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ANALYSES OF COOLED TURBINE AIRFOIL CONFIGURATIONS AT CONDITIONS OF SUPERSONIC FLIGHT

by Francis S. Stepka

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Cleveland, Ohio

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ABSTRACT

Summarized are the results of an analysis conducted by the General Electric Company under contract to NASA of various first stage turbine stator vane and rotor blade cooling designs which would have application to engines of aircraft for high-speed flight such as a supersonic transport. Also summarized are the influence of various factors on cooling design and performance of turbine vanes and blades. These included use of future high-temperature materials, use of higher material thermal conductivity, changes in design geometry, changes in combustor gas temperature profiles, foreign object damage, and the effects of steady-state and transient engine and flight conditions.

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ANALYSES OF COOLED TURBINE AIRFOIL CONFIGURATIONS AT CONDITIONS OF SUPERSONIC FLIGHT*

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Lewis Research Center

SUMMARY

This report summarizes the results of analyses conducted by the General Electric Company, under contract to the NASA, of various first-stage turbine vane and blade cooling designs which would have application to engines of aircraft for high-speed flight such as a supersonic transport aircraft. Summarized are the influence of various factors on airfoil cooling design, metal temperatures, and required cooling air flows. Cooled vanes and blades were designed for an assumed requirement of a 1000-hour life at conditions of flight at Mach 3 at 75 000 feet (22.8 km) with a first-stage turbine rotor blade inlet average gas temperature of 2200° F (1478 K).

The results indicated that various cooling designs utilizing various cooling methods can be made to meet the desired life. The high temperature of the compressor exit bleed air (about 1200° F (920 K)) limited the manner in which the air could be ducted within the blade and limited the benefits provided by changes in blade design or cooling methods. As a consequence, the required cooling air flows for the various designs did not differ much. At design flight conditions, cooling air-to-engine air flow ratios for the first-stage turbine airfoils ranged from 0.027 to 0.033 for six of the vane designs and 0.025 to 0.029 for five of the blade designs which were essentially able to meet the metal temperature-life criteria for a material with properties of IN 100.

Transient analysis of metal temperatures indicated that large excursions in metal temperatures occurred during engine acceleration and deceleration between idle and take-off conditions. Transient temperature differences between leading or trailing edges and the airfoil interior were as large as 747° F (415 K). The range of maximum metal temperature differences experienced by the airfoils during the excursions between engine accelerations and decelerations was least for a stator vane design which was cooled by impingement and convection and least for a blade design also had the best predicted low-cycle fatigue life of three blade designs which were detail stress analyzed.

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INTRODUCTION

Advanced high performance gas turbine engines will utilize high turbine inlet gas temperatures that will require cooling of the turbines with air bled from the engine compressor. At conditions of high supersonic cruise, the cooling of these turbines becomes particularly difficult because of the high temperatures of the compressor exit bleed air (of the order of 1200° F). Minimizing the amount of bleed air by use of effective cooling methods is necessary to attain and maintain the high performance gains indicated by the use of high cycle temperatures. As a result, an analytical investigation was conducted by the General Electric Company under contract to NASA (refs. 1 to 5) to evolve, evaluate, and compare a wide variety of turbine stator vane and rotor blade cooling configurations so as to provide some guidance on the cooling methods and/or cooling configurations that should be utilized in advanced engines. The purpose of this report is to provide a reasonably concise summary of the large amount of information presented in references 1 to 5.

These references examined, respectively, the following:

(1) Various cooling methods applied to the leading and trailing edge regions of cooled turbine airfoils

(2) Complete convection air cooled vane and blade designs

(3) Complete transpiration and film cooled vane and blade designs

(4) Vane and blade designs with combined methods of cooling

(5) Vane and blade cyclic life

The basic design point for the analyses was flight at Mach 3 at an altitude of 75 000 feet (22.8 km) with a turbine rotor inlet average gas temperature of 2200° F (1478 K). The cooled vane and blade designs evolved were made to meet a 1000-hour life at these conditions. The designs were also analyzed to determine the effects on metal temperatures and cooling flow requirements of: using assumed future high temperature materials, higher metal thermal conductivity, changes in design geometry, changes in combustor temperature profiles, foreign object damage (resulting in dents or holes in the surface or plugging of internal holes), and other turbine inlet gas temperatures and flight conditions. Transient metal temperatures resulting from engine light-off, acceleration, deceleration, and shutdown were also determined. Using these transient data, detailed stress analysis was made of selected airfoil cooling configurations and their cyclic lives were predicted.

CONDITIONS AND METHOD OF ANALYSIS

The study consisted of (1) designing first-stage vane and blade configurations, which

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utilize various methods or combined methods of cooling, and (2) evaluating various factors affecting their performance. This analytical study was not limited to configurations that are within the state-of-the-art for manufacturing at the present time. Should futuristic configurations show enough promise, it was believed manufacturing methods could be developed. The configurations were designed to meet a 1000-hour steady-state life at design point conditions of flight at Mach 3 at an altitude of 75 000 feet (22.8 km) and at a turbine rotor inlet gas temperature of 2200° F (1478 K). The metal temperatures and cooling air flows for the configurations evolved were determined for design point operation and for other conditions influencing airfoil cooling performance. These conditions included: use of future high temperature blade and vane materials, use of higher material thermal conductivity, changes in design geometries, and foreign object damage to the airfoils. Other conditions considered were changes in turbine rotor inlet gas temperatures ranging from 2000° to 2500° F (1367 to 1644 K), changes in flight Mach number from 1.2 to 3.5, and transient conditions of light-off, idle, takeoff, and shutdown.

References 1 to 5 present detailed information on the conditions of the analyses, the basis for selecting these conditions, and the methods of analysis which include the various equations, parameters, experimental data, and correlations used. The present report will only summarize this information.

Turbine External Gas Environment

The environment surrounding the turbine vanes and blades first required defining the engine cycle data. A summary of these data for selected conditions of flight, and at idle and takeoff, are shown in table 1. Using these data, turbine interstage conditions were calculated, airfoil aerodynamic profiles were designed, and the effective gas temperature, gas velocity, and pressure distributions around the airfoils were determined. Values of local gas-to-blade heat transfer coefficients were then calculated using equations presented in reference 1. The aerodynamic profiles of these vanes and blades at three span locations are shown in figure 1.

Because the combustor radial and circumferential gas temperature profiles influence blade metal temperature and life, the gas temperature profiles used in the analysis are typical of these expected in engines. Figure 2 shows the assumed design point gas temperature profiles entering the first-stage vane and blade rows. The profile for the stator vane is representative of that at a circumferential hot spot and was used in the design of the vane. The design hot spot temperature corresponds to a combustor pattern factor (ratio of the difference between the hot spot and average burner outlet temperatures to the difference between the average burner outlet and compressor exit tempera-

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tures) of 0.20. Shown in the figure are profiles that were also used in the analysis to represent departures from the design point profiles and pattern factor.

Coolant Conditions

The cooling air supply conditions to the vanes and blades were obtained by calculating the pressure changes and temperature rises along the flow path from the compressor exit to the airfoil bases. The assumptions and equations used to predict the cooling air pressures and temperatures at the inlet to the airfoils and within the airfoils are presented in reference 1. The equations used for determining the wall-to-coolant heat transfer coefficients for airfoils convectively cooled with air or liquid metal and the equations used for the cooling effectivenesses of film or transpiration cooled airfoils are presented in references 1 and 3, respectively.

Basis of Configuration Design

The criteria used to provide satisfactory designs of vanes and blades to meet a 1000-hour steady-state life are described in detail in appendix G of reference 1. The criteria are based on past analytical and experimental experience for typical cooled airfoil configurations with given mechanical and vibratory stresses, stress concentrations, and temperature distributions. The criteria for the levels of stress determined for the airfoils in the engine used in this analysis require that the first-stage vane and blade maximum metal temperatures and chordwise metal temperature differences not exceed the values specified in table II. The table shows these temperatures for three materials (A, B, and C) and for airfoil designs with and without film cooling holes. The reductions in allowable metal temperatures for designs with film cooling holes are to account for the influence of stress concentrations associated with the presence of these holes.

IN 100 was designated as material A. Materials which may be developed in the future and have properties that will permit 100° and 200° F (55 and 111 K) higher allowable metal temperatures than IN 100 were designated as materials B and C, respectively. The selection of allowable maximum temperatures for the porous materials that would be used for transpiration cooled vanes and blades was based on general considerations of both the expected strength properties of these materials and on the problem of blockage of the small openings in the materials caused by oxidation. The allowable maximum metal temperatures for porous materials such as N-155 was assumed to be 1400° F (1033 K) based primarily on oxidation and clogging problems. For other materials, material porosities, weaves, or transpiration cooling concepts, where the oxidation

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problems can be reduced, maximum allowable metal temperatures of 1600° F (1144 K) and 1800° F (1255 K) were assumed to apply to current state-of-the-art and future developments, respectively.

Heat Transfer Analysis

The required cooling air flows and local blade and vane metal temperatures were determined by computer programs utilizing two- and three-dimensional heat transfer analyses along with a one-dimensional compressible coolant flow analysis. The details of the program, which considered both steady-state and transient engine conditions, are described in reference 1. The nodal breakdown and boundary conditions (such as the local internal and external heat transfer coefficients, coolant and effective gas temperatures) needed for the analysis of the various cooling methods and airfoil configurations are presented in references 1 to 4. The heat transfer and flow equations and correlations used in the analysis were considered to be the best currently available. Some of these correlations and equations, however, are not considered to be completely satisfactory. The correlations used for film cooling, for example, are limited to application where the airfoil aerodynamic shape and film-cooling hole array is similar to that used in arriving at the correlations. Considerable experimental study is required to improve the correlations and/or provide a greater confidence in the accuracy of the correlations. This is particularly so for the film, transpiration, and impingement cooling methods.

Transient Conditions

The response of airfoil metal temperatures was determined for conditions of slow and fast engine acceleration and deceleration transients between idle and sea level takeoff conditions, and for typical combustor lightoff and engine shutdown conditions. The changes in vane inlet average gas temperature, cooling air supply temperature, cooling air supply pressure, and combustion gas flow rate for fast and slow engine acceleration and deceleration transients are shown in figures 3(a) and (b), respectively.

Life Predictions of Selected Airfoil Designs

A detailed analysis of selected airfoil cooling designs was conducted to determine whether the designs as evolved and based on the approximate criteria described in the section Basis of Configuration Design could meet the assumed required life of 1000 hours.

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The analysis determined the steady-state creep and rupture lives of the airfoils at both cruise and sea level takeoff conditions, the effect that repetition of these steady-state conditions would have on life, and the effect that slow engine accelerations and decelerations between idle and takeoff conditions, as shown in figure 3(b), would have on life. The analysis was applied to four cooled airfoil configurations. These consisted of three blade designs and one vane design. A general description of the method of analysis is presented in references 1 and 5.

CONFIGURATIONS ANALYZED

General Physical Descriptions

The initial evaluation of the potentials and limitations of various cooling designs and the use of various cooling methods was conducted in the analysis of reference 1. The analysis considered only two-dimensional heat transfer and was primarily concerned with the critical leading and trailing edge regions. Results of the analysis provided the basis for the design of complete vanes and blades.

The cooled airfoil designs evolved were obtained from three separate phases of a design and analysis schedule. The phases were divided according to the method or methods of cooling which were applied to the airfoils. The cooled configurations designed in the first phase, and reported in reference 2, were convection, air-cooled. Those in the second phase, reported in reference 3 were film or transpiration air-cooled. Those in the last phase, reported in reference 4, utilized combined methods of cooling, such as film and convection air-cooling or convection liquid metal and convection air-cooling. A total of eight vanes and seven blades were designed and analyzed. Various fabrication methods were considered. They included: casting, forging, and forming from sheet metal.

Vane Detail Descriptions

Cast or forged convection cooled vane, design 1. - This cooling configuration shown in figure 4(a) utilized radial flow of cooling air through a series of 0.10 and 0.125 inch (2.54 and 3.18 mm) diameter holes in the relatively low heat flux region at the midchord of the airfoil. The air from these holes near the tip enters a central cavity which serves as a plenum to supply cooling air to the trailing edge. The air exits from the trailing edge in the direction of the gas flow through a series of 0.04 by 0.16 inch (1.02 to 4.07 mm) slots. The leading edge is cooled by a separate supply of air which enters one of the ends of an oval shaped tube running the length of the vane. The air from this

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tube then exits from 0.055 inch (1.4 mm) diameter holes spaced at 0.20 inch (5.08 mm) to impingement cool the leading edge. This air then reverses direction and enters the central cavity of the airfoil after flowing over horizontal fins that support the tube and augment the cooling in the region around the tube.

Strut insert, convection cooled vane design II. - Most of the cooling air to this configuration, shown in figure 4(b), enters a cavity in the strut insert near the leading edge. Part of this air exits through 0.053 inch (1.35 mm) diameter holes spaced at 0.10 inch (2.54 mm) to impingement cool the leading edge. This air is then ducted through a series of staggered slots shown in the expanded views in figure 4(b), to enter a cavity in the center of the vane which supplies air to the trailing edge. The other part of the air from the leading edge supply cavity enters a series of horizontal slots along the periphery of the airfoil. These slots are formed by fins on the strut which are 0.010 inch (0.25 mm) thick and spaced on 0.030 inch (0.76 mm) centers. The height of the fins (or slot depth) is initially 0.050 inch (1.27 mm) but tapers along the chord to 0.040 inch (1.02 mm). At about three-quarters of the chord, the air from these slots is joined by the air from the central cavity to cool the trailing edge. The trailing edge exit passages are 0.02 inch (0.51 mm) high by 0.045 inch (1.14 mm) wide separated by a 0.1 inch (0.25 mm) thick web for about one-fourth of the span near root. For the remainder of the span the passages have the same height and web thickness but are 0.06 inch (1.52 mm) wide. The trailing edge cooling is supplemented by an additional air flow from a 0.18 inch (4.57 mm) diameter opening in the base which directly enters the central cavity. The slots around the periphery of the airfoil were assumed formed by the joining of the shell of the airfoil to fins cast or forged on the strut insert.

Corrugated sheet metal insert, convection cooled vane design III. - This design utilized the advantages of large heat transfer surface areas and lightness of construction offered by sheet metal and the use of corrugations. The cooling flow in the vane, shown in figure 4(c), follows three separate circuits. The leading edge cooling circuit is fed from a single tapered radial passage. This passage feeds three radial corrugated segments. The heated air from each of the segments is collected in a second tapered radial passage and eventually discharged at the tip of the airfoil. These corrugations are 0.030 inch (0.76 mm) high with a 0.040 inch (1.02 mm) pitch. The midchord section is cooled by radial flow of air through corrugated passages fed from the vane base and discharged at the tip. The flow to these corrugations, which are 0.040 inch (1.02 mm) high with a 0.060 inch (1.53 mm) pitch, is separately metered to the suction and pressure surfaces. The trailing edge cooling is accomplished by supplying air to three radial segments of corrugations. The air to the segment at the tip and midspan is supplied from cavities in the center of the vane. The flow to the cavity forward of the midchord which supplies the tip segment is metered by a 0.165 inch (4.19 mm) diameter orifice at the base. The flow to the other cavity in the vane center which supplies the segment at the midspan is

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metered by a 0.091 inch (2.31 mm) diameter orifice at the base. The air to the third corrugated segment is supplied directly from the vane base. The lower end of each of these segments has a small plenum which distributes the air around the periphery of the segment. The air then flows radially through each of these segments to a point at which the corrugations are cut off diagonally. The flow is then ejected into the narrow region of the trailing edge and finally out through a 0.010 inch (0.254 mm) wide slot. The corrugation height and pitch in this region are the same as for those in the leading edge region. The trailing edge required additional cooling because of the large rise in cooling air temperatures in these segments. This cooling was provided by a supplemental flow of air through openings at the tops of each segment bled from the two cavities in the center of the vane.

Transpiration cooled vane, design IV. - This design, shown in figure 4(d), consists of a porous shell bonded to a cast or forged strut insert. A central cavity in the strut serves as a plenum for distributing the air to 40 individually sized metering holes over the surface of the strut. The air from these holes enters cavities (forming a waffle pattern) in the external surface of the strut from which it is distributed to the porous wall. The fins forming the cavities were 0.50 inch (1.27 mm) high and thick. The diameter of the holes for the vane designed for an allowable wall temperature of 1600° F (1145 K) ranged from 0.058 to 0.298 inch (1.47 to 7.56 mm). The porous wall was 0.025 inch (0.64 mm) thick and had a permeability of 2.085×10^{-10} feet (63.5 pm, where pm is pico-meters) at the leading edge region and 0.33×10^{-10} feet (10.1 pm) for the remainder of the airfoil. Permeability is a measure of the quantity of coolant flow through a porous wall for given differences of the squares of the pressure on either side of the wall. The sizes and distributions of the holes for metering air to the porous wall and further details on the design are provided in reference 3.

Film cooled vane, design V. - This design, shown in figure 4(e), is essentially a hollow airfoil with a multiplicity of holes over the surface which meter and distribute the air for film cooling of the metal. There are three supports inside the airfoil interconnecting opposite walls. Two of the supports have large holes along the span to provide a uniform static pressure in the cavities formed by these supports. The support at the leading edge has 0.055 inch (1.5 mm) diameter holes spaced at 0.150 inch (3.8 mm) along its length to provide impingement cooling of the leading edge. This cooling supplements the combination of convection and film cooling provided by holes in the wall at the leading edge. The leading edge of the airfoil has five groups of holes consisting of two rows each. The rows run the length of the span. The holes are normal to the wall surface and are 0.004 inch (0.102 mm) in diameter and are spaced on 0.020 inch (0.51 mm) centers in each row with the rows spaced on 0.008 inch (0.203 mm) centers. The film cooling holes in the remainder of the airfoil are at 35° to the wall surface and are tapered from either 0.003 to 0.007 inch (0.076 to 0.178 mm) diameter or from 0.004 to

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0.008 inch (0.102 to 0.203 mm) diameter from inlet to exit. The film-cooling of the trailing edge is supplemented by convection cooling provided by the air flowing from the aft cavity into and through 0.025 inch (0.635 mm) diameter holes in the trailing edge. These holes are center-spaced at 0.055 inch (1.40 mm).

