coefficient increased. This poorer cooling with a high blowing ratio is similar to that shown in figure 16. The results from the Stanford University tests also indicated that at a constant blowing ratio, inferior cooling (higher Stanton number) was obtained with the large hole spacing (10-hole diameter) compared to the smaller hole spacing (5-hole diameter). This inferior cooling was also obtained when the two configurations of hole spacings had the same blowing ratio for a given wall area. The resulting higher velocity from the fewer holes at the large spacing for this blowing condition increased the turbulence and also penetrated further into the boundary layer and resulted in less cooling of the wall.

Integral equations were developed for the prediction of the heat transfer data for cases when the injected fluid is at mainstream or at wall temperature. Also developed was a two-dimensional boundary layer program (ref. 63) which was modified as described in reference 62 to predict the thermal performance of full-coverage film cooled walls. A finite difference scheme was used. It was based on the computer program originated by Patankar and Spalding (ref. 64). The results of using the program show that good agreement with experiments have been obtained. The solutions obtained are dependent on obtaining empirical constants for coolant penetration and augmentation of turbulent mixing. As a consequence, research is continuing to determine the general applicability or changes required in the constants for various injection angles and hole geometries and spacings.

Although these investigations have contributed data to develop analytical models for film cooling, a better understanding was still needed of the fluid dynamics under conditions of injection of a film through holes into a gas stream. A method undertaken to achieve this understanding was flow visualization. A carbon dioxide-water vapor fog was used at the University of Minnesota, reference 58, to visualize the flow field and turbulence associated with injection of air through a single hole. Although some insight was obtained from the study, the fog diffused so rapidly due to the high turbulent mixing in and near the injection region that only the large-scale turbulent motion near the hole was visible. Another study, conducted at the Lewis Research Center and reported in references 65 and 66, utilized air seeded with small neutrally buoyant helium-filled bubbles injected into a turbulent boundary layer through discrete holes in a flat plate test tunnel. The paths traced by the bubbles map streakline patterns of the injected film air mixing with the mainstream. Unlike fog or smoke which

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diffuses rapidly, the bubble movements are clearly identifiable as continuous thread-like streaks in the injected film layer. Visualization studies were conducted of injection normal to the wall, slanted at 30° to the wall in the direction of the mainstream, and at a compound angle: 30° to the wall and 45° to the mainstream flow. Observance of the bubble streaklines for normal injection showed a high intensity small-scale turbulence in a vortex produced downstream of the injection hole. This high turbulence is detrimental in that it causes extensive mixing of the film with the mainstream resulting in increases in the heat-transfer coefficient and aerodynamic losses. For normal injection the film was seen to detach from the surface at blowing rates as low as 0.3. This would indicate a reduction of the cooling effects of the film. The streaklines obtained with injection slanted 30° in the direction of the mainstream, shown in figure 17, have a much lower intensity of turbulence than normal injection. Also, the streaklines show little spreading of the jet as it progresses downstream. The figure also shows that the film remains very close to the wall at a blowing ratio of 0.3. It was not until the blowing exceeded 0.5 that the film began to detach from the surface and allow the boundary layer to wrap around beneath the jets. The maximizing and reduction of cooling effectiveness above a blowing ratio of 0.5 previously shown in figure 16 is substantiated and explained by the detachment of the film. The detachment of the film is clearly shown in figure 17, at a blowing ratio of 0.8, and a penetration into the free stream at a high blowing ratio of 1.4. The boundary-layer thickness, δ, is indicated in the figure by the arrow.

The results of the flow visualization of a film injected at a compound angle are shown in figure 18. The oblique angle that the film makes with the mainstream generates a single vortex filament downstream of each hole. This vortex begins forming at blowing rates of about 0.3 (fig. 18(a)) and becomes most pronounced at blowing ratios between 0.7 and

0.9. Notice the very tight "winding" of the streaklines in figure 18(b) at a blowing ratio of 0.75. A closeup top view of the upstream injection region for this blowing ratio is shown in figure 18(c). The most important feature of compound angle injection is that this strong vortex motion keeps the film attached to the surface even at high blowing ratios resulting in lower wall temperatures. Observation of the streaklines with compound angle injection from a side view of the tunnel indicated little difference in the penetration distance for blowing ratios between 0.3 and 0.9 (fig. 19(a)) compared with the increase in penetration observed for in-line injection when the blowing ratio is increased from 0.3 to 0.8 (fig. 19(b)). The cooling characteristics of compound angle injection was examined in a turbine vane cascade discussed in the following section.

CASCADE AND ENGINE INVESTIGATIONS - A large number of tests were made at NASA to evaluate the thermal performances of various configurations of air-cooled vanes and blades (refs. 67 to 91). These tests were conducted in an annular sector cascade consisting of four turbine vanes and in a research turbojet engine modified to accept cooled test vanes and blades. The facilities, which operated to gas temperature of 1644 K, are described in detail in reference 79. The effects of impingement cooling on the inner surface, film cooling on the outer surface, combinations of film and impingement cooling, and the effects of thermal barrier coatings were investigated.

The data in both references 73 and 74 show that the effect of crossflow over impinging jets is to enhance the downstream heat transfer rather than degrade the heat transfer as has been reported by other investigators. Figure 20 from reference 74 shows that for a given jet Reynolds number, Re_d, the local Nusselt number, Nu_d, increases as the crossflow increases cumulatively along the vane periphery from thermocouple locations 10 to 12.

During the evaluation of the vanes and blades which utilized local film cooling (refs. 82 to 87) adverse effects of film cooling were observed. It was found that injection of the film can increase the heat-transfer coefficient by increasing the turbulence or tripping a laminar boundary layer. This adverse effect at low coolant flow resulted in higher metal temperature than would exist if no film cooling holes were used. This is illustrated in figure 21, obtained from reference 83. A measurement at a single thermocouple location is illustrated although other measurements upstream of the location gave similar results. The figure shows that when film was initially injected the temperature difference ratio φ decreased (or the measured metal temperature increased). It was only after the film coolant to gas flow ratio was increased to beyond 0.01 that the metal temperature was restored to the level before film injection.

The influence of injection angle on the cooling of the suction surface of a vane was investigated in reference 87. The study was undertaken to determine whether coolant injected at a compound angle would provide better cooling with increases in blowing ratio than in-line injection. This investigation was intended to also verify the expected benefits of better flow attachment to the wall with the compound angle injection as seen in the flow visualization studies described earlier (ref. 65). The results of the cascade test are shown in figure 22 for the suction side of a vane where an adverse pressure gradient existed. Compound angle injection resulted in a continual improvement in cooling of the wall with increases in blowing ratio. This is seen as the continual increase in the temperature difference ratio. On the other hand, as expected, the slanted in-line injection peaked in cooling at a blowing ratio of about 0.5, after which the wall temperature increased with additional coolant injection. The importance of combining convection cooling with film cooling was experimentally verified in reference 86 and was analytically predicted in reference 88. Figure 23, from reference 88, shows that for a constant total coolant flow the outer surface of a wall which combines film and convection cooling has a lower average temperature than with either film cooling alone or convection cooling alone. Analysis supported by experimental data indicated that at typical engine conditions and coolant-to-gas flow ratios (about 0.05) combinations of external film and internal convection cooling can result in lower metal temperatures than either method alone.

Full coverage film cooling with effective internal cooling, such as impingement cooling, provides the means (ref. 26) to attain the higher gas temperature and pressure expected of turbines of the future without excessive coolant flow requirements. The fabrication of these airfoil, however, is both complex and costly, and the ejection of coolant over the blade surface has the built-in aerodynamic penalties described earlier. A potential alternate method not only for use on future turbines but to provide benefits to current ones is the use of a ceramic thermal barrier coating over the outer surface of the airfoil. Such a coating shown on a blade in figure 24 has been investigated and is under further development at the Lewis Research Center (refs. 89 to 91). The coating as shown in the figure consists of a thin nickel alloy bond coat of about 0.01 centimeter covered with zirconia with a thickness of about 0.038 centimeter. The coating is applied by plasma spraying. The details of the coating and application process are described in reference 91. The coating which serves as an insulation to reduce heat flux to the blade was found to have a potential for significantly reducing blade metal temperatures and coolant flow requirements.