Film-convection cooled vane, design VI. - This design, shown in figure 4(f), is composed of two separate cooling circuits. The coolant to the leading edge flows up a radial hole, then through a series of slots 0.115 by 0.275 inch (2.92 by 7.0 mm) along the span to impingement cool the leading edge. This air then flows from the cavity at the leading edge through holes in the wall to film cool the airfoil. There are seven rows of film cooling holes in this region of the vane; two rows on each side of the suction and pressure surface aft of the leading edge (designated as "gill holes"), a row at the stagnation point, and a row on each side of the stagnation point. The three rows of holes at the leading edge region were inclined upwards 30° to the airfoil surface. These holes were 0.012 inch (0.305 mm) in diameter and spaced at 0.0363 inch (0.923 mm). The two rows of gill holes on the pressure surface were 0.04 inch (1.01 mm) in diameter and spaced at 0.0705 inch (1.79 mm). The two rows of gill holes on the suction surface were 0.04 inch (1.01 mm) in diameter and spaced at 0.141 inch (3.58 mm).

The midchord and trailing edge is cooled by air which is fed at the root into radial passages that direct the flow outward to the tip where the air reverses flow direction. At the base the flow direction again reverses after which the air exits horizontally through 0.014 by 0.07 inch (0.356 by 1.78 mm) slots in the trailing edge. This flow convectively cools the midchord and trailing edge region and supplements the film cooling provided by the holes in the leading edge region.

Transpiration and convection cooled vane, design VII. - This design, shown in figure 4(g), consists of porous walls at the leading and trailing edge regions attached to a cast or forged shell in the midchord region. A hollow strut insert to which this combination wall is attached, provides support for the wall and provides a means for metering and distributing the cooling air around the airfoil. The air enters the central cavity in the strut and flows to the leading edge porous section without restriction to the stagnation region and by means of metering holes to the suction and pressure sides. The bulk of the flow from the central cavity flows into and through chordwise passages to cool the midchord by forced convection. These passages are 0.05 inch (1.27 mm) wide and 0.02 inch (0.51 mm) high with 0.02 inch (0.51 mm) between the passages. The flow to these passages is supplied from the central cavity through spanwise openings at three positions along the chord. The entire flow from these passages then exits through the porous trailing edge region. The porous walls are 0.025 inch (0.635 mm) thick and have a permeability of 3.176×10^{-10} feet (96.8 pm) at the leading edge and 3×10^{-10} feet (91.4 pm) at the trailing edge.

Impingement and convection cooled vane, design VIII. - This design, shown in figure 4(h), consists of a double shell. The inner shell is a 0.008 inch (0.203 mm) thick sheet metal insert through which numerous small holes are machined to provide cooling by the impingement of jets on the outer shell in the midchord and leading edge regions. This insert is not directly connected to the outer shell. The spacing between these two members is kept by means of formed dimples in the insert. The insert is supported by attachment at the inner and outer vane bands. The insert has constant diameter holes in each of the rows running along the span. The spacing of these rows along the periphery and the sizes of the holes in each row were determined by the local cooling requirements. It can be seen from figure 4(h) that the largest concentration of the rows and smallest diameter holes were located on the suction side of the vane to obtain a higher coolant flow and heat transfer coefficient in this higher heat flux region. The holes in the insert ranged from 0.012 to 0.052 inch (0.305 to 1.32 mm) diameter. The trailing edge of this configuration is the most difficult region to cool. To pass the required flow to cool the pressure side of the trailing edge, 0.90 by 0.036 inch (22.9 by 0.915 mm) slots were required. Cooling of the suction side of the trailing edge could not be accomplished by convection cooling alone and required supplemental cooling. The additional cooling was required because of the low convective coefficients existing in the trailing edge cooling passages combined with the necessity to use air that was already heated prior to reaching the trailing edge. Cooling of the suction side of the trailing edge was accomplished by convection cooling with 0.02 inch (0.51 mm) diameter holes spaced at 0.06 inch (1.53 mm) along the span supplemented by film cooling from 0.03 inch (0.76 mm) diameter holes spaced at 0.177 inch (4.5 mm) upstream of the previous holes.

Blade Detail Descriptions

Cast or forged convection cooled blade, design I. - This design, shown in figure 5(a), is essentially a one-piece construction utilizing radial flow of the air to convectively cool the leading edge and midchord regions. The trailing edge is convectively cooled by chordwise flow of air through holes in the trailing edge. Although the cooling ability of this design was not expected to be good, it was selected for analyses because of its known capability for manufacture and because it could be used as a basis for comparison with other cooling configurations which, although having better cooling characteristics, involved more complex construction.

The coolant flow is supplied through two openings in the blade base. One supplies a radial passage at the leading edge from which the air is ducted near the tip to turn in the chordwise direction. Part of this air exits through a 0.3 by 0.12 inch (7.6 by 3.05 mm) opening at the tip of the blade near the trailing edge and the remainder through a 0.5 by 0.03 inch (12.7 by 0.76 mm) slot in the trailing edge. The other opening in the blade base supplies air to a radial hole from which the air exits near the tip to enter into another radial passage which has openings near the tip and root. From this passage the air supplies a plenum passage which distributes the air to and through a series of holes in the trailing edge. The holes in the outer two-thirds of the span at the trailing edge are 0.04 inch (1.02 mm) diameter spaced radially at 0.0825 inch (2.1 mm) and the holes near the root are 0.02 inch (0.51 mm) diameter spaced at 0.05 inch (1.27 mm).

Strut insert, convection cooled blade, design II. - This design, shown in figure 5(b), consists of inner and outer shells with chordwise fins separating the two shells. The fins can be an integral part of one of these shells and are assumed to have a good metallurgical bond to the other. The cooling air from the base enters a central cavity, formed by the inner shell and exits through a spanwise row of 0.055 inch (1.4 mm) diameter holes spaced at 0.125 inch (3.18 mm) to impingement cool the leading edge region. The flow from this region divides and flows through 0.03 by 0.06 inch (0.76 by 1.52 mm) slots formed by the fins around the suction and pressure sides of the airfoil; these sides had 68 and 45 slots, respectively. Near the trailing edge the flow combines and exits through 0.025 by 0.04 (0.635 by 1.02 mm) slots in the trailing edge.

Corrugated sheet metal insert, convection cooled blade, design III. - This design, shown in figure 5(c), was chosen because of its mechanical simplicity. The design consists of spanwise corrugated sheet metal spacers separating an inner and outer shell. A good metallurgical bond between these parts is assumed. The air enters the blade base and flows radially outward through the space formed by the two shells and corrugation, then exits through six 0.01 inch (0.25 mm) wide slots in the trailing edge. In order to accommodate the flow required to cool the blade, most of the trailing edge exit area had to be used. The corrugations and the inner shell are therefore cut away toward the trailing edge to allow passage of the flow. The corrugations in the midchord region of the airfoil have a height of 0.040 inch (1.02 mm) with pitch of 0.22 inch (5.6 mm). The corrugation aft of the midchord region and extending toward the trailing edge have a height of 0.020 inch (0.51 mm) and a pitch of 0.040 inch (1.02 mm). The leading edge did not utilize corrugations, since the combination of the high heat flux in this region and the rapid temperature rise of the cooling air when corrugations were considered, precluded their use.

Transpiration cooled blade, design IV. - The general construction of this design, shown in figure 5(d), is similar to that of vane design IV. It consists of a 0.025 inch (0.635 mm) thick porous shell bonded to a cast or forged strut insert. A central cavity in the strut serves as a plenum for distributing the air to 40 individually sized metering holes over the surface of the strut. The air from these holes enters cavities formed by the external strut surface from which it is distributed to the porous wall. The diameter of the metering holes for the blade designed for allowable wall temperatures of 1600° F (1145 K) ranges from 0.032 inch (0.64 mm) on the suction side to 0.28 inch (7.1 mm) at the leading edge. The permeability of the wall is 0.716×10^{-10} feet (21.8 pm) at the leading edge and 0.10×10^{-10} feet (3.05 pm) for the remainder of the blade.

Film cooled blade, design V. - This design, shown in figure 5(e), is similar in construction to vane design V. The blade is essentially a hollow airfoil with a multiplicity of holes over the surface which meter and distribute the air for film cooling of the metal. There are three support members inside the airfoil; two of these have large interconnecting holes to provide uniform static pressures in the cavities formed by the supports. The support at the leading edge has 0.050 inch (1.27 mm) diameter holes spaced at 0.150 inch (3.81 mm) along the span to provide impingement cooling of the leading edge. This cooling is supplemented by a combination of convection and film cooling that is provided by holes in the wall at the leading edge. This region has four rows of 0.004 inch (0.102 mm) diameter holes normal to the blade wall and radially spaced at 0.012 inch (0.305 mm). The film cooling holes in the remainder of the blade are at 35° to the wall and consist of groups of holes with two spanwise rows in each. One group of holes is located in each side aft of the leading edge. The holes in this group are 0.004 inch (0.102 mm) diameter with a radial spacing of 0.016 inch (0.406 mm) in each row. The rows were spaced at 0.008 inch (0.203 mm). The other groups of holes which were distributed around the periphery have holes which taper from 0.004 to 0.008 inch (0.102 and 0.203 mm) diameter from the inside to the outside wall. These holes are spaced at 0.024 inch (0.61 mm) in a row and the rows are spaced at 0.008 inch (0.230 mm). The cooling of the trailing edge is supplemented with convection cooling with air flowing from the aft internal cavity into and through 0.010 inch (0.25 mm) diameter holes spaced at 0.020 inch (0.51 mm) in the trailing edge.

Film and convection cooled blade, design VI. - This design, shown in figure 5(f), is composed of two separate cooling circuits. The cooling air to the leading edge circuit starts radially upward at the midchord and takes a triple reversal of flow as it flows through three radial passages toward the leading edge. From the most forward of these radial passages, the air flows through a spanwise row of 0.0271 by 0.05 inch (0.69 by 1.27 mm) slots which were spaced at 0.06 inch (1.52 mm). The air from these slots cools the leading edge. Additional cooling of the extreme leading edge region was obtained by a combination of convection and film cooling provided by two rows of 0.014 inch (0.36 mm) diameter holes radially spaced at 0.030 inch (0.762 mm) and inclined upward 30° to the surface of the wall. Some of the air from the leading edge cavity is used to film cool the region aft of the leading edge by means of two rows of holes on either side of the blade. The holes on the suction side are 0.014 inch (0.36 mm) diameter and are spaced radially at 0.035 inch (0.89 mm). The holes on the pressure side are 0.015 inch (0.38 mm) diameter and spaced radially at 0.062 inch (1.57 mm). Convection cooling of part of the midchord and trailing edge is accomplished by flow of air up a radial passage, then downward from the tip into another radial passage from which the air enters a series of 0.0085 by 0.05 inch (0.216 by 1.27 mm) slots in the trailing edge which had a spacing of 0.06 inch (1.52 mm). Although the convection cooling of the trailing edge was supplemented by the film cooling air from the holes aft of the leading edge, an additional row of film cooling holes upstream of the trailing edge on the pressure side was required. The holes in the row were 0.01 inch (0.25 mm) diameter with a radial spacing of 0.033 inch (0.84 mm).

Liquid metal and air convection cooled blade, design VII. - This design, shown in figure 5(g), consists of four radial passages filled with potassium for cooling the leading edge and midchord regions. The metal thickness at the trailing edge region did not permit installation of adequate size liquid metal passages to cool the region to the desired temperature. The region was convection cooled by air flow through a radial cavity from which the air exited through chordwise holes.

The liquid metal in blade design VII is used in a boiling-condensing system. The airfoil section serves as the boiler to vaporize the liquid and absorb the heat that was transferred from the gas stream to the blade. A heat exchanger at the blade shank transfers this heat to cooling air and serves as a condenser for the vapor. The circulation of the liquid metal results from the difference in densities between liquid and vapor and the large centrifugal force field created by the blade rotation.

The condensing unit consists of a series of small oval shaped slots around the circumference of the extension of the radial passages in the airfoil. The diameters of these radial passages are 0.10 inch (2.54 mm) for the passage at the leading edge and 0.20 inch (5.08 mm) for the three others aft of this region. The oval slots of the condenser are 0.020 by 0.035 inches (0.51 by 0.89 mm). The length of the condenser needed to absorb the heat picked up by the liquid metal was 2.5 inches (63.5 mm). The radial air cooled passage near the trailing edge was triangular in shape 0.35 inches (8.9 mm) long with the width tapering from 0.14 to 0.05 inch (3.56 to 1.27 mm). The chordwise holes in the trailing edge were 0.016 inch (0.41 mm) diameter spaced at 0.035 inches (0.89 mm).

RESULTS AND DISCUSSIONS

The following paragraphs summarize the results that were obtained from a design and analytical study of cooled first-stage turbine vane and blade configurations applicable to Mach 3 transports. The results are presented in figures 6 to 8 and in tables III to IX.

Leading and Trailing Edge Regions

The initial phase of the study as reported in reference 1 analyzed the application of various cooling methods to the difficult-to-cool leading and trailing edge regions of vanes and blades. The results of the analysis not only provided the basis for the selection of leading and trailing edge configurations for complete blade designs which were evolved in the study but also provided an insight into the influence of various factors on metal temperature levels and distributions. Some of these factors were the coolant passage sizes and geometry, coolant flow path, coolant flow rate, coolant supply pressure, and the gas pressure level at which the coolant exits from the airfoil.

The results of this phase of the analysis indicated that the high temperature of the turbine cooling air (compressor exit bleed air) places severe restrictions on the manner in which the air can be used. The temperature of this air as supplied to the bases of the first-stage vanes and blades at conditions of cruise at Mach 3, for example, was about 1190° F (918 K) and 1270° F (962 K), respectively.

The study, in general, indicated that the following rules should be observed in the design of cooled turbine airfoils intended for application to conditions of high turbine inlet gas temperature and high speed flight:

(1) The flow path of the cooling air in the leading and trailing edge regions should be short to avoid rapid and excessive heating of the air and metal.

(2) Impingement cooling of the leading edge region should be utilized because of the high heat flux in this region.

(3) Convection cooling of the long thin trailing edges by means of air flow through slots or holes should be utilized.

(4) Use of film cooling should be considered as a means of both expelling heated air and supplementing other methods of cooling.

Complete Airfoil Designs

This section of the report will summarize the results of the phase of the analysis (refs. 2 to 4) in which complete vane and blade cooling designs were evolved. The configurations were designed to meet the requirements of a 1000-hour steady-state life at a turbine rotor inlet gas temperature of 2200° F (1478 K) at conditions of Mach 3 flight at an altitude of 75 000 feet (22.8 km).

A total of 8 vane and 7 blade cooling designs were evolved. Local metal temperatures were determined for these airfoils at steady-state design point conditions, transient engine conditions, and at several other conditions. These included: changes in blade and vane materials, changes in metal thermal conductivity, changes in design geometry, changes in combustor gas temperature profile and pattern factor, foreign object damage, and changes in flight conditions and turbine inlet gas temperatures. The vane and blade designs that were evolved are illustrated in figures 4 and 5 and are described in the section CONFIGURATIONS ANALYZED.

Design point metal temperatures and coolant flows. - In order to summarize the results of the analysis, only the extremes in local metal temperatures at approximately the critical midspan region of the airfoil at design point conditions were considered. These temperatures are shown in figure 6 plotted against the cooling air-to-engine air flow ratios that were determined as required with the available cooling air supply pressures. Also shown in the figure as a horizontal band are the allowable metal temperatures for a 1000-hour steady-state life for vanes and blades made of material A (IN-100) as obtained from table II. For blades with and without holes in their surfaces, the allowable maximum temperatures are 1700° and 1792° F (1200 and 1250 K), respectively. The allowable temperatures for the vanes are 1800° and 1840° F (1255 and 1278 K), respectively. Also shown in figure 6 is a line representing the maximum allowable metal temperature of 1600° F (1144 K) for porous materials used for transpiration cooling. This temperature was assumed on the basis of the current state-of-the-art with such materials.

All of the designs were intended to meet these metal temperatures. However, after detail computer analysis of the final designs evolved, two of the vane and blade designs, designs I and III, exceeded the maximum allowable metal temperature by over 50° F (28 K) and/or were restricted from increases in coolant flows and reductions in metal temperatures by coolant pressure drop and Mach number limitations. These two designs, which were convection cooled, are represented in figure 6 by the closed data points. These vanes and blade designs can only be made to meet a 1000-hour life with an improved material of the future, designated as material B, which would have a 100° F (55 K) higher temperature capability than IN-100.

In addition to the maximum allowable metal temperature requirement, the criteria described in reference 1 requires that the metal temperature differences across the airfoils not exceed 145° F (81 K) and 200° F (110 K) for the vanes and blades, respectively.

All the rotor blade designs evolved had metal temperature differences less than allowed by the criteria. Stator vane designs I, II, IV, and VIII has metal temperature differences less than allowed but the others exceeded the allowed difference. Refinements of these designs to correct this deficiency were not made. It was assumed, however, that adjustment in the cooling air flow distribution and geometry of all the designs except design VII, perhaps, could probably be made such as to meet the allowed metal temperature difference.

On comparing the various cooling configurations and cooling methods, figure 6 indicates that the spread in required coolant flows is not large. The cooling air-to-engine air flow ratios ranged from 0.027 to 0.033 for six of the vane designs and 0.025 and 0.029 for the five blade designs which were essentially able to meet the temperature criteria for the 1000-hour life with material A. The vanes and blades of design II required the least flow. These blade and vane designs were impingement cooled at the leading edges and convection cooled by horizontal flow of the air around the periphery of the airfoils.

Vane design IV, which was transpiration cooled, required the most flow to meet a maximum metal temperature of 1600° F (1144 K). The blade design which required the most flow was design V. This blade design was completely film cooled by a multiplicity of small holes distributed over the airfoil surface.

More detailed information on local metal temperatures and their distribution at each of several span positions of each vane and blade can be obtained from references 2 to 4. These references also provide data on the coolant flow distributions within the individual designs including such information as the cooling air inlet pressures and temperatures, local cooling air exit pressure, temperatures, and Mach numbers and pressure ratios across openings such as the film cooling holes. The information on the convection cooled designs I, II, and III is presented in reference 2; that for the transpiration cooled vane and blade design IV and film cooled vane and blade design V is presented in reference 3; and that for the other vane and blade designs which utilized combined methods of cooling is presented in reference 4.

Effect of changes in materials. - The analysis of references 2 and 4 determined the reduction in coolant flow that can result when the vanes and blades designed for use with material A are made of improved materials of the future that would have a 100° and 200° F (55 and 111 K) higher temperature capability than material A. These improved materials were designated materials B and C, respectively. The required coolant flows for the airfoil cooling designs which met the allowable metal temperature criteria for a steady-state life of 1000-hours with material A and the coolant flows required with these changes in materials is shown in table III. Improving the material in general resulted in a greater percentage reduction of coolant flow for the blades than for the vanes. The reductions in coolant flow by changing from material A to B, for example, ranged from about 8 to 29 percent for the vanes and 26 to 42 percent for the blades. The airfoil designs most sensitive to this change of material were design VIII for the vanes and design VII for the blades.

The effects on coolant flow requirements of raising or lowering the allowable temperature of the porous material of the transpiration cooled designs about the assumed allowable metal temperature of 1600° F (1144 K) are shown in table IV. The table indicates that if problems occurred in the operation of the airfoils, such as excessive oxidation of the material for example, which would require a reduction in the design metal temperature by 200° F (110 K), large increases in cooling air flow would be required. The coolant flow would need to be increased by about 118 and 198 percent for the vanes and blades, respectively. On the other hand, if improvements in the oxidation and strength properties of porous material can be accomplished in the future, such as to permit the use of allowable design temperatures of 1800° F (1255 K) instead of 1600° F (1144 K), the cooling air requirements can be reduced by 29 and 12 percent for the vanes and blades, respectively.

A comparison of the cooling air requirements for the transpiration cooled vane design at 1800° F (1255 K) with that for the vanes cooled by other methods and having the same allowable metal temperature can be made by comparing the data in table IV with the data for material A in table III. These data, as would be expected, indicate that the transpiration cooled vane would require the least cooling air flow. The transpiration cooled vane design, however, would require only about 9 percent less flow than the strut insert convection cooled vane (design II) which had the lowest coolant flow requirement of the vane designs in table III. It is, however, not practical to expect transpiration cooled airfoils to operate at the same metal temperatures as solid-wall convection cooled airfoils. One reason is that the greater exposed metal surface area of porous walls compared to solid walls would increase oxidation problems, particularly that of clogging of the small coolant passages or pores. Another reason is that porous materials cannot be expected to have the strength of solid materials.

The data in table III showing the reductions in coolant flow that occur with changes in material (allowable metal temperatures) were determined for fixed cooling geometries in which restrictive orifices at the airfoil coolant supply inlet were assumed to exist. This procedure was used because of the large amount of design and calculation effort that would be required to modify each of the designs to the new conditions. The procedure used as a result does not provide the optimum geometry for each change in condition. As a consequence, the procedure gives required coolant flow values, as shown in table III, that are slightly higher than would be required if each cooling configuration was specifically redesigned or modified for a new allowable metal operating temperature. The data in table IV showing changes in cooling air flow requirements for the transpiration cooled airfoils (design IV) with changes in allowable metal temperatures was accomplished by changing both the permeabilities and metering orifices within the airfoils.

Effect of changes in airfoil metal thermal conductivity. - The effect on metal temperatures of increasing the thermal conductivity of vane and blade design II at design point flight conditions is shown in table V. The increase in thermal conductivity was obtained by assuming a change in airfoil material from IN-100 to TD nickel. The conductivities of these materials at a typical operating temperature of 1700° F (1200 K) are 11.8 and 28.0 Btu per hour per foot °F (20.4 and 48.4 J/(sec)(m)(K)), respectively.

The results obtained indicate that this large increase (137 percent) in conductivity had only a small effect in decreasing metal temperatures. The maximum leading edge temperatures at the midspan decreased by only 17° and 9° F (9 and 5 K) for the vanes and blades, respectively. The maximum trailing edge temperatures decreased by only 8° and 2° F (4 and 1 K), respectively.

Effect of changes in airfoil geometries. - A summary of the effects on metal temperatures and coolant flows of assumed variations in the design geometries of the airfoils is presented in table VI. The changes that were assumed to occur were as follows:

(1) A ±50 percent change in the permeability of the porous walls of vane and blade design IV
(2) A ±12.5 to ±25 percent change in the discharge flow areas of the film cooling holes of vane and blade design V
(3) A ±20 and a ±30 percent change in the discharge flow areas of vane and blade designs VI, respectively
(4) A ±20 percent change in the discharge flow areas of vane and blade design VII
(5) A ±20 percent change in the impingement hole flow areas within vane design VIII

The results shown in the table indicate that the changes in airflow caused by these changes in geometry had a large effect on metal temperatures, particularly those at the leading edges.

A comparison of the various vane and blade designs on the basis of sensitivity of metal temperature change to a given coolant flow change indicated that the trailing edge temperatures of all the vanes were about equally sensitive to cooling airflow change. The leading edge temperatures of vane designs IV, V, and VI, however, were more sensitive to airflow change than designs VII and VIII. The latter two vane designs had about one-third and one-half as large a metal temperature change with flow as the other designs.

The trailing edge temperatures of blade designs IV, V, and VI were about equally sensitive to airflow change. The trailing edge of blade design VII, however, was only about one-half as sensitive to airflow change. The leading edge temperatures of blade designs V and VI were equally sensitive to airflow change. The leading edge temperatures of these designs, however, were about twice as sensitive to airflow change as blade designs IV and VII.

Effect of changes in gas temperature profiles. - A summary of the effects of changing the gas temperature profiles at design point flight conditions on maximum leading and trailing edge metal temperatures at the midspan of 6 vane designs (II and IV to VII) and 4 blade designs (II, IV, VI, and VII) is shown in table VII. The results obtained with blade design V and presented in reference 3 were found to be in error and are not presented herein. For the vanes, the assumed higher gas temperature profile had a hot spot temperature 55° F (30 K) higher than design and was equivalent to a combuster pattern factor increase from 0.20 to 0.25. For the blades, two different gas profiles were assumed: the higher temperature profile had a maximum gas temperature 40° F (22 K) higher than design, and the lower temperature profile had a maximum temperature 40° F (22 K) lower than design. These profiles are shown in figure 2.

The table indicates that for the assumed changes in maximum gas temperatures, relatively large changes in metal temperatures occurred. The maximum metal temperatures at the leading and trailing edge of the group of the most sensitive vane designs, designs V, VI, and VIII, increased by 47 to 60 percent of the change in hot spot gas temperature. The other vane designs analyzed, designs II, IV, and VII, had increases in

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metal temperatures from 9 to 44 percent of the gas temperature change.

The maximum metal temperatures at the leading and trailing edges of the most sensitive blade design, design VI increased by as much as 68 percent of the maximum gas temperature increase assumed, and decreased by as much as 60 percent of the gas temperature decrease. Three of the other blade designs analyzed, design II, IV, and VII, had changes in these metal temperatures of from 37 to 55 percent of the gas temperature changes.

Effect of foreign object damage. - The effect of foreign object damage on metal tempreatures and coolant flows was determined only for design II vanes and blades. The analysis was applied to these airfoils operating at design point conditions. The damages considered were a dent in the surface, clogging of internal flow passages, and a hole in the surface. For the vane, the dent was assumed to occur at the midspan suction surface just aft of the stagnation point. The extent of the dent was assumed to be such as to restrict all the air from flowing through the central one-third of the span at the leading edge region. The clogging was assumed to occur to one-third of the impingement holes near the vane tip.

The hole in the airfoil was assumed to be 1/8 inch (3.2 mm) in diameter and located on the suction surface at the root and extend through the wall into the cavity supplying air for the impingement holes. For the blades, the dent and hole were assumed to be in similar locations to that on the vanes. The dent on the blade, however, was assumed to reduce the cooling flow area by a factor of two-thirds in the region of the dent rather than completely blocking the flow. The dent in the blade was assumed to be about 0.9 inch (23 mm) long in the spanwise direction and 0.4 inch (10 mm) wide in the chordwise direction.

A summary of the effects of these assumed foreign object damages on (1) the maximum leading edge metal temperatures at the midspan and tip, (2) the maximum trailing edge metal temperatures at the midspan, and (3) the coolant flow rate is shown in table VIII.

The dent on the vane increased the maximum leading edge temperature from 1832° F (1270 K) to 1951° F (1340 K). The cooling flow rate for this case was not significantly reduced. The redistribution of the air that occurred, however, reduced the maximum midspan trailing edge temperature from 1833° F (1270 K) to 1797° F (1250 K). The effect of the assumed dent on the blade was to increase the maximum temperature at the leading edge at the midspan from 1716° F (1210 K) to 1743° F (1225 K). The effect on maximum trailing edge temperature and cooling air flow rate was small.

The plugging of the impingement holes (which was considered only for the vanes) resulted in an increase in the maximum leading edge metal temperature at the tip from 1815° F (1260 K) to 2066° F (1400 K). This increase in metal temperature would obviously result in a reduced life. The effect of the assumed plugging at the tip had no sig-

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nificant effect on coolant flow rate. The redistribution of the flow within the airfoil, however, resulted in reductions in maximum leading and trailing edge temperatures at the midspan of the vanes by 35° F (19 K) and 49° F (27 K), respectively.

The effects on the 1/8 inch (3.2 mm) diameter hole in vanes and blades was found to be small. The maximum leading edge temperatures at the midspans increased by about 12° F (7 K) while those at the trailing edge decreased only slightly. The presence of the hole increased the coolant flow to the vanes and blades by 5 and 3 percent, respectively.

Effect of changes in flight and turbine inlet gas temperature. - A summary of changes in maximum midspan metal temperatures at the leading and trailing edges, and in coolant flows that result when selected airfoil cooling designs are subjected to conditions of flight and turbine inlet gas temperature other than design are shown in table IX. The conditions shown in the table include flight at Mach numbers from 2.5 to 3.5 at altitudes from 65 000 to 85 000 feet (19.9 to 25.9 km) with turbine rotor inlet gas temperatures from 2000° to 2500° F (1367 to 1644 K). The vane and blade designs analyzed at these conditions were designs II, IV, and V. Local metal temperature distributions for entire cross sections of these designs at these and for lower flight Mach number and altitude conditions are presented in references 2 to 4.

The results of the analysis of the vane designs indicates the following:

(1) Vane design II can meet material A temperature criteria for all of the off-design conditions considered, except that of flight at Mach 3.5 at 85 000 feet (25.9 km) with a turbine rotor inlet gas temperature of 2200° F (1478 K). For these conditions, vane metal temperatures exceeded the maximum allowable value because of the limitation set for the cooling airflow by the air supply pressure and the fixed geometry of the design.

(2) Vane design IV can be made to meet the allowable maximum metal temperature for porous walls of 1600° F (1145 K) at turbine rotor inlet gas temperatures of 2200° F (1478 K) or less at flight Mach numbers of 2.5 and 3.0. At two of these conditions, as indicated in table IX, the vanes had to be overcooled to prevent inflow of combustion gases. Operation at the other flight conditions at higher turbine rotor inlet gas temperatures, or operation at a gas temperature of 2000° F (1367 K) at Mach 3.5, at 85 000 feet (25.9 km) was limited by the cooling air supply pressure and flows which resulted in metal temperatures substantially exceeding the assumed allowable limit.

(3) Vane design V was analyzed for capability of operation for the case of the vanes made of material B. The results indicate that the design can essentially meet material B temperature limit of 1900° F (1311 K) for flight Mach numbers from 2.5 to 3.5 at turbine rotor inlet gas temperatures of 2200° F (1478 K) or less, and up to a Mach number 2.5 when the inlet gas temperature is 2400° F (1590 K). At several of these conditions, as indicated in table IX, portions of the vanes needed to be overcooled so as to prevent inflow of combustion gases. Operation at the other flight conditions and at higher turbine rotor inlet gas temperatures was limited by cooling air supply pressures and

flows which resulted in metal temperatures substantially exceeding the material B allowable temperature limit.

The results of the analysis of the blades indicate the following:

(1) Blade design II can essentially meet material A temperature criteria for all of the off-design conditions considered except flight at Mach 3.5 at 85 000 feet (25.9 km) and at a turbine rotor inlet gas temperature of 2200° F (1478 K). The cooling air supply pressure and flow at these conditions were the limiting factors and resulted in metal temperatures exceeding the maximum allowable value.

(2) Blade design IV could not meet the allowable porous wall metal temperature for any of the conditions other than design within even 50° F (27 K) except for turbine inlet gas temperatures less than design at design Mach number or at lower flight Mach numbers at design turbine inlet gas temperature.

(3) Blade design V was analyzed for the capability of operating at other flight conditions assuming it was made of material B. The results indicate that the design can be cooled to meet the allowable temperature criteria for this material for the following conditions: flight at Mach 2.5 at turbine rotor inlet gas temperatures up to 2500° F (1644 K), flight at Mach 3.0 at turbine rotor inlet gas temperatures up to 2400° F (1590 K), and flight at Mach 3.5 at turbine inlet gas temperatures up to and slightly above 2200° F (1478 K). At the other conditions shown in table IX, the cooling air supply pressure and fixed geometry of the design resulted in excessive blade metal temperatures.

Transient metal temperatures. - Although analysis and experience described in appendix G of reference 1 has indicated that a reasonably satisfactory basis is available for specifying initial values of vane and blade metal temperatures that will provide a desired life at steady-state engine conditions, no such criteria has been established for these parts under conditions of transient engine operation. Of the many factors influencing the life of airfoils at these conditions, the transient chordwise metal temperature differences that occur during engine acceleration and deceleration are of prime importance. Although a detailed stress analysis which considers the cyclic temperature history of each design would be required to predict airfoil cyclic life, a knowledge of representative transient chordwise metal temperature differences (which are indicative of strain to which the airfoil is subjected) can provide an initial basis for comparing airfoil designs. The transient metal temperatures which were considered as having the most significance on the cyclic lives of airfoils were the temperature differences between the leading or trailing edges and the interior of the airfoil at the critical midspan region. Calculated values of leading edge, trailing edge, and interior metal temperatures with time resulting from engine transient operation for the various airfoils analyzed is presented in references 2 to 4. In this report only the values of maximum positive or negative metal temperature differences between leading or trailing edges and interior that occurred during acceleration and deceleration transients, respectively, are presented.

A tabulation of these calculated maximum positive and negative transient metal temperature differences and the times at which they occurred for the vane and blade designs that were analyzed is presented in figure 7. Figures 7(a) and (b) show the temperature difference between the leading edge and interior and trailing edge and interior, respectively, for the vanes. Figures 7(c) and (d), respectively, show these data for the blades. The data are presented for two rates of engine accelerations and decelerations. The times to affect a maximum change in gas temperature in the process of accelerating from idle to take-off conditions, and decelerating from this condition to idle were assumed to take 8 and 5 seconds, respectively, for the slow transient case, and 4 and 2. 5 seconds for the fast transient case.

The results shown in figure 7 indicate that large excursions in metal temperature differences occur during engine transients. The metal temperature differences between the leading edge and interior for the configurations analyzed ranged from 141° to 648° F (78 to 360 K) during accelerations and 38° to -747° F (21 to -415 K) during engine decelerations. The airfoils that had these extremes in temperature differences during engine accelerations were vanes (designs VIII and V), and those during engine decelerations were blades (designs VI and VII).

The metal temperature differences between the trailing edge and interior ranged from 70° to 680° F (39 to 378 K) during engine accelerations and 65° to -489° F (36 to -272 K) during decelerations. The airfoils that had these extremes in temperature differences during accelerations were vanes (designs VI and IV), and those during engine decelerations were a blade (design VI) and a vane (design V).

The results in figure 7 also indicate that the effects of the two transient times on the magnitude of the maximum positive and negative temperature differences, in general were not large, ranging from 0° to 183° F (0 to 102 K). Vane design IV and blade design VII were the airfoils most sensitive to engine transient times. The temperatures most sensitive on this vane design were the differences between the trailing edge and interior and those for the blade design were between the leading edge and interior. These temperature differences on the vane and blade were 175° F (97 K) and 183° F (102 K), respectively, lower for fast engine accelerations than for slow engine accelerations.

The primary effect of the engine transient rate was to change the time at which the maximum metal temperature difference occurred. These temperature differences were, in general, reached within about 1. 5 seconds of the time required for the maximum change in gas temperature. The exceptions were vane designs IV, V, and VII and blade designs VI and VII. Vane designs IV and V and blade design VI required about 4, 5, and over 10 seconds, respectively, after the maximum gas temperature difference occurred to reach their maximum chordwise metal temperature differences. Vane design VII and blade design VII had maximum metal temperature differences occur at 4 and 3 seconds,

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respectively, ahead of the time at which the maximum gas temperature differences were reached.

Although the information in the preceding paragraphs describe the transient operating characteristics of the various blade and vane designs, the most significant information that can be obtained from figure 7 is the absolute magnitude of sum of the positive and negative metal temperature differences that occur during a complete engine acceleration-deceleration cycle. The total metal temperature difference range covered during one of these cycles is representative of the strain range to which the airfoil is subjected. The maximum metal temperature difference range experienced by each vane and blade design is therefore used as a basis for initially comparing cyclic performance of the various designs. This comparison is shown in figure 8 for both fast and slow acceleration-deceleration cycles. Figures 8(a) and (b) show the comparisons of the metal temperature difference range between the leading edge and interior and trailing edge and interior, respectively, for the vane designs, and figures 8(c) and (d), respectively, for the blade designs. These data indicate that vane design VIII was among the two designs (II and VIII) which had the smallest metal temperature difference range between the leading edge and interior and was among the three vane designs (VI, VII, and VIII) which had the smallest temperature difference range between the trailing edge and interior. The vane designs with the largest temperature difference ranges were designs IV and V. These designs had temperature difference ranges that were about twice as large as those of design VIII.

A comparison of the data for the blades indicate that design VI clearly had the smallest temperature difference ranges. Blade design VII had considerably larger metal temperature difference ranges. This was particularly the case for the temperature difference range between the leading edge and interior. This metal temperature difference range for blade design VII was as much as 5 times that of design VI for the fast engine accelerations-decelerations cycles.