Predicted reductions in bulk turbine-vane metal temperatures and coolant-to-gas flow ratios with increases in ceramic coating thickness on vanes in an advanced core engine turbine as obtained from reference 89 are shown in figure 25. Bulk wall metal temperature (integrated average temperature over entire vane) was substantially reduced as ceramic coating thickness was increased. The bulk wall metal temperature for the impingement-cooled turbine vane analyzed could be reduced by as much as 390 K at a coolant-to-gas flow ratio of 0.10 when coated with a 0.051-centimeter thickness of zirconia. Alternatively, when both coolant flow and wall metal temperature were allowed to vary, large changes in both metal temperature and coolant flow were predicted. Vanes coated with a 0.051-centimeter thickness of zirconia could have both an eight-fold decrease in coolant flow and 110 K reduction in metal temperature compared to the uncoated vane. The coolant flow was reduced from 0.10 for an uncoated vane to 0.0125 for a coated vane with a corresponding vane metal temperature reduction from 1390 to 1280 K, respectively.

The dashed portions of the curves in the figure illustrate a limitation associated with using the current ceramic composite coatings. The limitation is the ability of the ceramic coating to adhere to the bond coating when the temperature at the interface between these two layers exceeds 1367 K. This limiting temperature was determined with furnace tests described in reference 89.

Experimental evidence of the potential reductions in metal temperature was obtained with a coated and uncoated vane operating in a research engine. These results shown in figure 26 show approximately a 190 K reduction in the metal temperature of the vane coated with a 0.028-centimeter thickness of zirconia compared to an uncoated vane. The figure also shows good agreement between calculated and measured wall temperatures.

Another calculation of the coolant requirements of an advanced convection-cooled turbine, a full-coverage film-cooled turbine and a thermal barrier coated convection-cooled turbine is shown in figure 27. The comparison shows that the coated turbine has the potential for operation at coolant flows similar to one that is full-coverage film cooled. A

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ceramic thickness of 0.038 centimeter was assumed in the analysis.

The durability of various coatings on air-cooled vanes and blades was investigated in cyclic tests in furnaces and burner rigs and under both steady state and transient engine operations. These tests are described in reference 91. The coatings on one group of blades successfully completed as many as 150 hours in steady state engine operation at gas temperatures as high as 1644 K and on another group of blades completing 500 2-minute cycles between full power (at a gas temperature of 1644 K) and flame-out (at a gas temperature of 1000 K). Based on material and processing costs and the results of the cyclic engine tests, yttria-stabilized zirconia was considered the best of the ceramic coatings investigated.

SUMMARY OF MAJOR RESULTS

The Lewis Research Center has a continuing turbine program directed toward the use of higher temperatures with good efficiency and acceptable vane and blade life. Several results of various efforts within this program are general in nature and can be stated rather briefly.

1. The aerodynamic penalty associated with film coolant ejection varies greatly with location on the vane profile. Air ejected from the back part of the suction surface realized only 10 to 50 percent of its ideal kinetic energy, based on vane exit pressure, whereas that ejected from the forward part of the suction surface and all of the pressure surface delivered up to 80 percent of its available energy.

2. Aerodynamic effects of coolant ejection through successive rows of holes were neither compounded nor diminished through interaction. Coolant flow quantities and losses were simply additive.

3. The angle at which coolant is ejected influences both cooling effectiveness and aerodynamic loss. Blade surface regions with decelerating flow require cooling-aero compromises, and good cooling

effectiveness and relatively low loss are best obtained when the coolant leaves the blade through a compound angle with a large chordwise velocity component and a somewhat smaller spanwise component. Leading edges and areas of accelerating flow are aerodynamically less sensitive, so greater freedom may be exercised in the selection of coolant ejection angle.

4. Thermal barrier ceramic coatings approximately 0.4 millimeter thick have shown durability through steady state and cyclic engine operation with internal convective blade and vane cooling. The thermal performance of a ceramic coating on internally convectively cooled vanes and blades is potentially as effective as full-coverage film cooling. The aerodynamic loss of a coated vane is no larger than that of a smooth metal vane with the same trailing edge thickness.

5. Two-dimensional vane studies have shown that profile losses at actual engine temperature ratios may be reasonably predicted from cold test data obtained with equal primary and coolant temperatures.

6. Thermodynamic efficiencies of advanced high temperature core turbines can be severely affected by full-coverage film cooling. Turbine loss derivatives for two core turbine cooling configurations ranged from 0.4 to 0.5 percent reduction in turbine efficiency for each percent of coolant. The blade profile loss derivatives were approximately double those of the vane profile, and current in-house studies indicate that vane endwall loss derivatives are as high or higher than the vane profile derivatives.

7. Film cooling can cause an increase in the external heat transfer to a vane or blade which has a fixed internal convective cooling flow. This results in higher metal temperatures. The mechanism involves an increase in turbulence and in some cases the tripping of a laminar boundary layer.

8. At typical engine conditions and coolant flow rates combinations of external film and internal convection cooling can result in lower average blade

metal temperatures than either method alone.

9. For given hot side flow conditions and total coolant flow, combined impingement and crossflow on the cool side provided significantly lower metal temperatures than did impingement alone.

CURRENT PROGRAMS

Turbine cooling studies now in progress include a number of contracts and grants as well as in-house work in both experimental and analytical areas. Film cooling, impingement, endwalls, thermal barrier coatings, and even liquid cooling are receiving attention.

FILM COOLING - Film cooling is the most important area because at present it is the only demonstrated cooling method that can protect vanes and blades from the high gas temperatures, 1530 to 1900 K, current and planned for the advanced civil and military aircraft. A warm core turbine facility is planned for operation in late 1977. This facility, shown schematically in figure 28, with inlet temperatures up to 870 K will permit aerodynamic studies of cooled turbines with primary-to-coolant temperature ratios from one to three. The warm core turbine investigation described previously will be continued to further define losses associated with the various coolant flow paths. Also, the test equipment will serve as a vehicle to evalute modifications to endwall contours, clearance geometry and coolant ejection schemes. This facility will also be used to evaluate a turbine designed for the requirement of the new Energy Efficient Engine. A companion warm core cascade rig is nearing completion. Full sets of vanes will be studied in detail in the cascade rig prior to operation with a rotor in the turbine rig.

A hot turbine facility is also under construction and will be operational in 1978. This facility will provide pressures up to 40 atmospheres and temperatures up to 2480 K. Figure 29 shows some of the facility's features. Vane, blade, and wall temperatures with advanced cooling methods will be measured during turbine operation at design conditions with its very high heat fluxes. This facility and its full capabilities are described in detail in reference 92.

Other film-cooling programs involve one-vane tunnel hot tests, two-dimensional vane cascade tests, flat plate heat-transfer measurements, and flow visualization with neutrally buoyant helium bubbles.

ENDWALL COOLING - A contract effort at United Technology Research Center is developing a three-dimensional turbulent shear flow computer program. This program will calculate heat-transfer coefficients and viscous losses with coolant injection through a turbulent boundary layer.

At the Lewis Research Center an annular cascade is being used to investigate the influence of coolant injection on the secondary flows and loss accumulations in the blade wakes.

IMPINGEMENT COOLING - The variable that determine heat-transfer coefficients in impingement with crossflow are being examined systematically at Arizona State University under a NASA Grant. Related in-house work is underway at Lewis in cascades and planned for a flat-plate heat-transfer tunnel and a flow-visualization tunnel.

THERMAL BARRIER COATINGS - Work to date with the stabilized coatings has been sufficiently promising that a substantial experimental program is in progress. Thermal properties, resistance to erosion and foreign object damage, and life properties will be examined in detail. Tests planned include hot cascade, furnace, and extensive engine operations with high temperatures and large numbers of thermal cycles.

A turbine designed for the energy efficient engine requirements but employing the thermal barrier coating instead of film cooling will be tested in the Warm Core Turbine Facility. Aerodynamic performance of this turbine will be compared with the film-cooled version for an accurate assessment of the potential performance gain offered by the coating. Correlation based on previous cascade and turbine tests indicate a potential of three points in first-stage thermodynamic efficiency.

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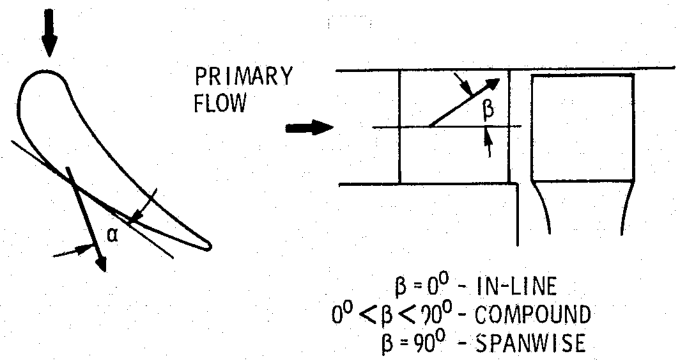

Figure 1. - Coolant hole angle orientation.

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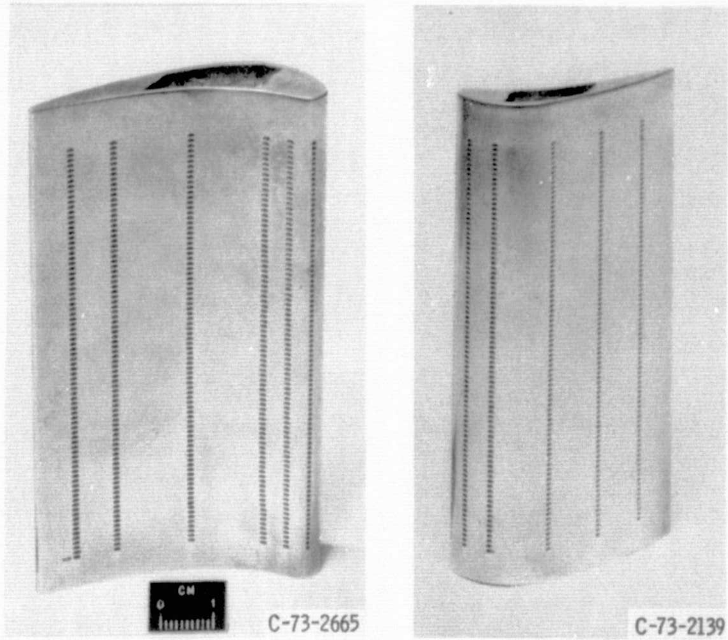
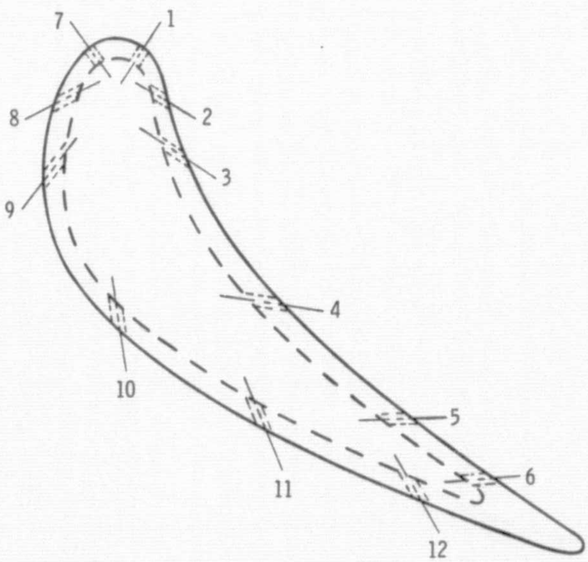


PRESSURE-SURFACE VIEW

SUCTION-SURFACE VIEW

(a) TESTED STATOR BLADE.



(b) CROSS-SECTIONAL SKETCH OF COOLED STATOR BLADE.

Figure 2. - Single and multirow ejection.

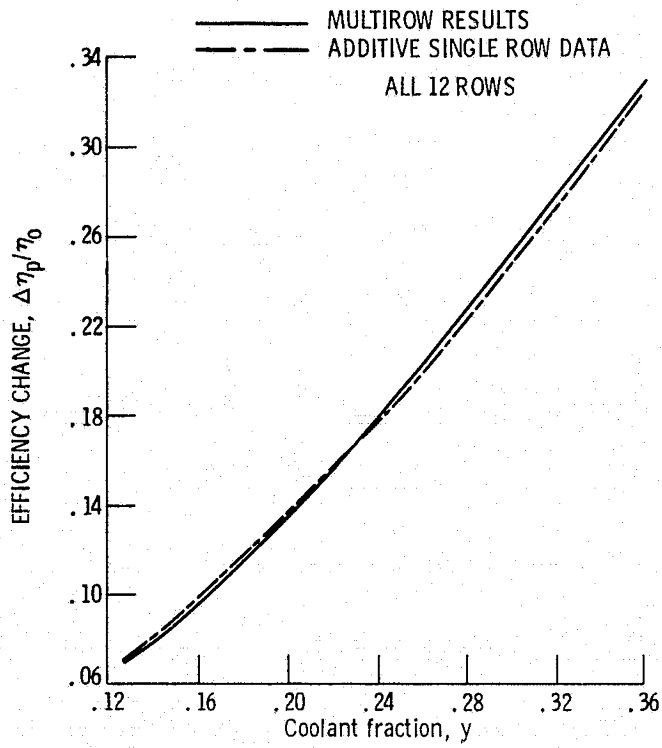


Figure 3. - Multirow vs. additive single row ejection.

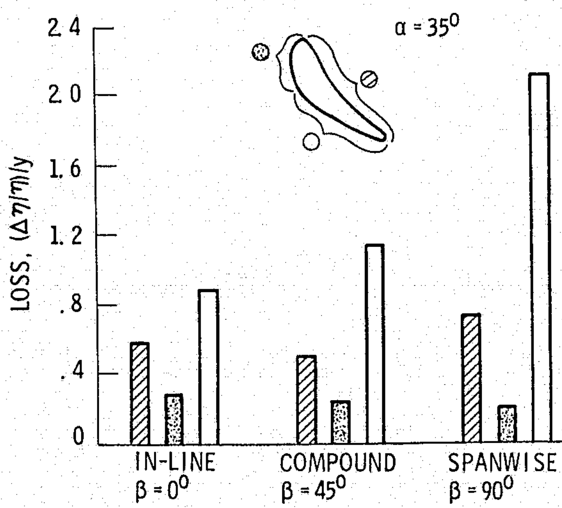


Figure 4. - Core vane loss derivatives.