<u>Predicted lives of selected configurations.</u> - A summary of the predicted lives for the four cooling designs analyzed, (vane design VI and blade designs II, V, and VI), is shown in table X. The lives are shown for conditions of: (1) steady-state creep or stress rupture at sea level take-off or cruise, (2) gradual repetitions of steady-state sea level take-off or cruise conditions such as to exclude transient effects, and (3) engine transient cycles (engine accelerations and decelerations between idle and take-off). Blade designs V and VI were assumed to have a tapered shell. The shell of blade design II was assumed untapered. The table indicates that steady-state lives of at least 1000 hours, the assumed design required life, should be obtained for all of the designs. All of the designs are also able to tolerate an unlimited gradual repetition of steady-state cruise conditions. Only one of the airfoil designs, blade design VI, however, is expected to tolerate unlimited repetitions of the steady-state sea level take-off conditions. Vane de-

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sign VI, blade design II and V are limited to 1000, 2600, and 2000 repetitions of steady-state take-off conditions. Assuming for one of these repetions would occur for each flight and that the flight time for a supersonic transport would be of the order of 2 to 3 hours, even the airfoil cooling design with the smallest allowable number of repetitions (1000) should have a total airfoil life of at least 2000 hours. As a result, the data in table X indicates that attainment of the assumed 1000-hours life requirement would only be limited by the tolearnce of the airfoils to engine transient operations. Vane design VI, blade design II, and V have predicted lives based on crack initiation of 100, 200, and 500 cycles, respectively. Blade design VI, however, was not limited by the engine transient operations.

The detailed stress analysis of the airfoils under transient engine operation indicated that the combination of stresses and temperatures at the leading edge regions of the designs limited their lives. Examination of the range of metal temperature differences between the leading edges and blade interiors caused by the engine transients and shown in figure 8(c) indicates the general level of strain experience by the blades. This figure indicates that the order of increasing strain range and decreasing life for the three blade designs analyzed would be designs VI, II, and V. The use of this gross comparison of the expected cyclic lives of the blades agrees with that of the detailed analysis in that blade VI is best, but reverses the order of the other two designs. However, as can be seen in figure 8(c), the differences in the temperature ranges between two designs is not large.

Although all blade designs did not completely meet the assumed required 1000-hour life, the results obtained indicate that the generalized design criteria as summarized in table II is satisfactory for starting initial cooling designs. Additional modifications and interactions of the initial design can be made to adjust local metal temperature, metal temperature differences, and stresses such as to increase the cyclic lives of each of the designs to meet the assumed life requirement. A discussion of specific recommendations of means for improving the lives of each design that was detail stress analyzed is presented in reference 5.

CONCLUSIONS

This report summarizes the results of analysis of various cooled turbine airfoil configurations which would have application to engines of aircraft for high speed flight such as a supersonic transport. The following are the conclusions that were obtained:

1. The high compressor exit bleed air temperature commonly used for reducing the turbine metal temperatures limits the manner in which the air can be ducted within the airfoil and limits the benefits provided by changes in blade design or cooling methods.

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2. The high cooling air supply temperatures and the external gas environment of the cooled airfoils dictate that the following rules, generally be observed in the design of the cooled turbine airfoils:

(a) The flow path of the coolant in the leading and trailing edge regions of con-convection cooled airfoils should be short to avoid excessive heating of air and metal.

(b) Impingement cooling of the leading edge region should be utilized.

(c) Convection cooling of the long trailing edges by means of airflow through slots or holes should be utilized.

(d) Film cooling should be considered as a means of both expelling heated air and supplementing other methods of cooling.

3. The required coolant flows were not much different for the various cooled turbine airfoil configurations or cooling methods utilized or evolved in this analysis. The cooling air-to-engine airflow ratios required for a turbine rotor inlet average gas temperature of 2200° F (1478 K) at flight conditions of Mach 3 at 75 000 feet ranged between 0.027 and 0.033 for six of the first-stage vane designs and 0.025 and 0.029 for five of the first-stage blade designs which were essentially able to meet the metal temperature criteria for 1000-hour life with a material with properties of IN-100.

4. The maximum steady-state metal temperature differences at the midspan of the vanes and blades, in general, were controlled to about 145° F (81 K) and 200° F (111 K), respectively. These temperatures along with allowable maximum metal temperatures were established from analysis and experimental experience to be required for a steady-state life of 1000-hours with a material such as IN-100. The maximum allowable local metal temperatures at the midspan of the blade and vanes were 1700° and 1800° F (1200 and 1255 K), respectively, for airfoils with film cooling holes. Without film cooling holes the allowable temperatures were 1792° and 1840° F (1250 and 1278 K), respectively.

5. The turbine vane design which required the least coolant flow (based on the criteria used for a steady-state 1000-hour life) was found to be a convection cooled design that used impingement cooling of the leading edge and chordwise flow of air through small passages around the periphery with exit of the flow through the trailing edge. A blade design using a similar cooling method also required the least coolant flow of the blade designs evolved.

6. Large excursions in metal temperatures occur with time during engine acceleration and deceleration between idle and take-off conditions. The metal temperature differences between the leading edge and the airfoil interior for the configurations analyzed, for example, ranged from about 141° to 648° F (78 to 360 K) for engine accelerations and 38° to -747° F (21 to -415 K) for decelerations. The maximum temperature differences between the trailing edge to airfoil interiors were found to range between 70° to 680° F

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(39 to 378 K) and 65^{o} to -489^{o} F (36 to -272 K) for the accelerations and decelerations, respectively.

7. The range of the maximum temperature difference experienced by the airfoils between the leading or trailing edges and the interior during the excursions between engine accelerations and decelerations was found to be least for a vane design that utilized a combination of impingement and convection cooling (design VIII) and least for a rotor blade design which utilized a combination of film and convection cooling (design VI). This blade design also had the best predicted low-cycle fatigue life of three blade designs which were detailed stress analyzed.

8. The changes in metal temperatures of the vane and blade leading and trailing edge regions corresponding to the assumed changes in combustor pattern factor or gas temperature profile about the design values were large. The most sensitive group of vane designs had changes in metal temperatures at the leading and trailing edges which were 60 percent of the maximum gas temperature change. These metal temperatures of the most sensitive blade design changed by as much as 68 percent of the gas temperature change. This percentage change in metal temperature with gas temperature was as much as 7 times greater than that for some of the other vane and blade designs.

9. Small decreases in leading and trailing edge temperatures of the order of 17^{o} F (10 K) and 8^{o} F (4 K), respectively, result from increasing airfoil metal thermal conductivity by as much as 137 percent (a change in the airfoil material from IN-100 to TD nickel).

10. Improvements in materials such as to permit increases in their use temperature by 100^{o} F (55 K) over that of IN-100 can result in reductions in required cooling airflow by as much as 29 and 42 percent, respectively, for the vane and blade designs considered. Improvements in the allowable use temperature of transpiration materials from an assumed current allowable value of 1600^{o} F (1144 K) to 1800^{o} F (1255 K) can reduce coolant flows by 29 and 12 percent, respectively, for the vane and blade design considered.

11. Changes in airfoil geometries as might be due to fabrication variations and the resultant changes in coolant flows and metal temperatures indicated that the metal temperatures of some of the designs were from 2 to 3 times as sensitive to coolant flow change as others.

12. The foreign object damages that were assumed to occur to a convection cooled vane and blade design did not have a significant effect in changing total coolant flow, but resulted in flow redistributions within the airfoils which resulted in reductions and in-

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creases in leading edge temperatures of as much as 35° F (19 K) and 250° F (139 K), respectively.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 10, 1968,
126-15-01-02-22.

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TABLE I. - CYCLE DATA FOR FOUR FLIGHT CONDITIONS AND SEA-LEVEL TAKEOFF AND IDLE

Parameter	Altitude, ft (km)					
	65 000 (19.8)	75 000 (22.8)	85 000 (25.9)	Sea-level takeoff	Sea-level idle	45 000 (13.7)
Flight Mach number	2.5	3.0	3.5	0	0	1.2
Engine speed, rpm	5050	5050	5050	5080	3080	5050
Engine inlet total pressure, psia; kN/m^2	12.2; 84.1	15.2; 104.8	18.2; 125.5	14.7; 101.3	14.7; 101.3	5.14; 35.4
Engine inlet total temperature, ^{o}F; K	416; 487	637; 609	896; 753	59; 283	59; 283	43; 279
Engine inlet airflow, lb/sec; kg/sec	208.8; 94.8	182.6; 82.9	151.7; 68.9	475; 215.6	151.6; 68.8	169.9; 169.9
Compressor total-pressure ratio	5.17	3.67	2.56	9.49	2.25	9.684
Compressor discharge total temperature, ^{o}F; K	990; 805	1190; 917	1390; 1028	593; 585	243; 390	575; 575
Compressor discharge total pressure, psia; kN/m^2	63.0; 434	55.6; 383	46.6; 321	139.5; 962	33.1; 228	49.8; 343
Ratio of cooling and recouped leakage airflow to compressor inlet airflow	0.078	0.095	0.128	0.114	0.104	0.115
Compressor efficiency	0.856	0.814	0.751	0.861	0.734	0.849
Combustor total-pressure ratio	0.927	0.919	0.912	0.942	0.931	0.943
Main combustor efficiency	0.983	0.981	0.975	0.980	0.963	0.976
Main fuel air ratio	0.020	0.0169	0.0138	0.0261	0.009	0.0265
Turbine rotor inlet total temperature, ^{o}F; K	2200; 1478	2200; 1478	2200; 1478	2200; 1478	874; 741	2200; 1478
Turbine efficiency	0.896	0.896	0.896	0.896	0.876	0.896
Turbine discharge total temperature (after mixing of cooling air), ^{o}F; K	1621; 1156	1620; 1155	1648; 1171	1600; 1144	641; 612	1588; 1138
Turbine discharge total pressure, psia; kN/m^2	19.80; 136	17.15; 118	15.08; 104	48.89; 337	15.83; 109	17.57; 121

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TABLE II. - METAL TEMPERATURE LIMITS

FOR 1000-HOUR LIFE[a]

(a) Vanes. Maximum shell temperature difference, 145° F (80 K)

Material	Holes	Maximum leading-edge metal temperature		Maximum trailing-edge metal temperature	
		°F	K	°F	K
A	With	1800	1255	1800	1255
	Without	1840	1278	1830	1272
B	With	1900	1311	1900	1311
	Without	1940	1333	1930	1328
C	With	2000	1366	2000	1366
	Without	2040	1389	2030	1383

(b) Blades

Material	Holes	Midspan[b]				Root[c]			
		Maximum metal temperature		Maximum metal temperature difference		Section maximum metal temperature		Maximum metal temperature difference	
		°F	K	°F	K	°F	K	°F	K
A	With	1700	1200	200	111	1718	1210	218	121
	Without	1792	1253	202	112	1725	1214	238	121
B	With	1800	1255	200	111	1818	1265	218	121
	Without	1892	1306	202	112	1825	1269	238	132
C	With	1900	1311	200	111	1918	1321	218	121
	Without	1992	1362	202	112	1925	1325	238	132

[a]Based on design criteria in ref. 1.
[b]Limiting mechanism, cycling.
[c]Limiting mechanism, rupture.

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TABLE III. - COOLING AIRFLOW REQUIREMENTS

FOR 1000-HOUR LIFE FOR FIRST-STAGE

TURBINE VANES AND BLADES

[Flight speed, Mach 3; altitude, 75 000 ft (2.28 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Design	Description	Material		
		A	B	C
		Ratio of cooling air to compressor flow		
Vane II	Strut insert; convection cooled	0.0275	0.0233	0.0190
Vane V	Film cooled	.0299	.0275	.0261
Vane VI	Film and convection cooled	.032	.0277	.0219
Vane VII	Transpiration and convection cooled	.0280	.0219	.0127
Vane VIII	Impingement and convection cooled	.0282	.0201	.0159
Blade II	Strut insert, convection cooled	0.0260	0.0181	0.0140
Blade V	Film cooled	.0295	.0172	.0135
Blade VI	Film and convection cooled	.0289	.0214	.0087
Blade VII	Liquid metal and air convection cooled	.0267	.0154	.0094

TABLE IV. - COOLING AIRFLOW REQUIREMENTS

FOR TRANSPIRATION-COOLED VANE AND

BLADE DESIGNED TO SEVERAL LEVELS

OF METAL TEMPERATURES

[Flight speed, Mach 3; altitude, 75 000 ft (22.8 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Design IV	Average material temperature, °F (K)		
	1400 (1030)	1600 (1140)	1800 (1260)
	Ratio of cooling air to compressor flow		
Vane	0.0719	0.0329	0.0234
Blade	.0793	.0266	.0234

TABLE V. - EFFECT OF CHANGE IN DESIGN AIRFOIL METAL

THERMAL CONDUCTIVITY ON METAL TEMPERATURES

[Flight speed, Mach 3; altitude, 75 000 ft (22.8 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Design	Material	Midspan metal temperature, °F (K)		Ratio of cooling air to engine flow
		Maximum leading edge	Maximum trailing edge	
Vane II	IN 100	1832 (1273)	1833 (1274)	0.0275
	TD nickel	1815 (1264)	1825 (1269)	.0277
Blade II	IN 100	1716 (1209)	1779 (1214)	0.0260
	TD nickel	1707 (1204)	1777 (1243)	.0259

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TABLE VI. - EFFECT OF CHANGES IN DESIGN GEOMETRY ON TURBINE AIRFOIL METAL TEMPERATURES

[Flight speed, Mach 3; altitude, 75 000 ft (22.8 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Design	Geometry	Ratio of cooling air to engine flow	Midspan metal temperature, °F (K)		Design point based on -
			Maximum leading edge	Maximum trailing edge	
Vane IV	Permeability + 50%	0.0379	1573 (1129)	1589 (1138)	Allowable metal temperature of 1600° F (1144 K)
	Design	.0329	1620 (1155)	1613 (1152)	
	Permeability - 50%	.0228	1822 (1268)	1692 (1195)	
Vane V	Discharge areas + 12.5%	0.0310	1846 (1281)	1763 (1235)	Use of material B
	Design	.0276	1916 (1320)	1828 (1271)	
	Discharge areas - 12.5%	.0229	1992 (1362)	1889 (1305)	
Vane VI	Discharge areas + 20%	0.0352	1787 (1248)	1769 (1238)	Use of material C
	Design	.0320	1844 (1280)	1837 (1276)	
	Discharge areas - 20%	.0277	1814 (1263)	1896 (1309)	
Vane VII	Discharge areas + 20%	0.0292	1561 (1123)	1664 (1180)	Use of material C
	Design	.0280	1603 (1146)	1647 (1170)	
	Discharge areas - 20%	.0219	1664 (1180)	1700 (1200)	
Vane VIII	Impingement hole areas + 20%	0.0352	1818 (1265)	1728 (1215)	Use of material C
	Design	.0282	1840 (1278)	1791 (1250)	
	Impingement hole areas - 20%	.0214	1870 (1294)	1865 (1291)	
Blade IV	Permeability + 50%	0.0321	1571 (1128)	1554 (1119)	Allowable metal temperature of 1600° F (1144 K)
	Design	.0266	1606 (1148)	1605 (1147)	
	Permeability - 50%	.0185	1681 (1189)	1699 (1199)	
Blade V	Discharge areas + 25%	0.0223	1709 (1204)	1723 (1213)	Use of material B
	Design	.0166	1738 (1221)	1763 (1235)	
	Discharge areas - 25%	.0137	1805 (1258)	1788 (1249)	
Blade VI	Discharge areas + 30%	0.0298	1686 (1192)	1729 (1216)	Use of material C
	Design	.0289	1687 (1193)	1736 (1220)	
	Discharge areas - 30%	.0262	1692 (1195)	1770 (1239)	
Blade VII	Discharge areas + 20%	0.0320	1756 (1231)	1775 (1241)	Use of material C
	Design	.0267	1773 (1240)	1801 (1256)	
	Discharge areas - 20%	.0214	1820 (1266)	1837 (1276)	

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TABLE VII. - EFFECT OF CHANGES IN DESIGN COMBUSTOR GAS TEMPERATURE PROFILE ON TURBINE AIRFOIL METAL TEMPERATURES

[Flight speed, Mach 3; altitude, 75 000 ft (22.8 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Design	Combustor gas temperature profile[a]	Midspan metal temperature, °F (K)		Ratio of cooling air to engine flow
		Maximum leading edge	Maximum trailing edge	
Vane II	Design	1832 (1273)	1833 (1274)	0.0275
	Higher temperature	1849 (1283)	1838 (1277)	.0278
Vane IV	Design	1620 (1155)	1613 (1151)	0.0329
	Higher temperature	1630 (1161)	1637 (1165)	.0325
Vane V	Design	1916 (1320)	1828 (1271)	0.0276
	Higher temperature	1949 (1338)	1854 (1285)	.0276
Vane VI	Design	1844 (1280)	1837 (1275)	0.0320
	Higher temperature	1873 (1296)	1863 (1290)	.0318
Vane VII	Design	1603 (1146)	1647 (1170)	0.0280
	Higher temperature	1619 (1155)	1663 (1179)	.0276
Vane VIII	Design	1840 (1278)	1791 (1250)	0.0282
	Higher temperature	1867 (1293)	1818 (1265)	.0282
Blade II	Design	1716 (1209)	1779 (1244)	0.0260
	Higher temperature	1736 (1220)	1800 (1255)	.0261
	Lower temperature	1694 (1197)	1757 (1231)	.0260
Blade IV	Design	1606 (1148)	1605 (1147)	0.0266
	Higher temperature	1621 (1156)	1620 (1155)	.0267
	Lower temperature	1591 (1139)	1589 (1138)	.0276
Blade VI	Design	1687 (1193)	1736 (1216)	0.0289
	Higher temperature	1713 (1207)	1763 (1235)	.0289
	Lower temperature	1667 (1181)	1712 (1206)	.0290
Blade VII	Design	1773 (1240)	1801 (1256)	0.0267
	Higher temperature	----------	1823 (1268)	.0267
	Lower temperature	----------	1778 (1243)	.0267

[a]See fig. 2.