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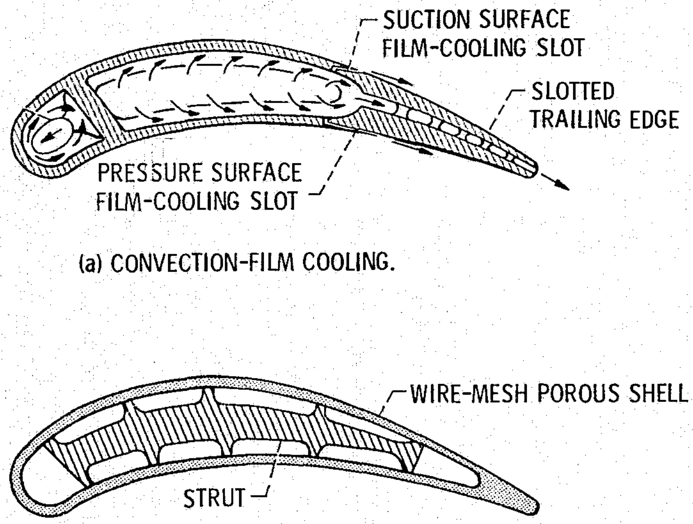
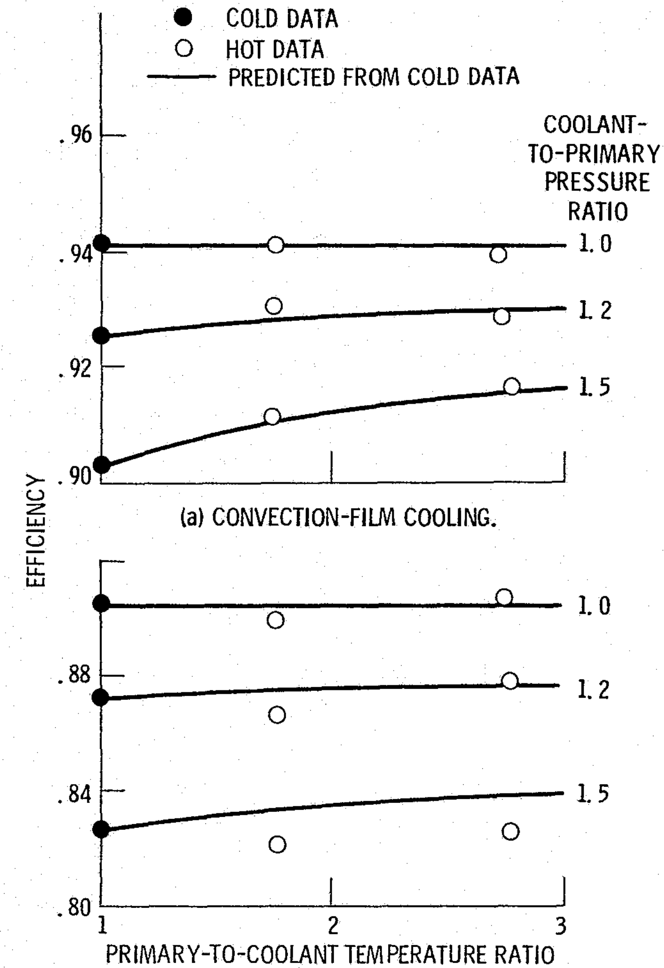


(a) CONVECTION-FILM COOLING.

(b) TRANSPIRATION COOLING.

Figure 5. - Vanes tested at varying primary-to-coolant temperature ratio.



(b) TRANSPIRATION COOLING.

Figure 6. - Use of cold data to predict hot results.

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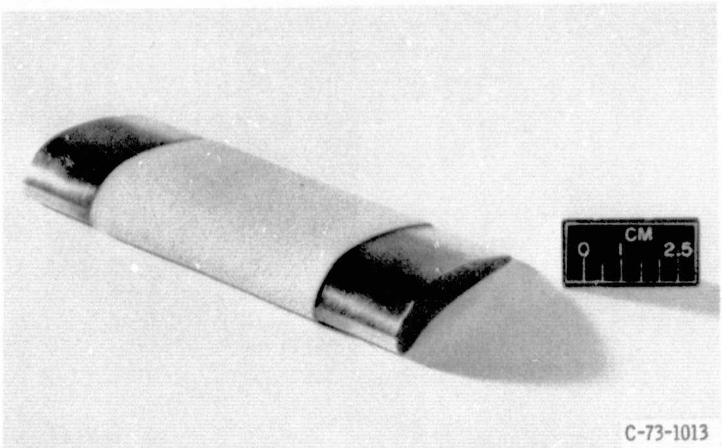


Figure 7. - Ceramic coated core test vane.

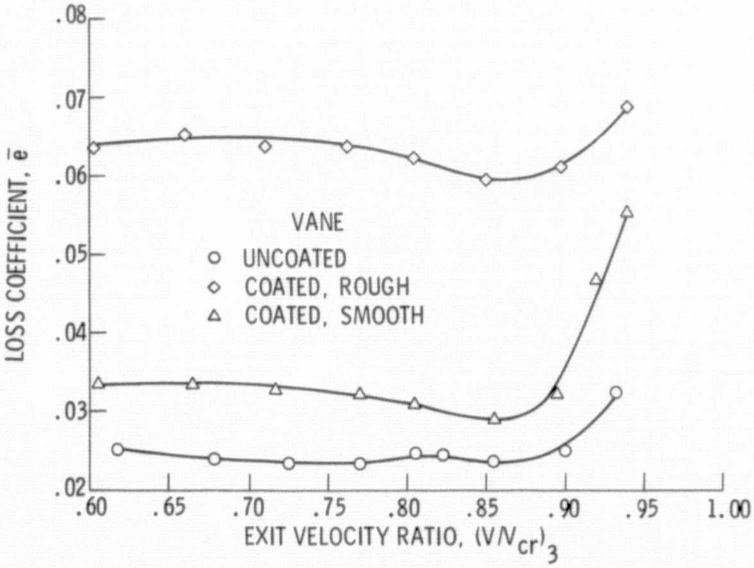
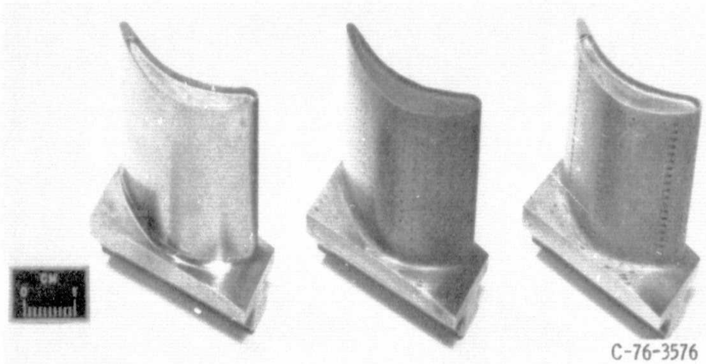
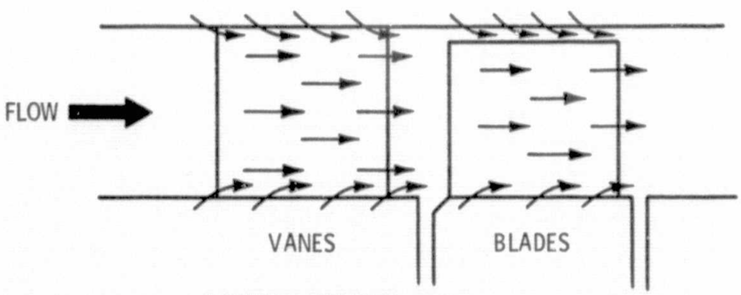


Figure 8. - Effect of ceramic coating on loss.



SOLID TURBINE NO. 1 TURBINE NO. 2

Figure 9. - Core turbine blades tested.



COOLING CIRCUITS

VANE
VANE OUTER WALL
VANE INNER WALL
BLADE & PLATFORM
STATIC SHROUD

Figure 10. - 20 Inch core turbine cooling circuits.

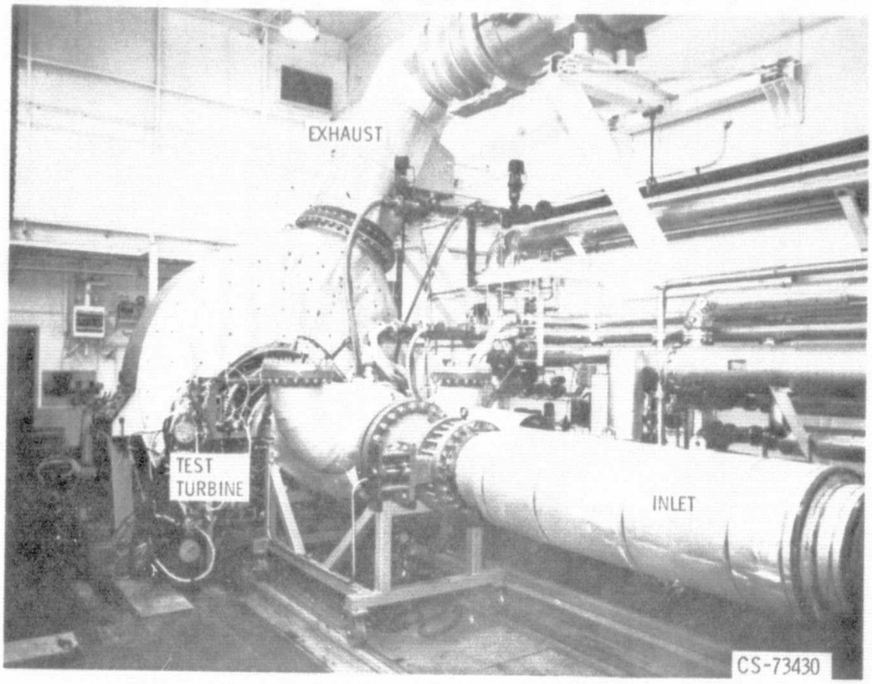


Figure 11. - Core turbine test facility.

	TURBINE #1	TURBINE #2
SOLID EFFICIENCY, η_o	0.880	0.880
COOLED EFFICIENCY, η_c	0.806	0.821
COOLANT FRACTION, y	0.178	0.170
STAGE LOSS DERIVATIVE $\left[\frac{(\Delta\eta/\eta)}{y}\right]_{STAGE}$	0.47	0.39

Figure 12. - Overall turbine test results.

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TURBINE CONFIGURATION

	#1	#2	#3	#4
VANES	SOLID	SOLID	COOLED	COOLED
BLADES	SOLID	COOLED	SOLID	COOLED

Figure 13. - Individual loss study.

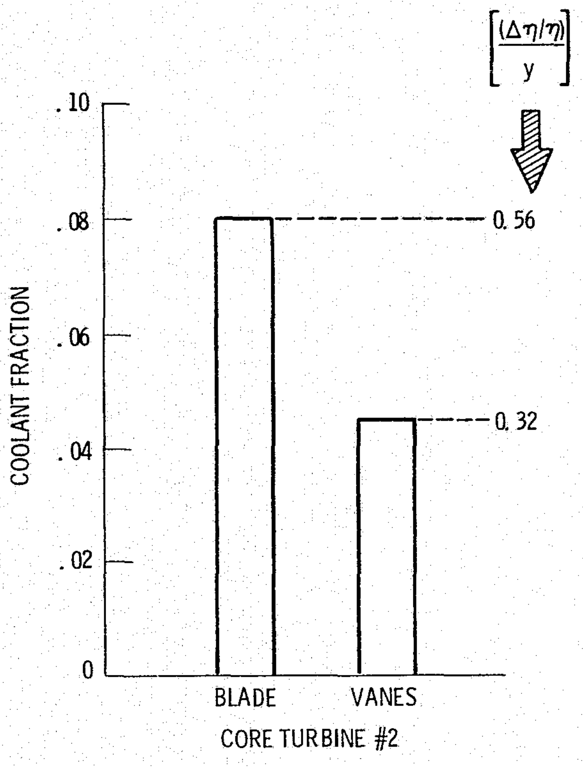

Figure 14. - Individual loss derivatives.

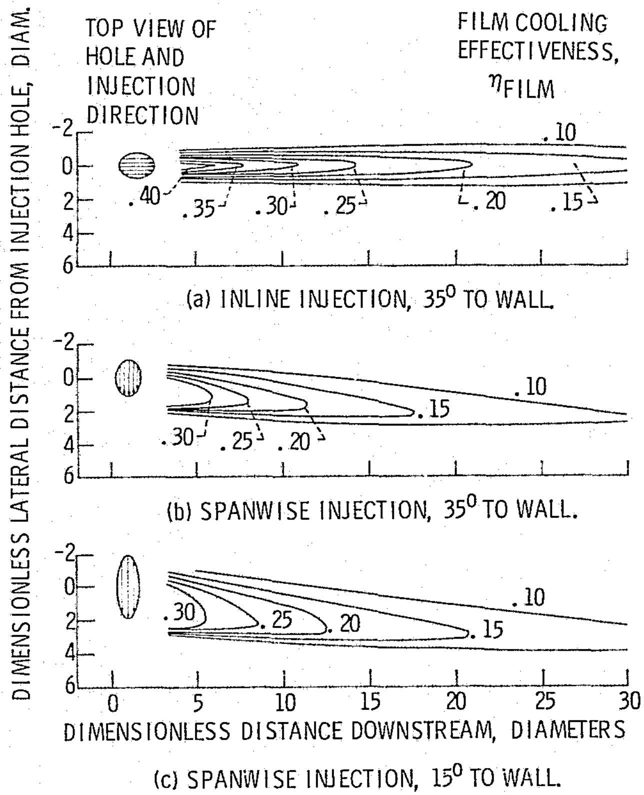

Figure 15. - Effect of injection angle on film effectiveness. (Blowing ratio, M = 0.5.)

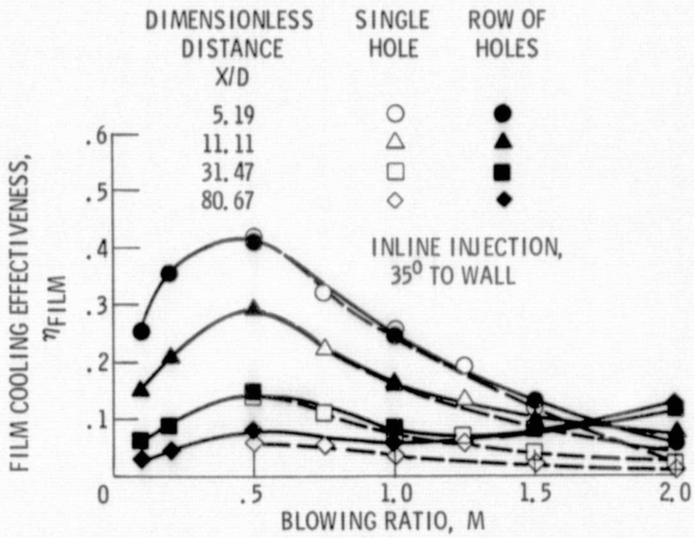

Figure 16. - Effect of blowing ratio and adjacent hole interaction on film effectiveness.