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TABLE VIII. - EFFECTS OF FOREIGN OBJECT DAMAGE

ON AIRFOIL METAL TEMPERATURES AT

ENGINE DESIGN CONDITIONS

[Flight speed, Mach 3; altitude, 75 000 ft (22.8 km); turbine rotor-inlet gas temperature, 2200° F (1478 K).]

Damage	Midspan metal temperature, °F (K)		Maximum leading edge temperature at tip, °F (K)	Ratio of cooling air to engine flow
	Maximum leading edge	Maximum trailing edge		
Vane desigh II				
None (design)	1832 (1273)	1833 (1274)	1815 (1264)	0.0275
Dent on surface	1951 (1339)	1797 (1254)	1890 (1305)	.0271
Plugging	1797 (1253)	1784 (1246)	2066 (1403)	.0277
Hole in surface	1834 (1274)	1815 (1264)	1819 (1266)	.0289
Blade design II				
None (design)	1716 (1209)	1779 (1244)	1704 (1202)	0.0260
Dent on surface	1743 (1224)	1777 (1243)	1713 (1207)	.0251
Hole in surface	1728 (1215)	1778 (1243)	1713 (1207)	.0268

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TABLE IX. - SUMMARY OF METAL TEMPERATURES AND COOLANT FLOWS

DUE TO CHANGES IN FLIGHT AND ENGINE CONDITIONS

(a) Vane designs

Mach number	Altitude, ft (km)	Turbine inlet temperature, ^{o}F (K)	Maximum midspan temperature, ^{o}F (K)		Ratio of cooling air-to-engine flow	Comments
			Leading edge	Trailing edge		
Vane design II - strut insert, convection cooled						
2.5	65 000 (19.8)	2000 (1367)	1726 (1214)	1768 (1238)	0.0155	----------------
3.0	75 000 (22.8)	2000 (1367)	1776 (1242)	1823 (1268)	.0192	----------------
3.5	85 000 (25.9)	2000 (1367)	1780 (1244)	1813 (1263)	.0246	----------------
2.5	65 000 (19.8)	2200 (1478)	1814 (1263)	1844 (1280)	.0219	----------------
3.5	85 000 (25.9)	2200 (1478)	1909 (1316)	1954 (1341)	.0264	(a)
3.0	75 000 (22.8)	2200 (1478)	1832 (1273)	1833 (1274)	.0275	Design (material A)
Vane design IV - transpiration cooled						
2.5	65 000 (19.8)	2000 (1367)	1462 (1068)	1397 (1032)	0.0318	(b)
3.0	75 000 (22.8)	2000 (1367)	1600 (1144)	1533 (1107)	.0309	----------------
3.5	85 000 (25.9)	2000 (1367)	1681 (1184)	1662 (1179)	.0315	(a)
2.5	65 000 (19.8)	2200 (1478)	1553 (1118)	1476 (1075)	.0318	(b)
3.5	85 000 (25.9)	2200 (1478)	1777 (1243)	1752 (1229)	.0315	(a)
2.5	65 000 (19.8)	2400 (1589)	1673 (1185)	1580 (1133)	.0318	
3.0	75 000 (22.8)	2400 (1589)	1715 (1208)	1706 (1203)	.0329	(a)
3.5	85 000 (25.9)	2400 (1589)	1880 (1300)	1849 (1283)	.0315	↓
2.5	65 000 (19.8)	2500 (1644)	1728 (1215)	1627 (1159)	.0318	↓
3.0	75 000 (22.8)	2500 (1644)	1763 (1235)	1754 (1230)	.0329	↓
3.5	85 000 (25.9)	2500 (1644)	1934 (1330)	1899 (1310)	.0315	↓
3.0	75 000 (22.8)	2200 (1478)	1620 (1155)	1613 (1152)	.0329	Design (allowable metal temperature, ~1600° F (1144 K))
Vane design V - film cooled						
2.5	65 000 (19.8)	2000 (1367)	1729 (1216)	1565 (1125)	0.0282	(b)
3.0	75 000 (22.8)	2000 (1367)	1835 (1275)	1712 (1207)	.0257	(b)
3.5	85 000 (25.9)	2000 (1367)	1898 (1310)	1819 (1266)	.0244	----------------
2.5	65 000 (19.8)	2200 (1478)	1855 (1286)	1664 (1180)	.0281	(b)
3.5	85 000 (25.9)	2200 (1478)	1901 (1311)	1910 (1316)	.0276	----------------
2.5	65 000 (19.8)	2400 (1589)	1909 (1316)	1787 (1248)	.0296	(b)
3.0	75 000 (22.8)	2400 (1589)	1943 (1335)	1944 (1335)	.0286	(a)
3.5	85 000 (25.9)	2400 (1589)	2047 (1393)	2055 (1397)	.027	↓
2.5	65 000 (19.8)	2500 (1644)	1941 (1334)	1855 (1286)	.030	↓
3.0	75 000 (22.8)	2500 (1644)	2029 (1383)	1944 (1335)	.0286	↓
3.5	85 000 (25.9)	2500 (1644)	2134 (1441)	2130 (1439)	.0268	↓
3.0	75 000 (22.8)	2200 (1478)	1916 (1320)	1828 (1271)	.0276	Design (material B)

[a] Limited by cooling air supply and exceeds maximum allowable metal temperature.

[b] Overcooled to prevent inflow of combustion gases.

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TABLE IX. - Concluded. SUMMARY OF METAL TEMPERATURES AND COOLAND FLOWS

DUE TO CHANGES IN FLIGHT AND ENGINE CONDITIONS

(b) Blade designs

Mach number	Altitude, ft (km)	Turbine inlet temperature, °F (K)	Maximum midspan temperature, °F (K)		Ratio of cooling air-to-engine flow	Comments
			Leading edge	Trailing edge		
Blade design II - strut insert, convection cooled						
2.5	85 000 (25.9)	2000 (1367)	1704 (1202)	1805 (1258)	0.0062	----------------
3.0	75 000 (22.8)	2000 (1367)	1704 (1202)	1792 (1251)	.0102	----------------
3.5	85 000 (25.9)	2000 (1367)	1757 (1231)	1814 (1263)	.0122	----------------
2.5	65 000 (19.8)	2200 (1478)	1694 (1196)	1777 (1243)	.0201	----------------
3.5	85 000 (25.9)	2200 (1478)	1811 (1261)	1862 (1290)	.0220	(a)
3.0	75 000 (22.8)	2200 (1478)	1716 (1209)	1779 (1244)	.0260	Design (material A)
Blade design IV - transpiration cooled						
2.5	65 000 (19.8)	2000 (1367)	1663 (1179)	1643 (1168)	0.0151	(a)
3.0	75 000 (22.8)	2000 (1367)	1616 (1150)	1611 (1150)	.0207	----------------
3.5	85 000 (25.9)	2000 (1367)	1657 (1176)	1611 (1150)	.0254	(a)
2.5	65 000 (19.8)	2200 (1478)	1643 (1168)	1648 (1171)	.0187	----------------
3.5	85 000 (25.9)	2200 (1478)	1745 (1225)	1755 (1230)	.0254	(a)
2.5	65 000 (19.8)	2400 (1589)	1654 (1174)	1662 (1179)	.0232	(a)
3.0	75 000 (22.8)	2400 (1589)	1706 (1203)	1706 (1203)	.0266	(a)
3.5	85 000 (25.9)	2400 (1589)	1844 (1280)	1835 (1275)	.0254	
2.5	65 000 (19.8)	2500 (1644)	1659 (1177)	1669 (1183)	.0250	
3.0	75 000 (22.8)	2500 (1644)	1757 (1231)	1757 (1231)	.0266	
3.5	85 000 (25.9)	2500 (1644)	1900 (1311)	1890 (1305)	.0254	
3.0	75 000 (22.8)	2200 (1478)	1606 (1148)	1605 (1147)	.0266	Design (allowable metal temperature, ~1600° F)
Blade design V - film cooled						
2.5	65 000 (19.8)	2000 (1367)	1696 (1198)	1606 (1147)	0.0131	(b)
3.0	75 000 (22.8)	2000 (1367)	1727 (1215)	1679 (1188)	.0131	(b)
3.5	85 000 (25.9)	2000 (1367)	1795 (1253)	1745 (1230)	.0110	----------------
2.5	65 000 (19.8)	2200 (1478)	1750 (1228)	1722 (1212)	.0153	----------------
3.5	85 000 (25.9)	2200 (1478)	1776 (1242)	1795 (1252)	.0220	----------------
2.5	65 000 (19.8)	2400 (1589)	1722 (1212)	1790 (1250)	.0223	(b)
3.0	75 000 (22.8)	2400 (1589)	1758 (1232)	1803 (1257)	.0285	----------------
3.5	85 000 (25.9)	2400 (1589)	1852 (1284)	1871 (1295)	.0260	(a)
2.5	65 000 (19.8)	2500 (1644)	1738 (1221)	1816 (1264)	.0255	----------------
3.0	75 000 (22.8)	2500 (1644)	1822 (1268)	1864 (1291)	.0280	(a)
3.5	85 000 (25.9)	2500 (1644)	1899 (1310)	1921 (1323)	.0258	(a)
3.0	75 000 (22.8)	2200 (1478)	1738 (1221)	1763 (1235)	.0166	Design (material B)

[a]Limited by cooling air supply and exceeds maximum allowable metal temperature.
[b]Overcooled to prevent inflow of combustion gases.

[redacted]

TABLE X. - PREDICTED VANE AND BLADE LIVES

[Cruise life based on repetition of steady-state conditions, unlimited; sea-level takeoff steady-state creep or rupture life, 100 000 hr.]

Design	Description	Cyclic life[a] due to engine transients, cycles	Sea-level takeoff life based on repetition of steady-state conditions, cycles	Cruise steady-state creep or rupture lives, hr (b)
Vane VI	Film and convection cooled	100	1000	4000
Blade II	Strut insert; convection cooled	200	2600	5000
Blade V	Film cooled	500	2000	2000
Blade VI	Film and convection cooled	Unlimited	Unlimited	3500

[a]Based on crack initiation.

[b]Value for blades, stress rupture, and for vanes, 0.5% creep life.

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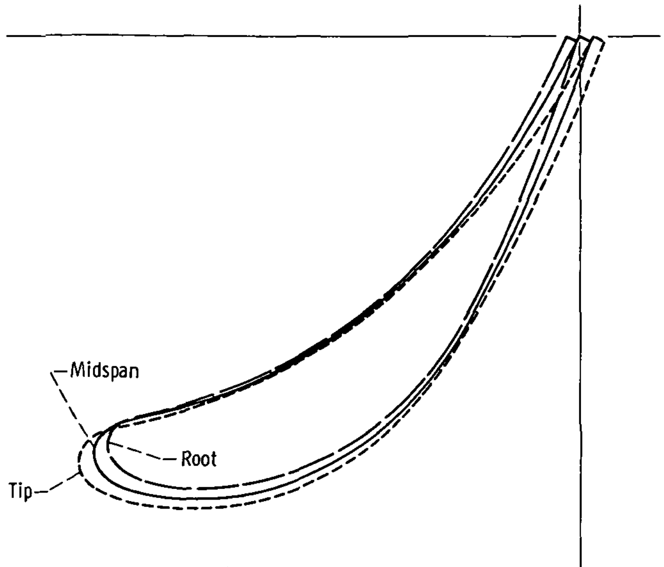
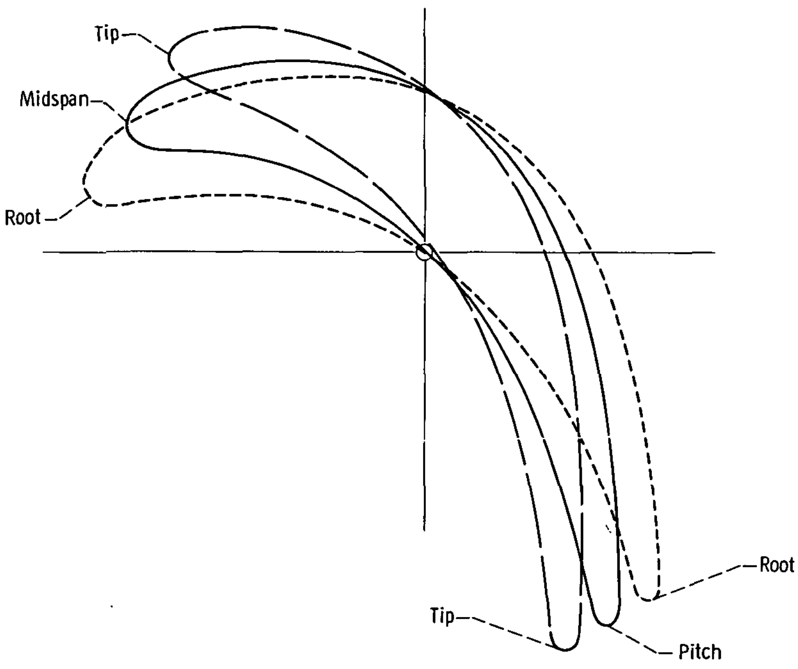


(a) First-stage vane (full scale).



(b) First-stage blade (twice scale).

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Figure 1. - Turbine airfoil aerodynamic profiles.

[redacted]

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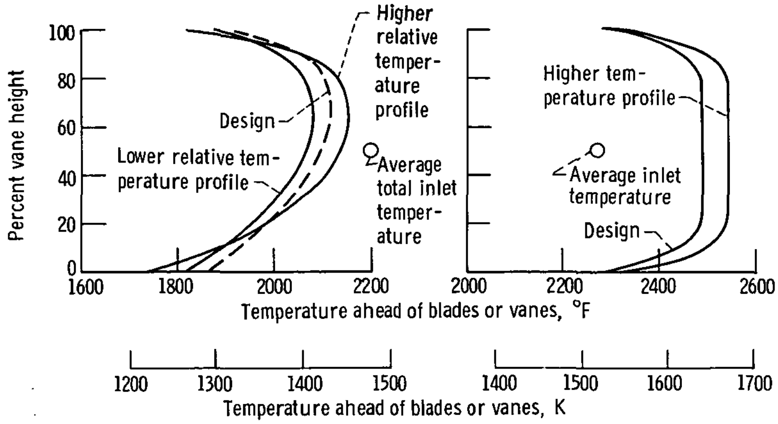
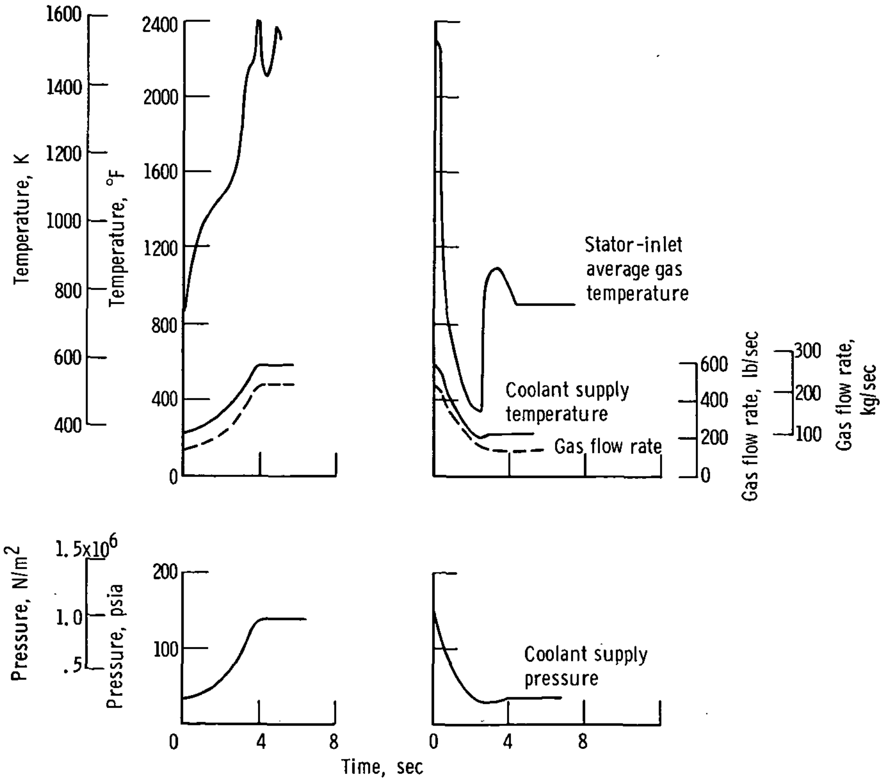


(a) Rotor blades. (b) Stator vanes.

Figure 2. - Gas temperature profiles.



(a) Fast acceleration and deceleration hot-gas and coolant transients.

Figure 3. - Transient conditions.

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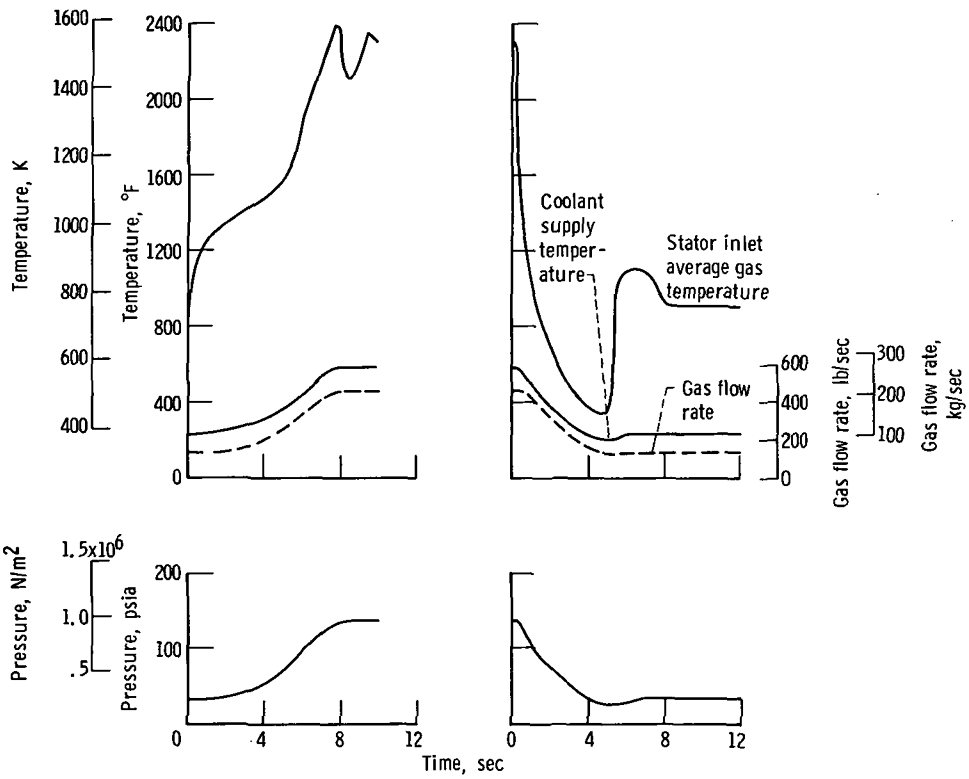

(b) Slow acceleration and deceleration hot-gas and coolant transients.