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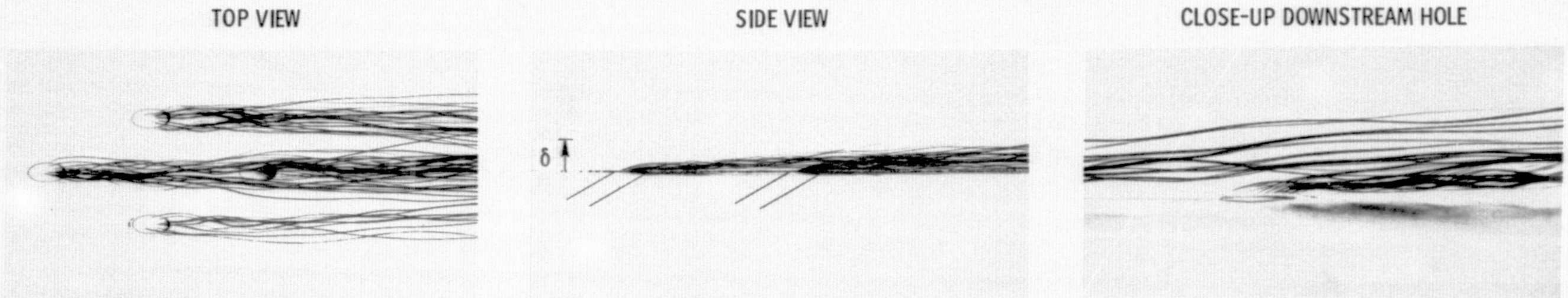
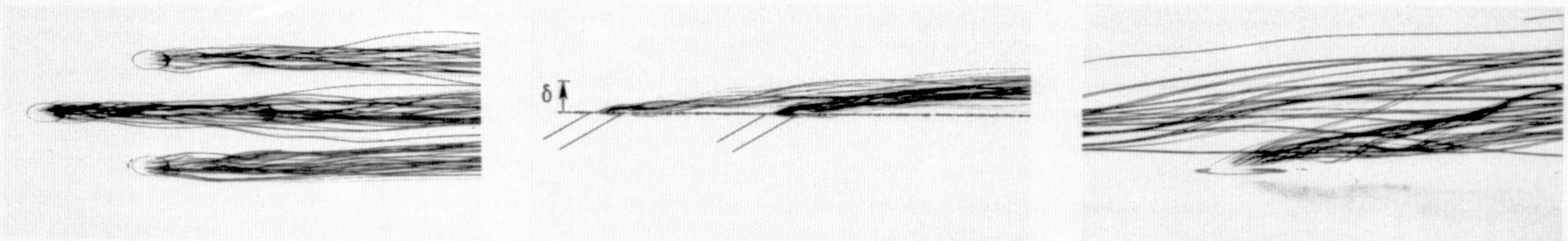
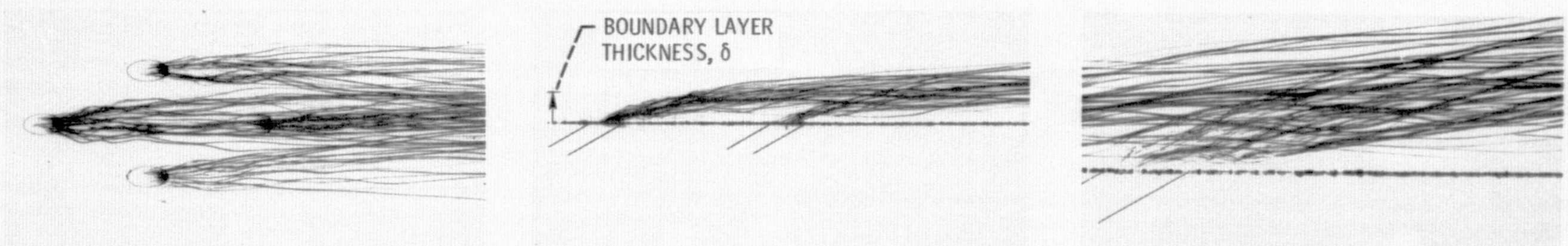


(a) BLOWING RATIO, M = 0.3.

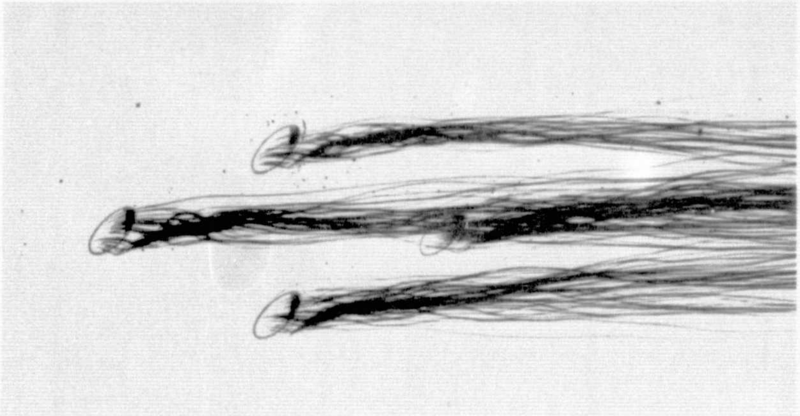


(b) BLOWING RATIO, M = 0.8.

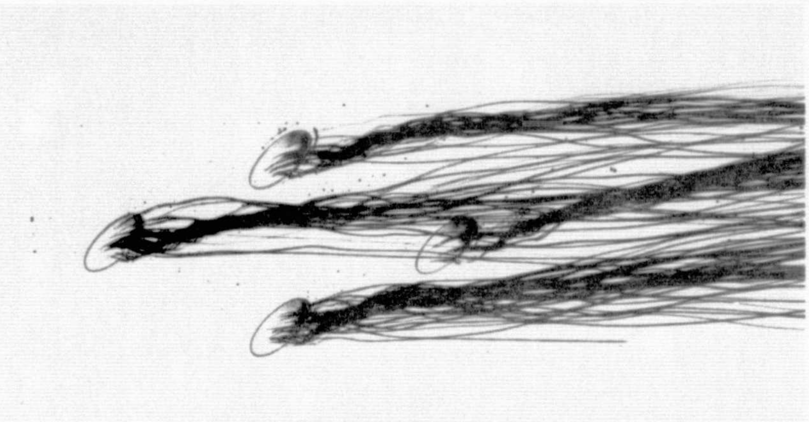


(c) BLOWING RATIO, M = 1.4.

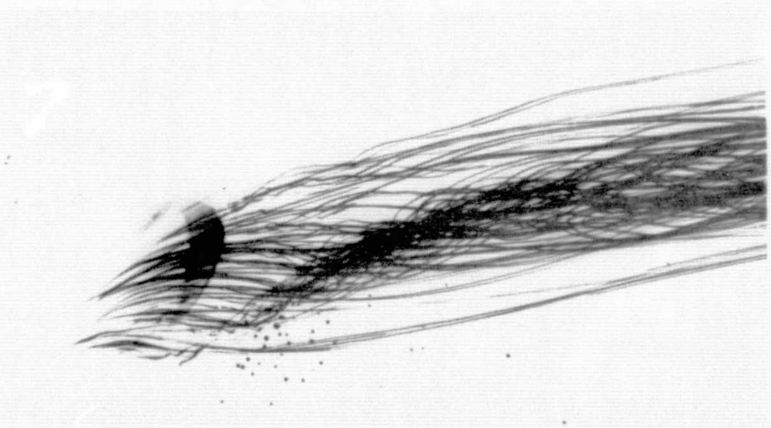
Figure 17. - Streaklines for in-line injection 30° to wall.



(a) BLOWING RATIO, M = 0.3.



(b) BLOWING RATIO, M = 0.75.



(c) BLOWING RATIO, M = 0.75. CLOSE-UP OF UPSTREAM HOLE.

Figure 18. - Streaklines for compound angle film injection.