Figure 3. - Concluded.

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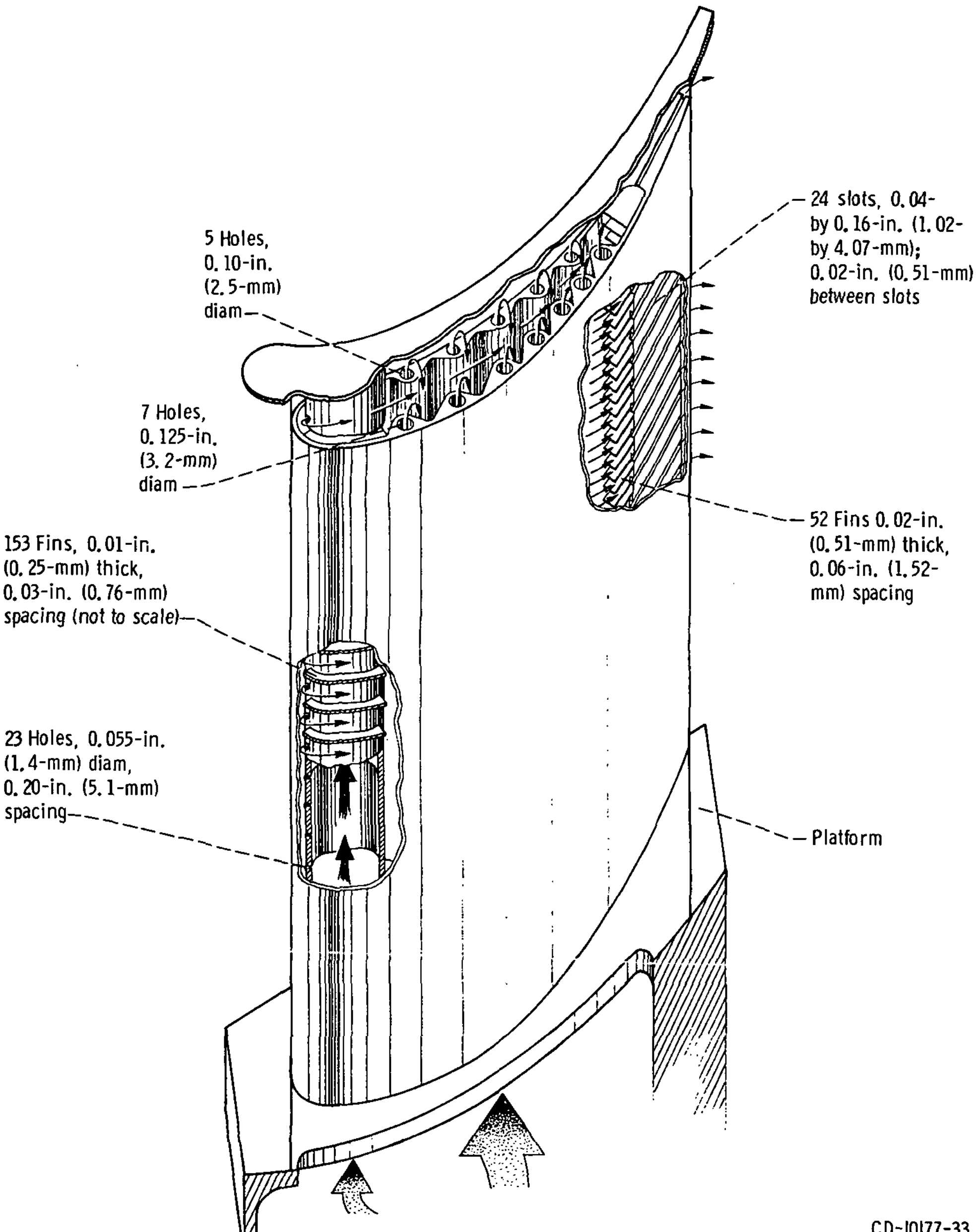
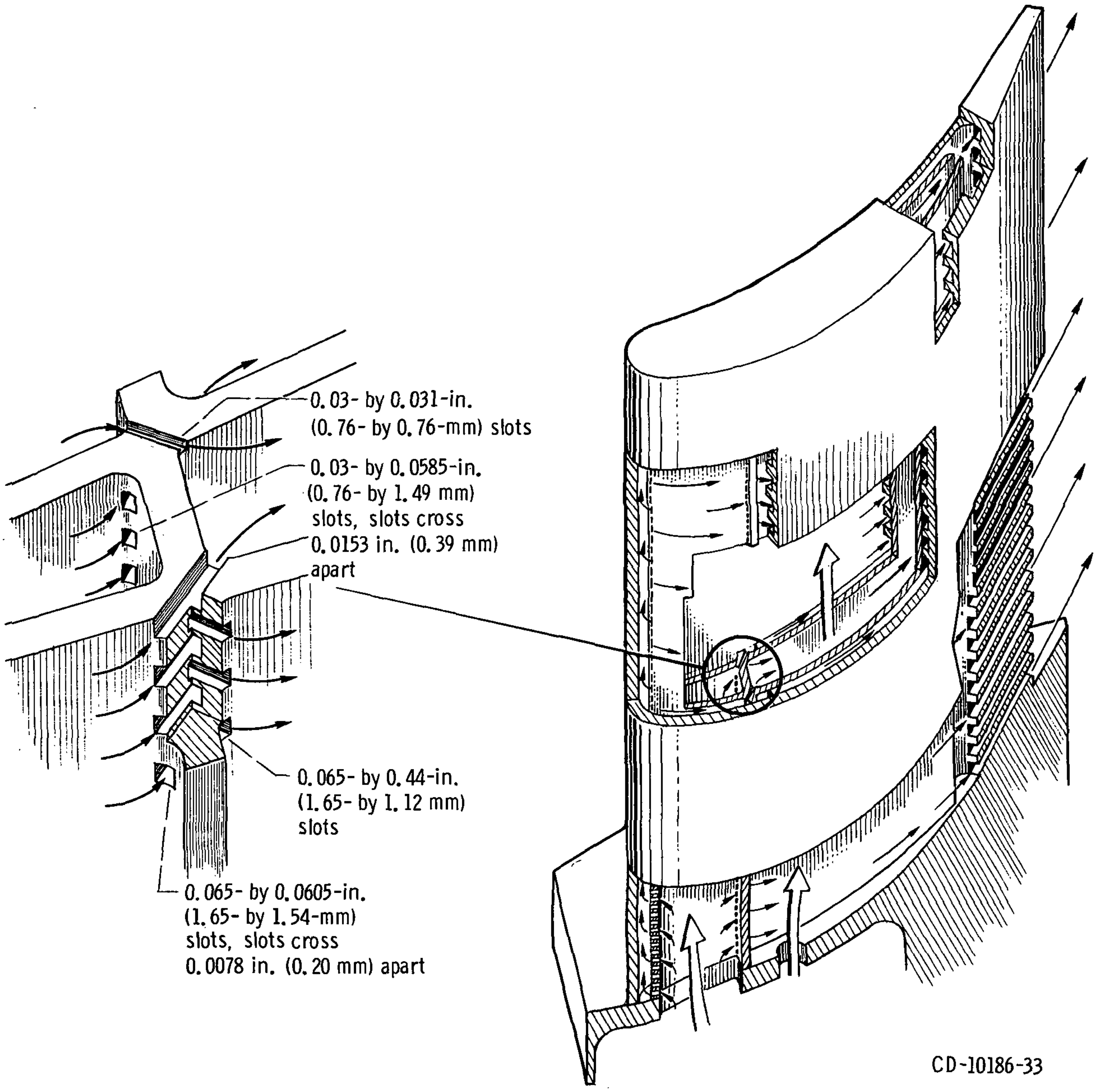


(a) Cast or forged convection-cooled vane design I.

Figure 4. - Vane designs.

[redacted]



(b) Strut insert; convection cooled vane design II.

Figure 4. - Continued.

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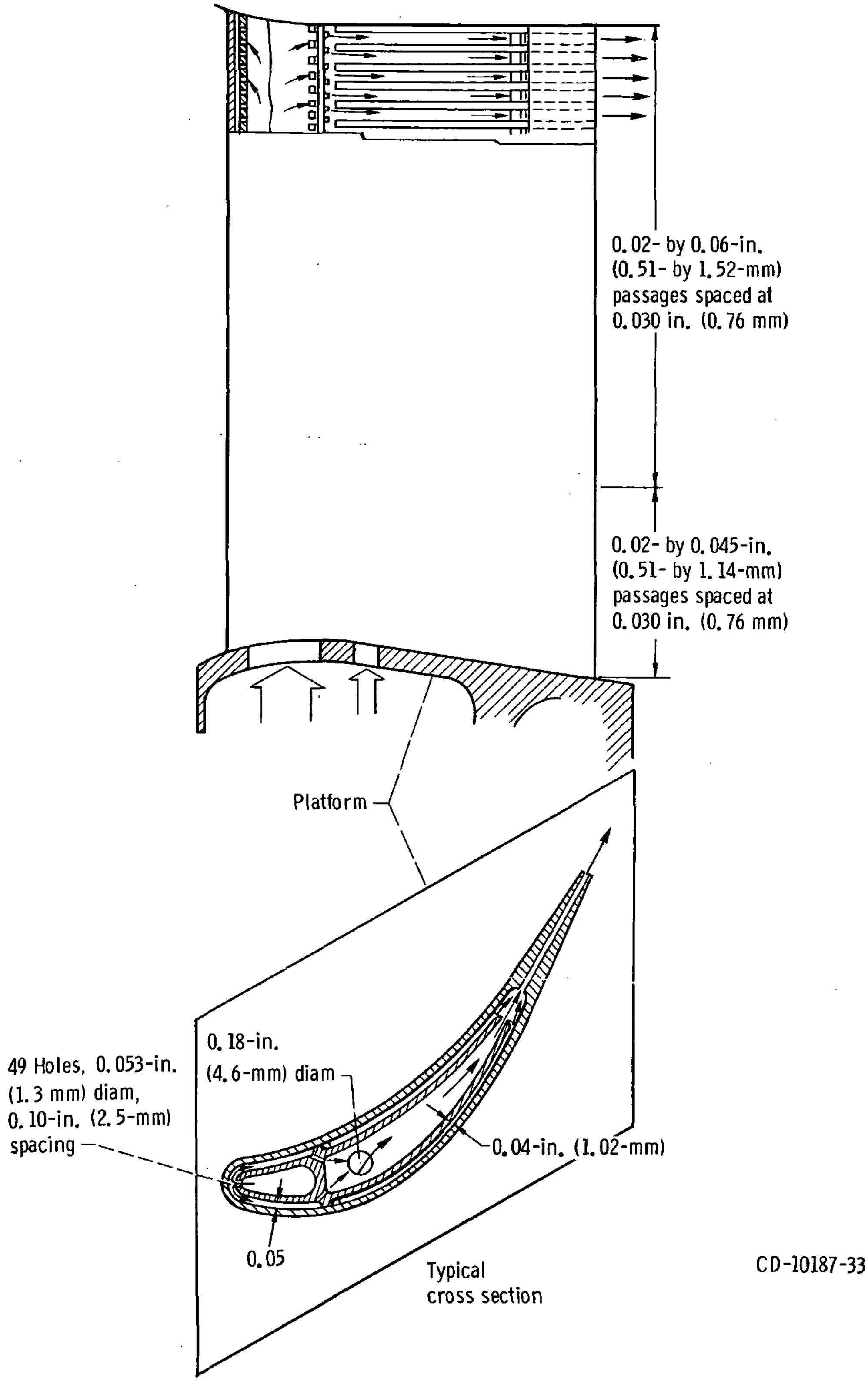

(b) Concluded.

Figure 4. - Continued.

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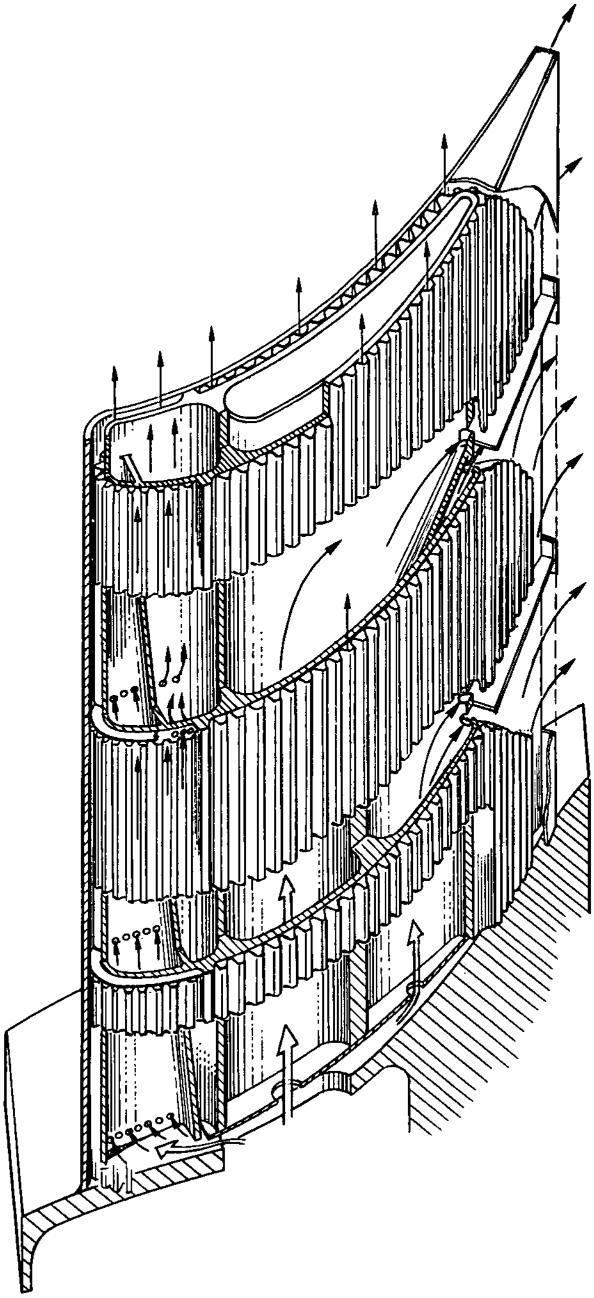

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(c) Corrugated sheet metal insert; convection cooled vane design III.

Figure 4. - Continued.

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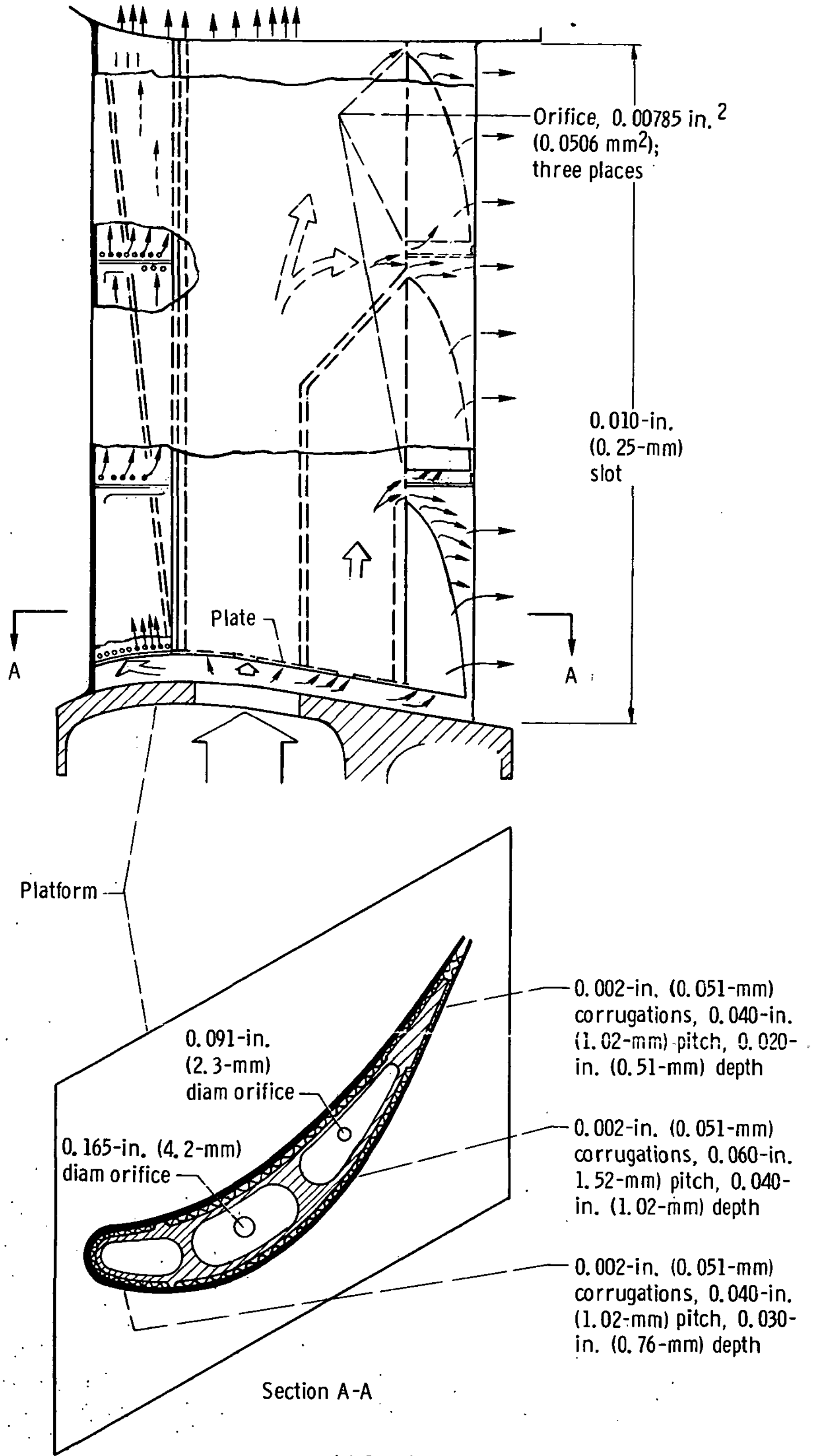

(c) Concluded.

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Figure 4. - Continued.

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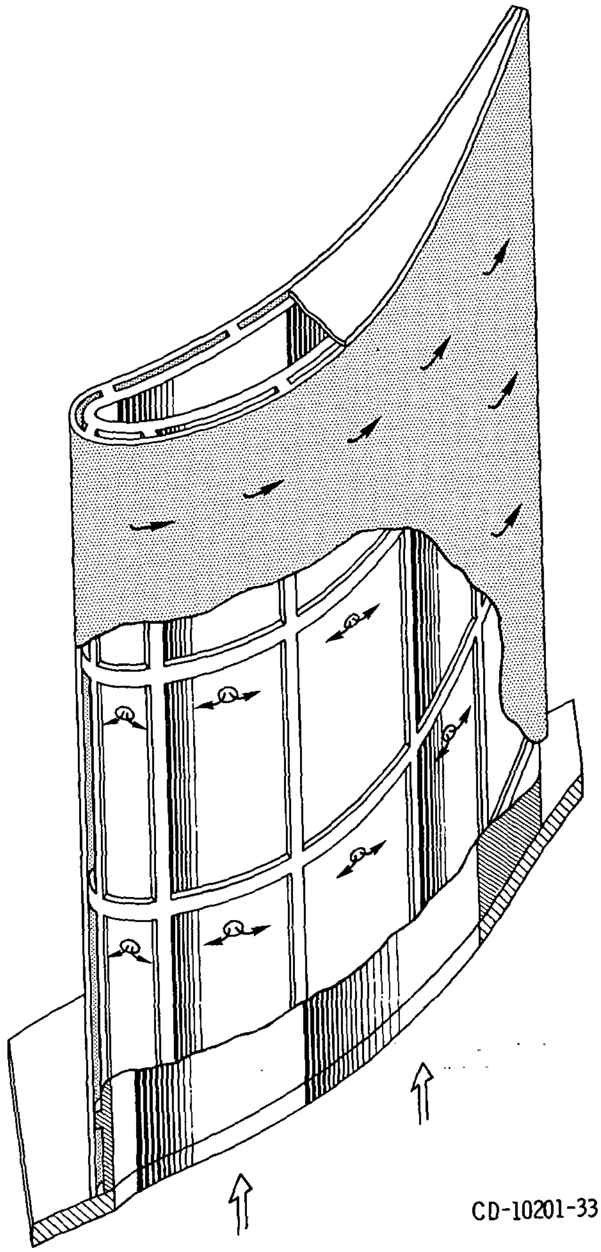

(d) Transpiration cooled vane design IV (not to scale).

Figure 4. - Continued.

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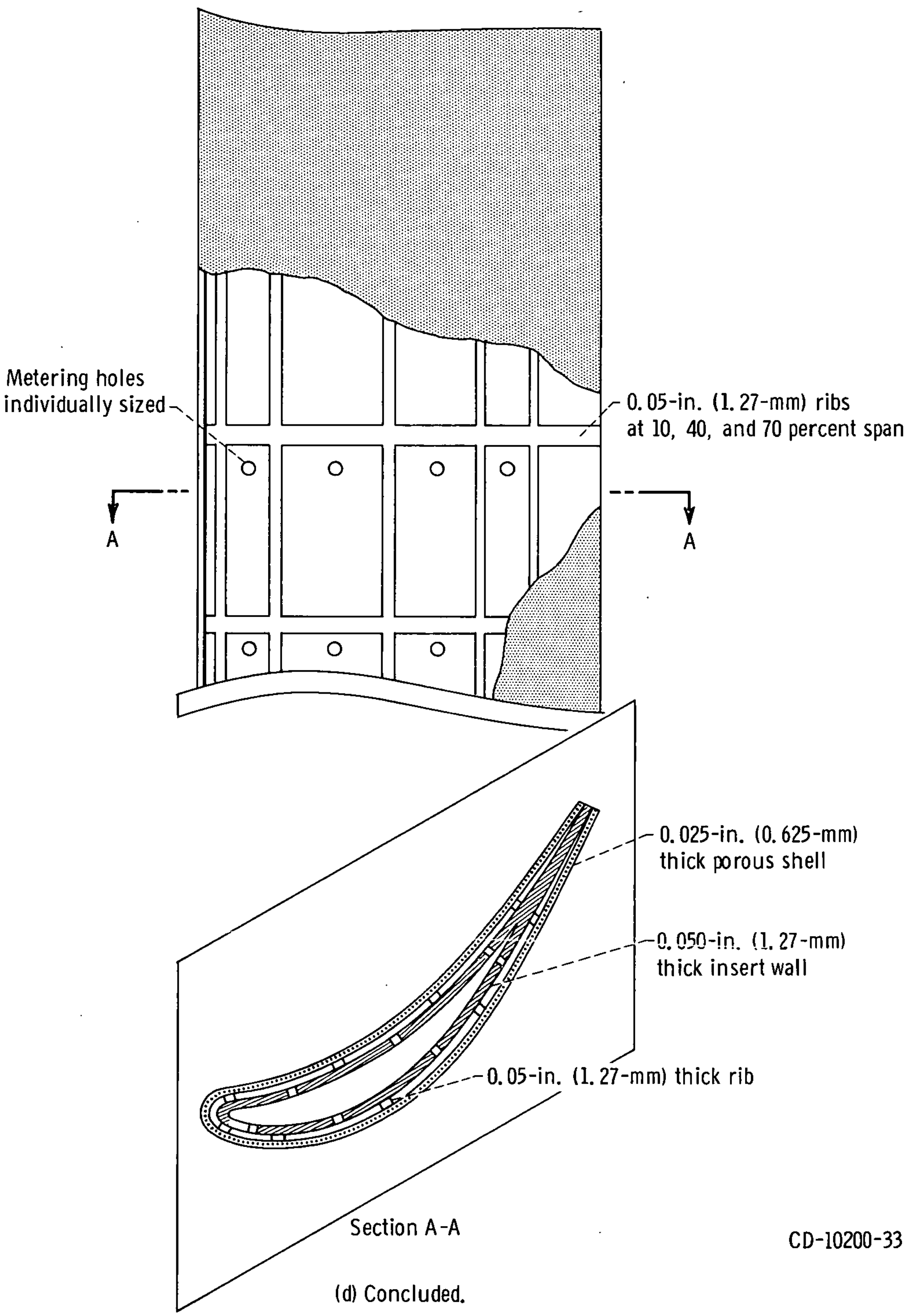

(d) Concluded.

Figure 4. - Continued.

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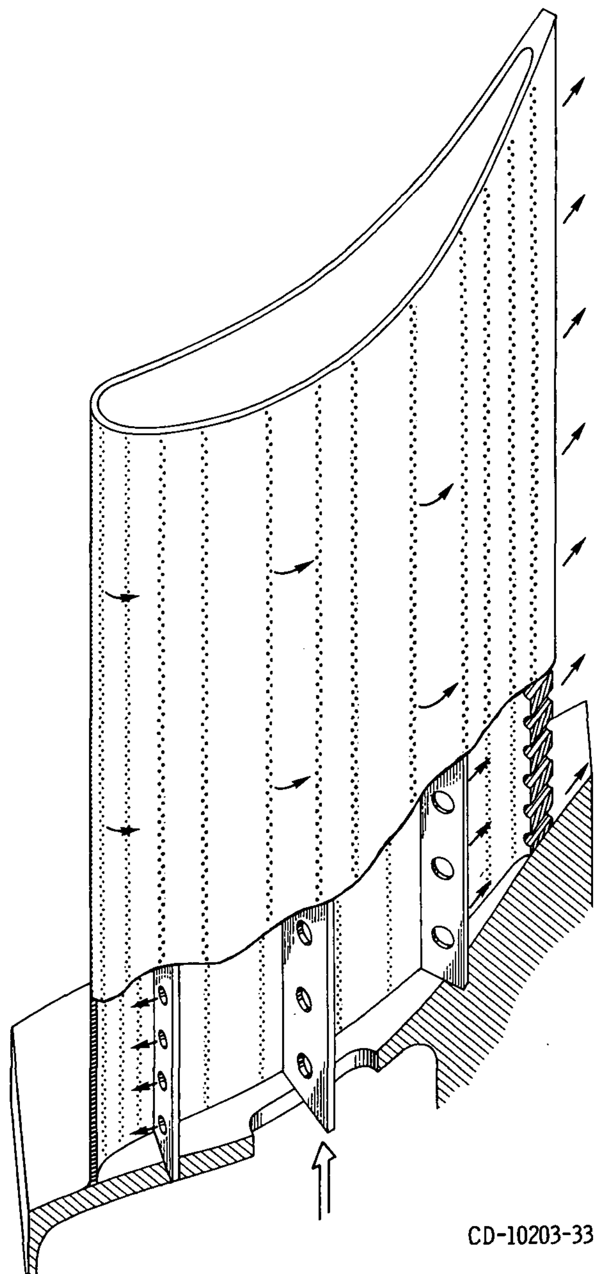

(e) Film cooled vane design V.

Figure 4. - Continued.

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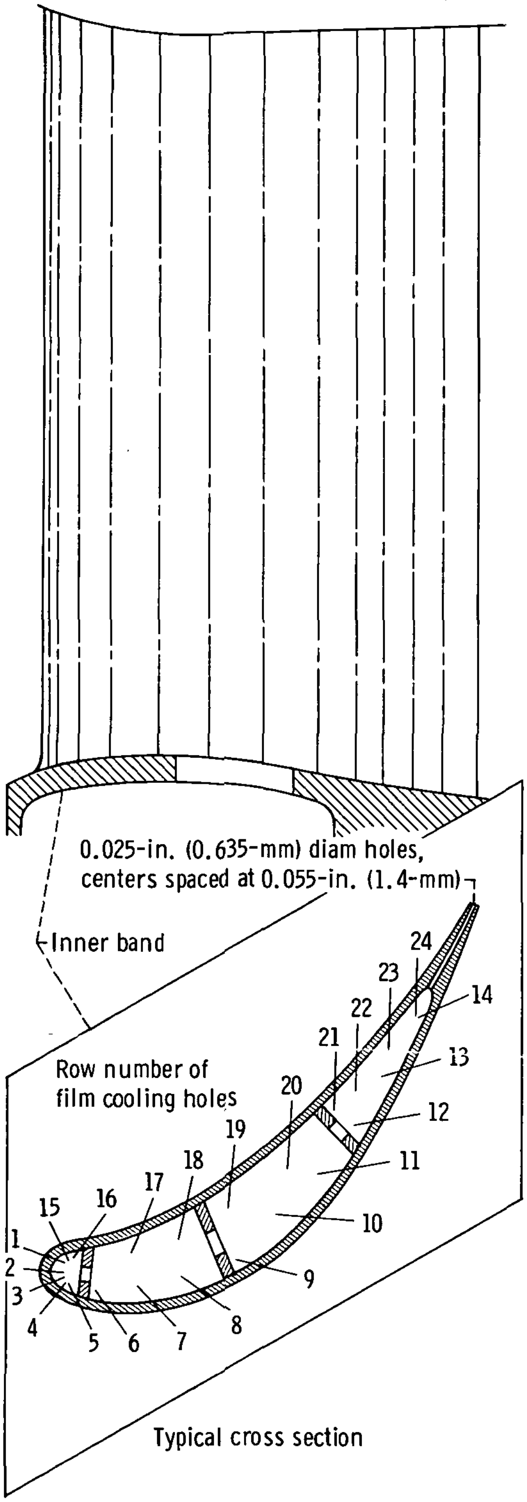
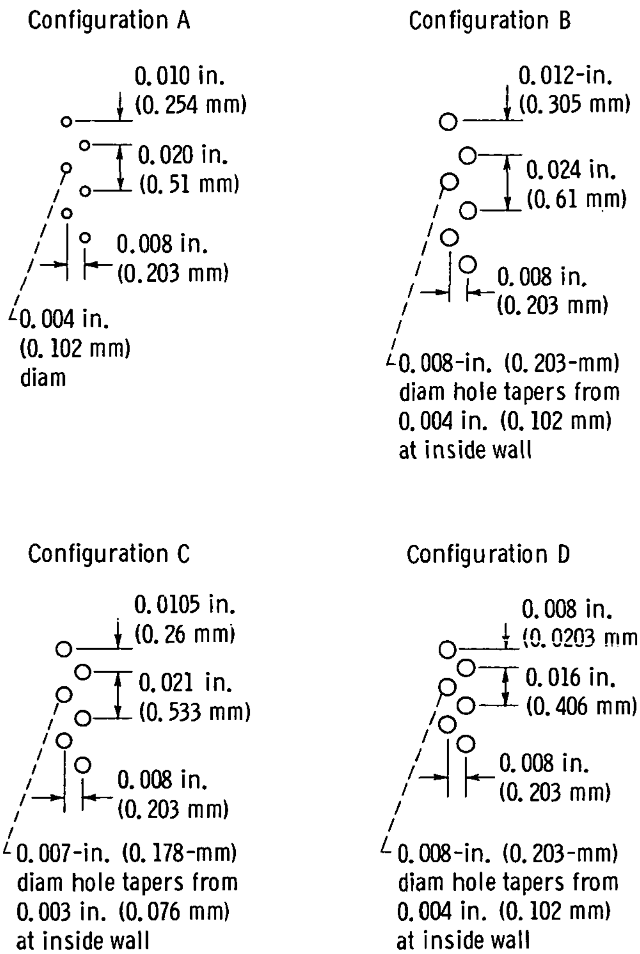


Row	Hole configuration
1-4 and 15	A
5, 8-12, 16, and 21-23	B
6, 7, and 17-20	C
13, 14, and 24	D



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(e) Concluded.

Figure 4. - Continued.

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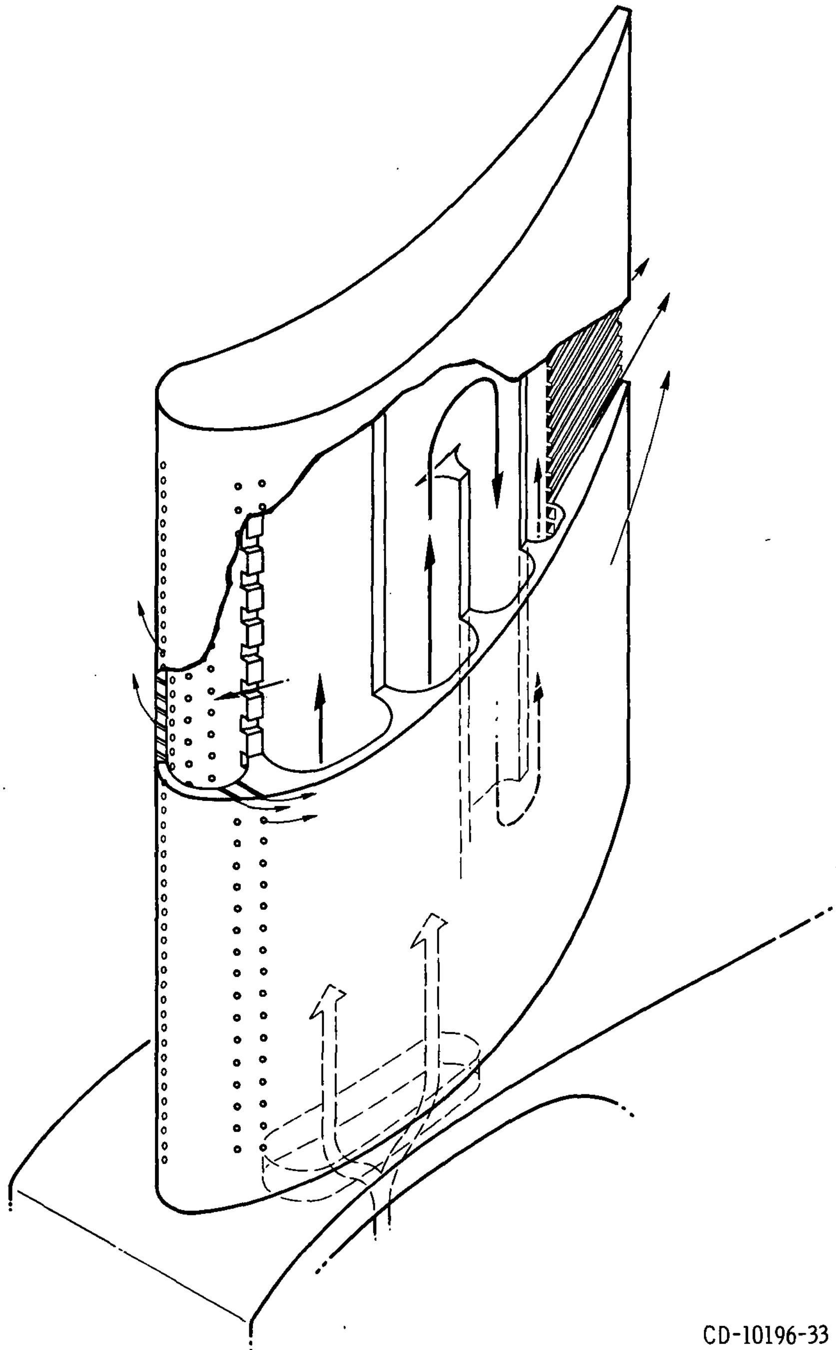

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(f) Film and convection cooled vane design VI.

Figure 4. - Continued.