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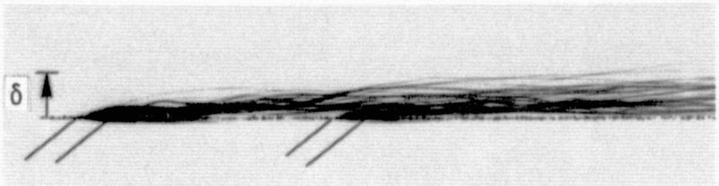
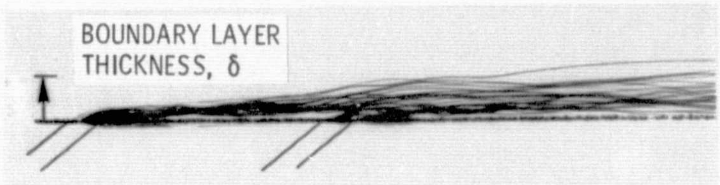
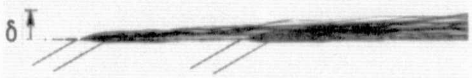


BLOWING RATIO, M = 0.3



BLOWING RATIO, M = 0.9

(a) COMPOUND ANGLE INJECTION.



BLOWING RATIO, M = 0.3

BLOWING RATIO, M = 0.8

(b) IN-LINE INJECTION.

Figure 19. - Side view comparison of streaklines of compound angle and in-line injection.

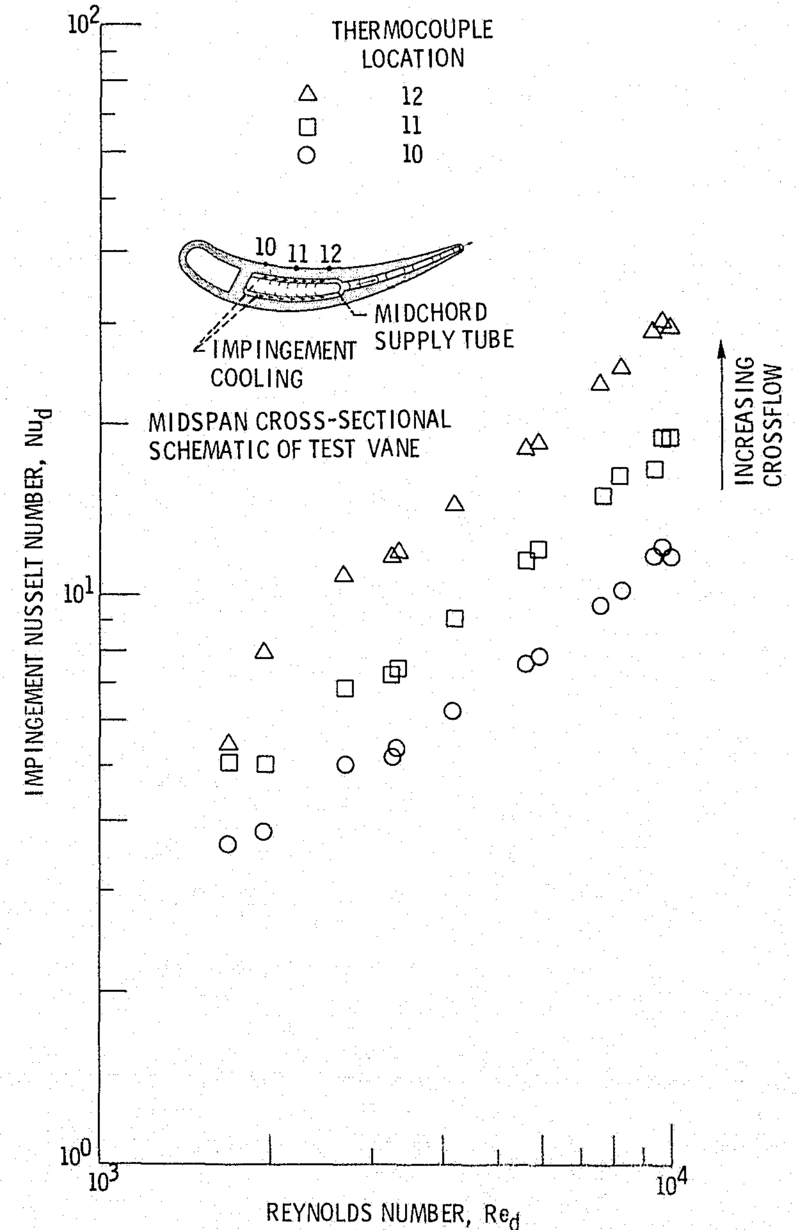


Figure 20. - Experimental coolant side Nusselt numbers as function of jet Reynolds number. (Pressure surface.)

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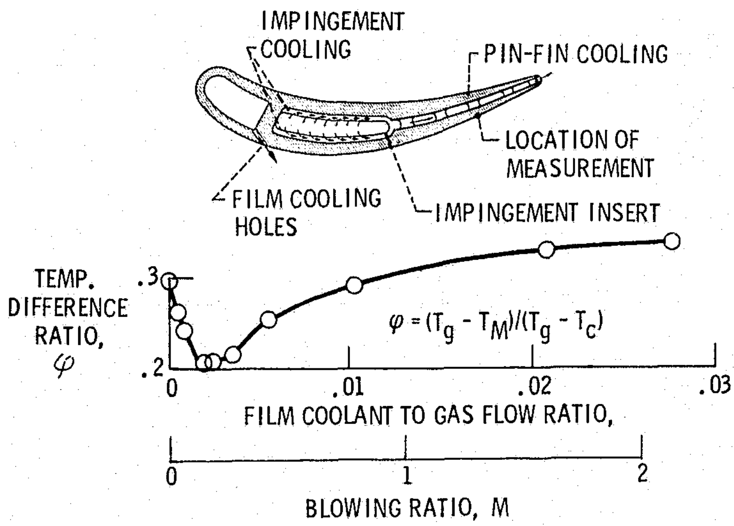

Figure 21. - Adverse effect of film cooling.

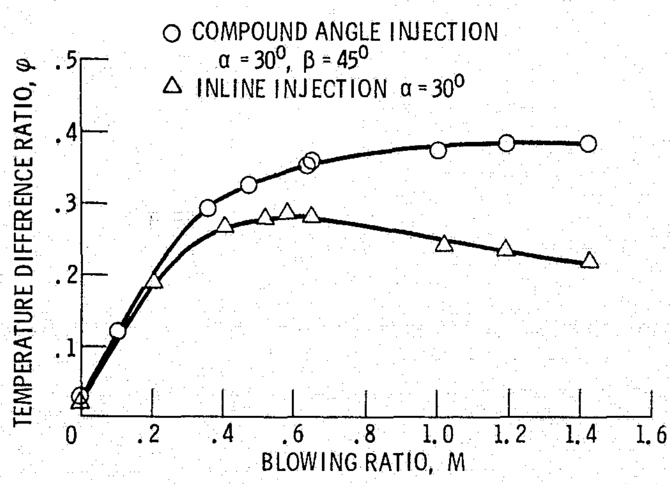

Figure 22. - Effect of injection angle on cooling of a turbine vane surface.

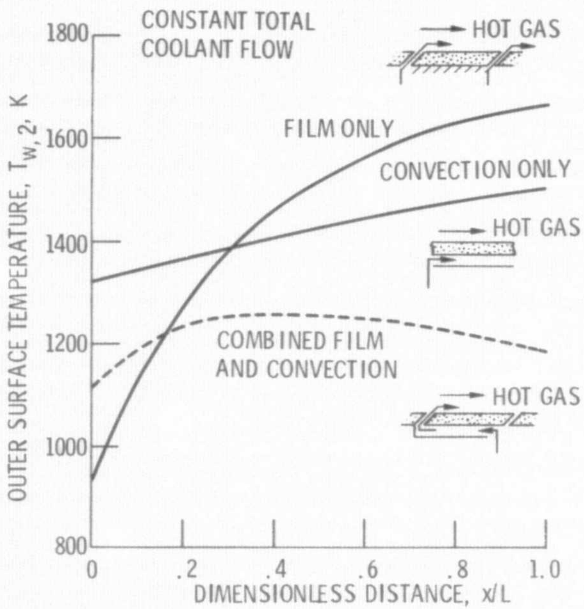

Figure 23. - Importance of combining convection and film cooling.

Figure 24. - Ceramic coated turbine blade.

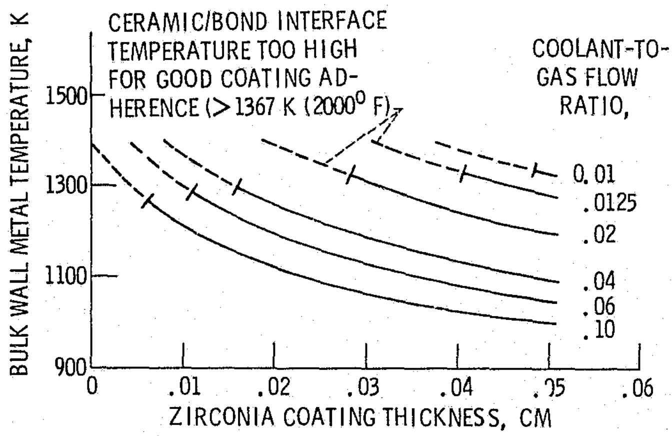

Figure 25. - Metal temperatures and coolant flows for turbine vane coated with a thermal barrier ceramic. (Advanced core turbine: inlet gas temperature, 2200 K, gas pressure, 40 atm; coolant temperature, 811 K.)

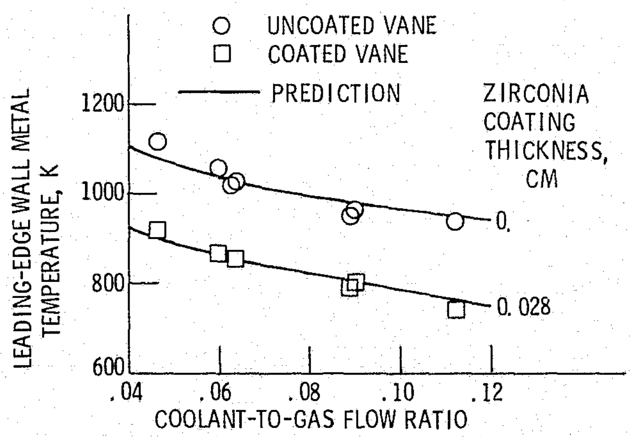

Figure 26. - Calculated and measured wall metal temperatures of uncoated and zirconia-coated turbine vanes operating in a research engine. (Inlet gas temperature, 1644 K, inlet gas pressure, 3 atm; coolant temperature, 319 K.)

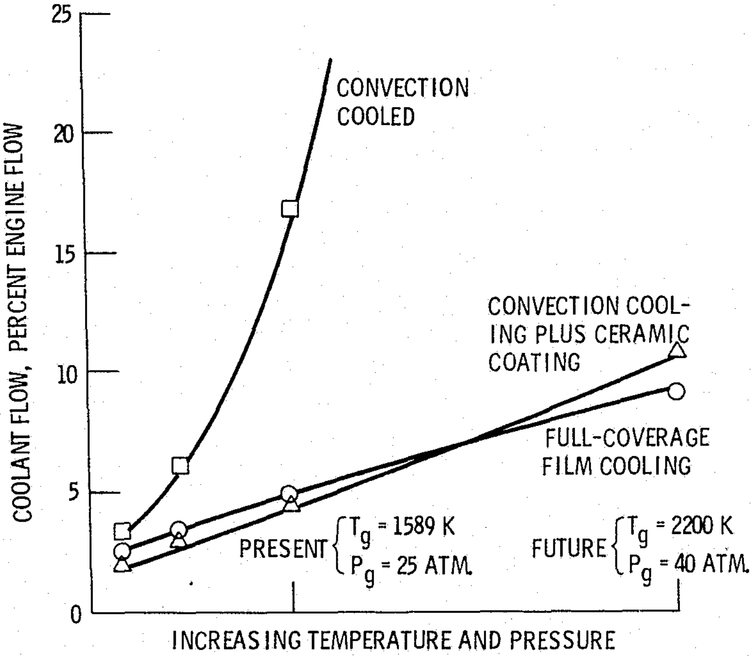

Figure 27. - Cooling requirements for several cooling methods.

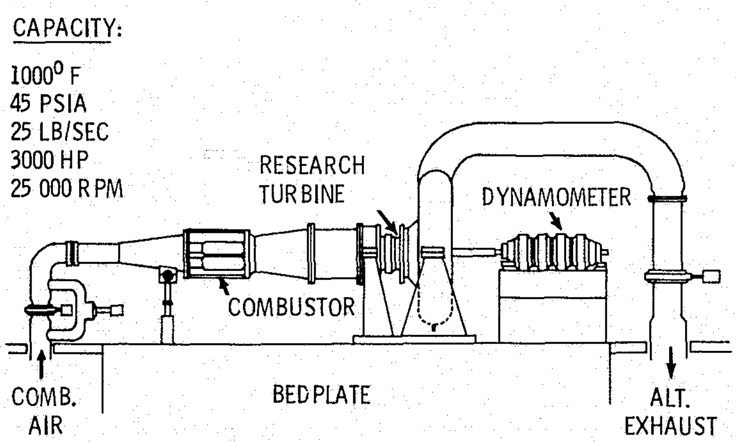

Figure 28. - Warm core turbine facility.

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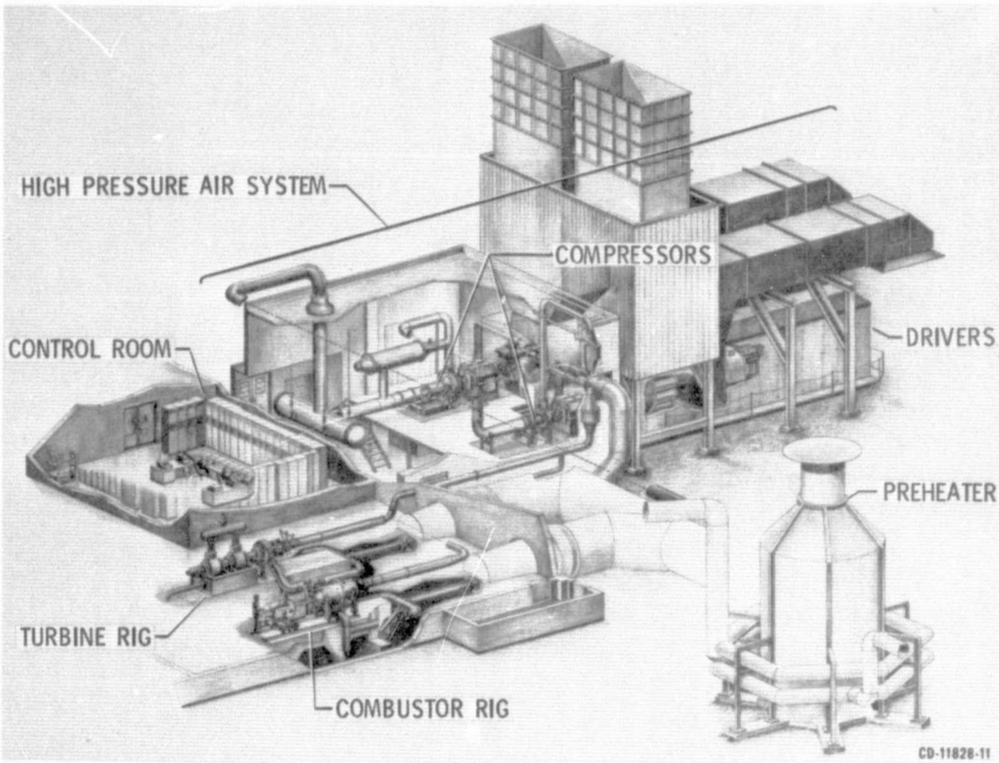

Figure 29. - High pressure facility.