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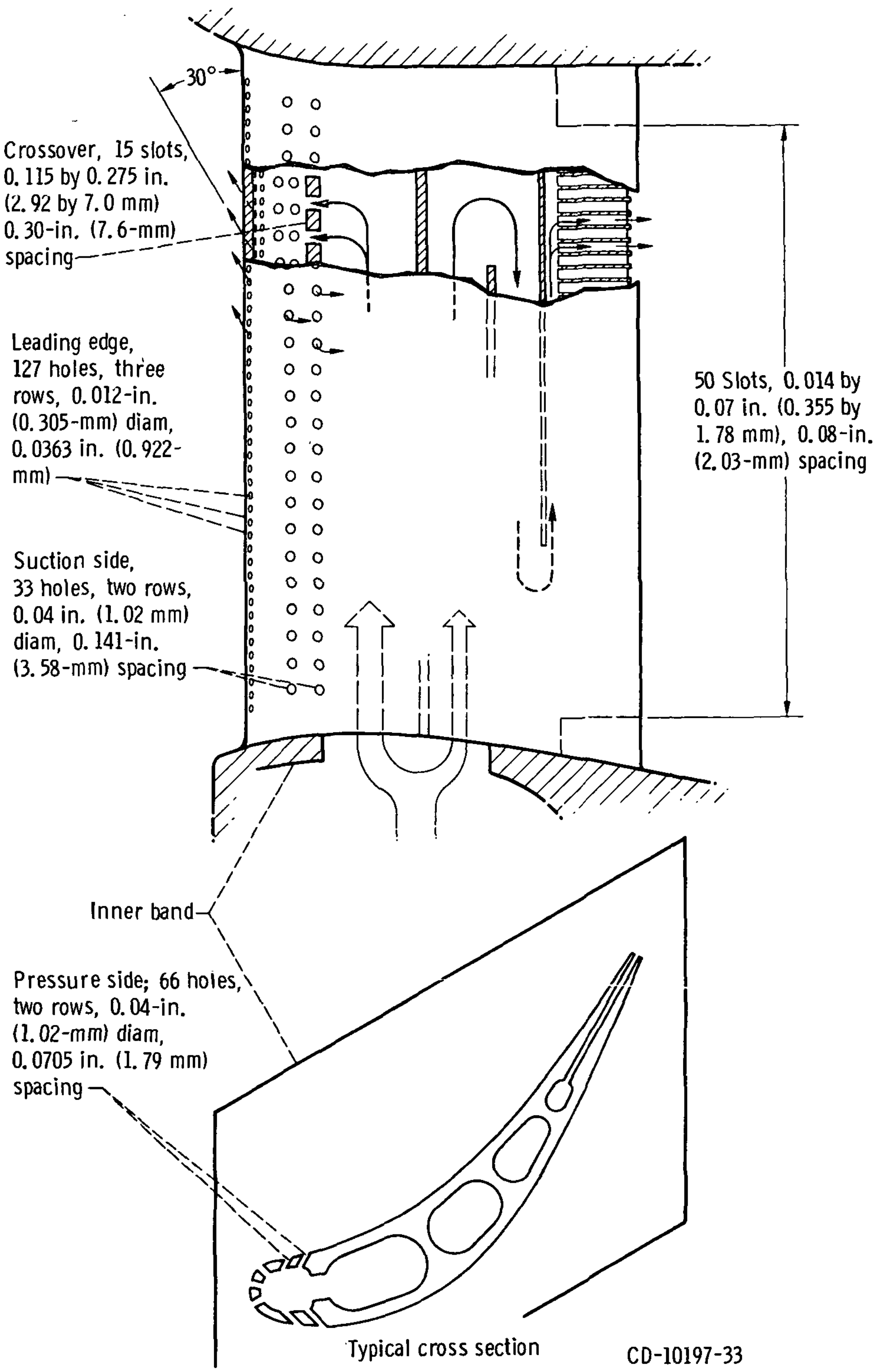

(f) Concluded.

Figure 4. - Continued.

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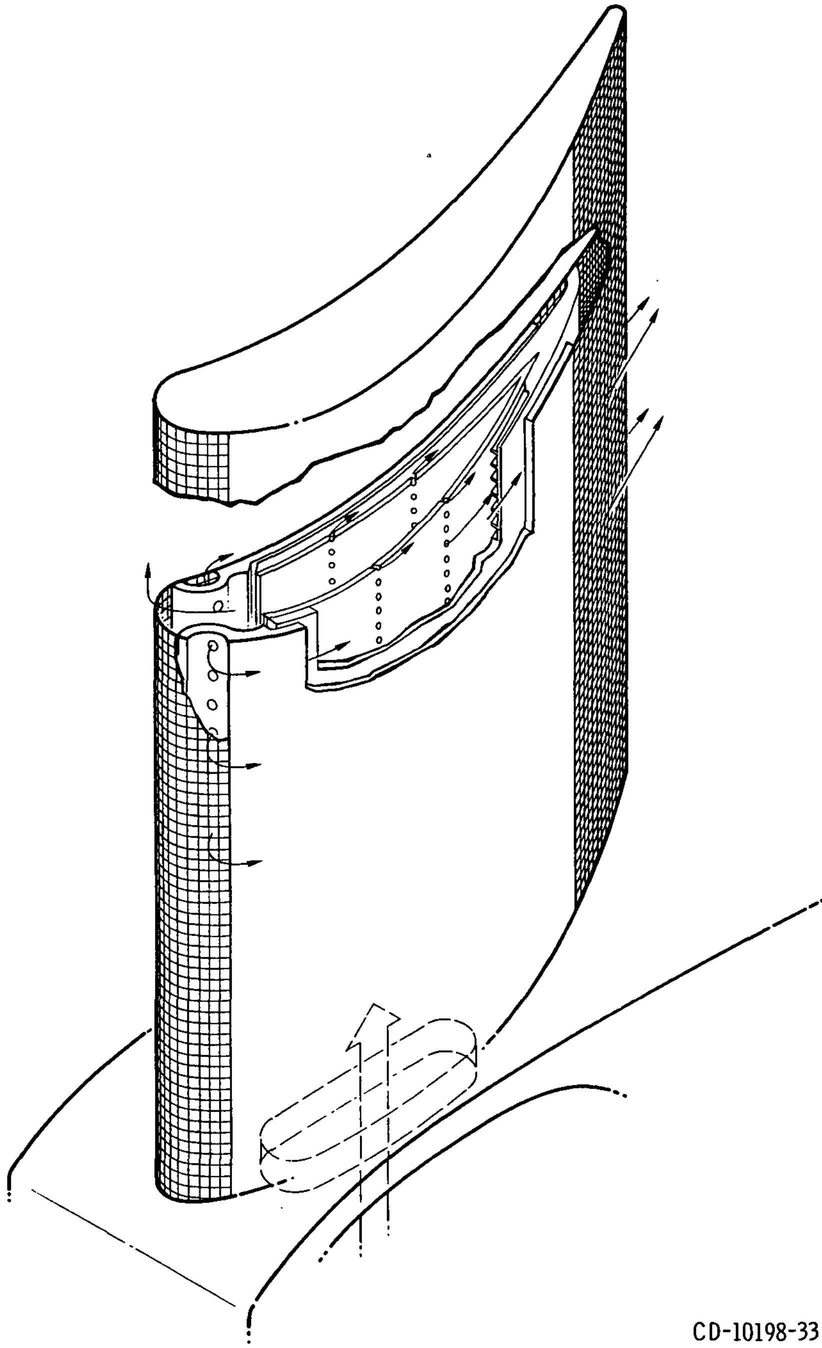

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(g) Transpiration and convection cooled vane design VII.

Figure 4. - Continued.

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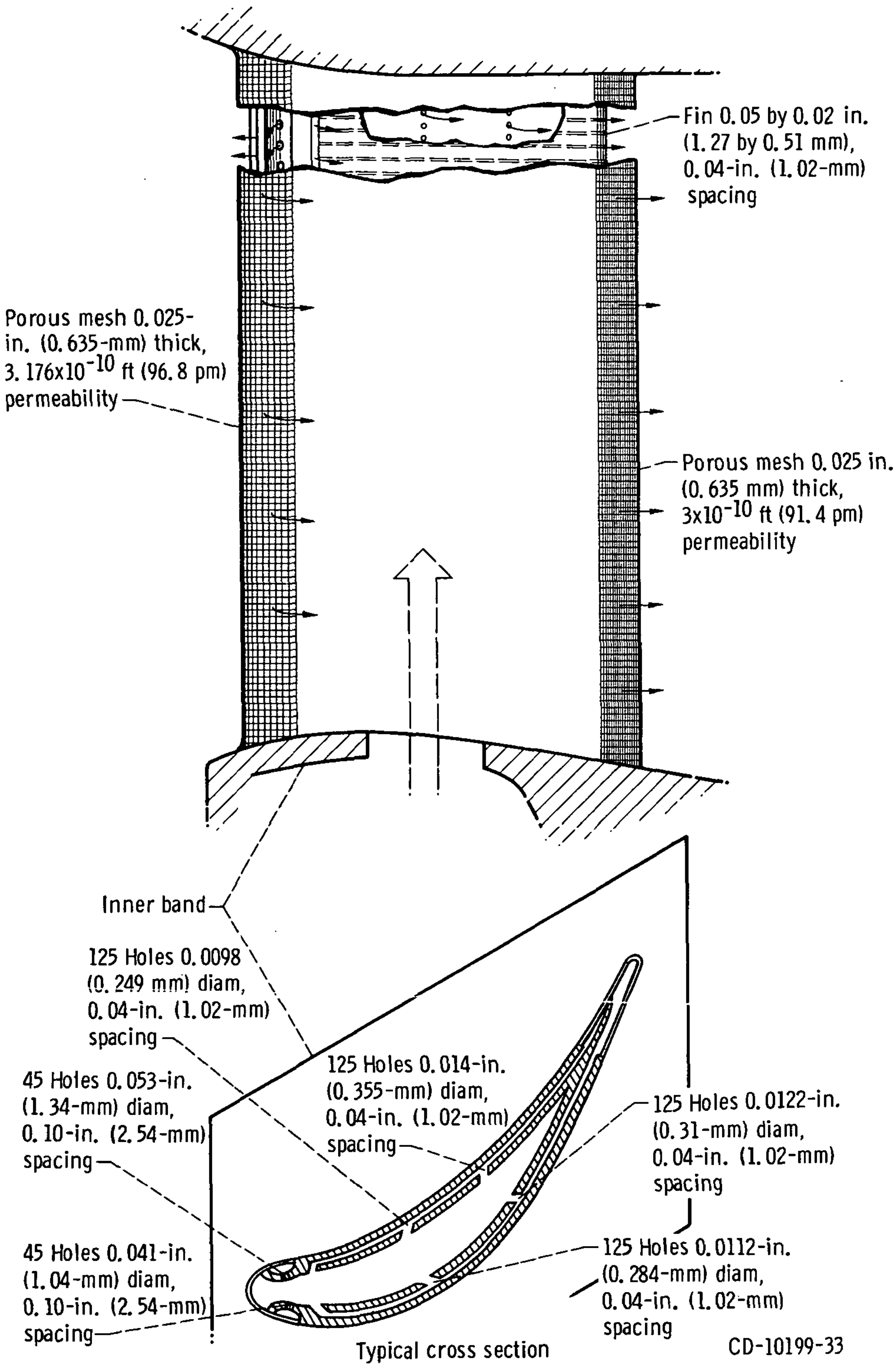

(g) Concluded.

Figure 4. - Continued.

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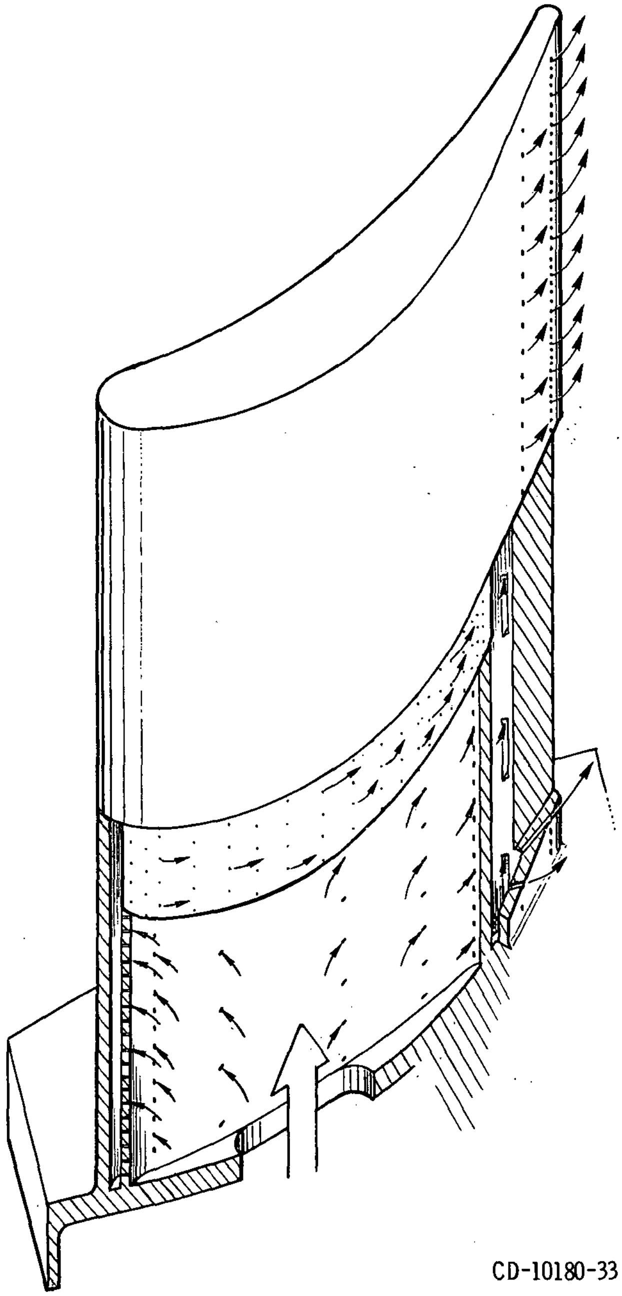

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(h) Impingement and convection cooled vane design VIII.

Figure 4. - Continued.

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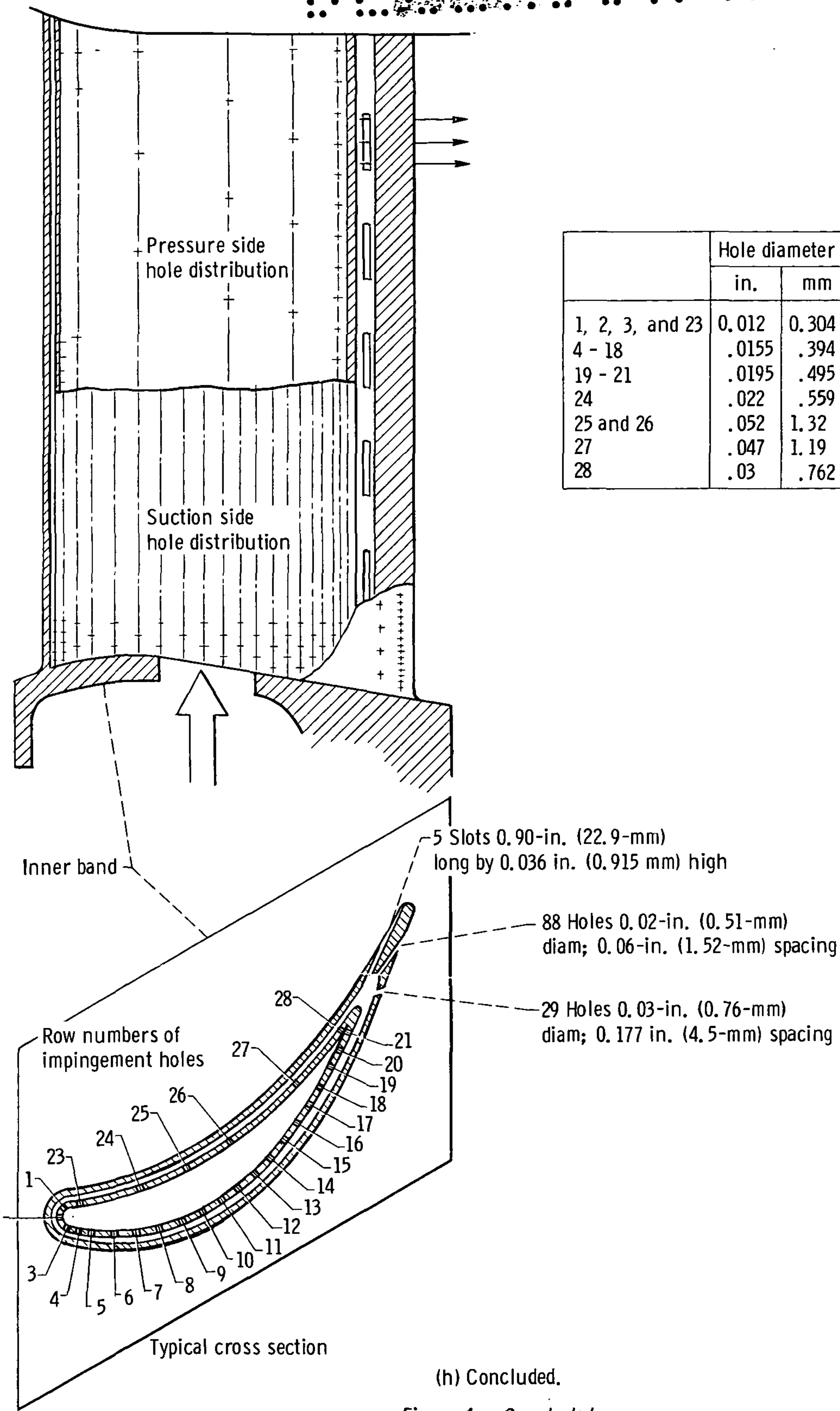

	Hole diameter		Hole spacing	
	in.	mm	in.	mm
1, 2, 3, and 23	0.012	0.304	0.084	2.137
4 - 18	.0155	.394	.175	4.45
19 - 21	.0195	.495	.22	5.59
24	.022	.559	.27	6.85
25 and 26	.052	1.32	.80	20.3
27	.047	1.19	.70	17.8
28	.03	.762	.25	6.35

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(h) Concluded.

Figure 4. - Concluded.

[redacted]

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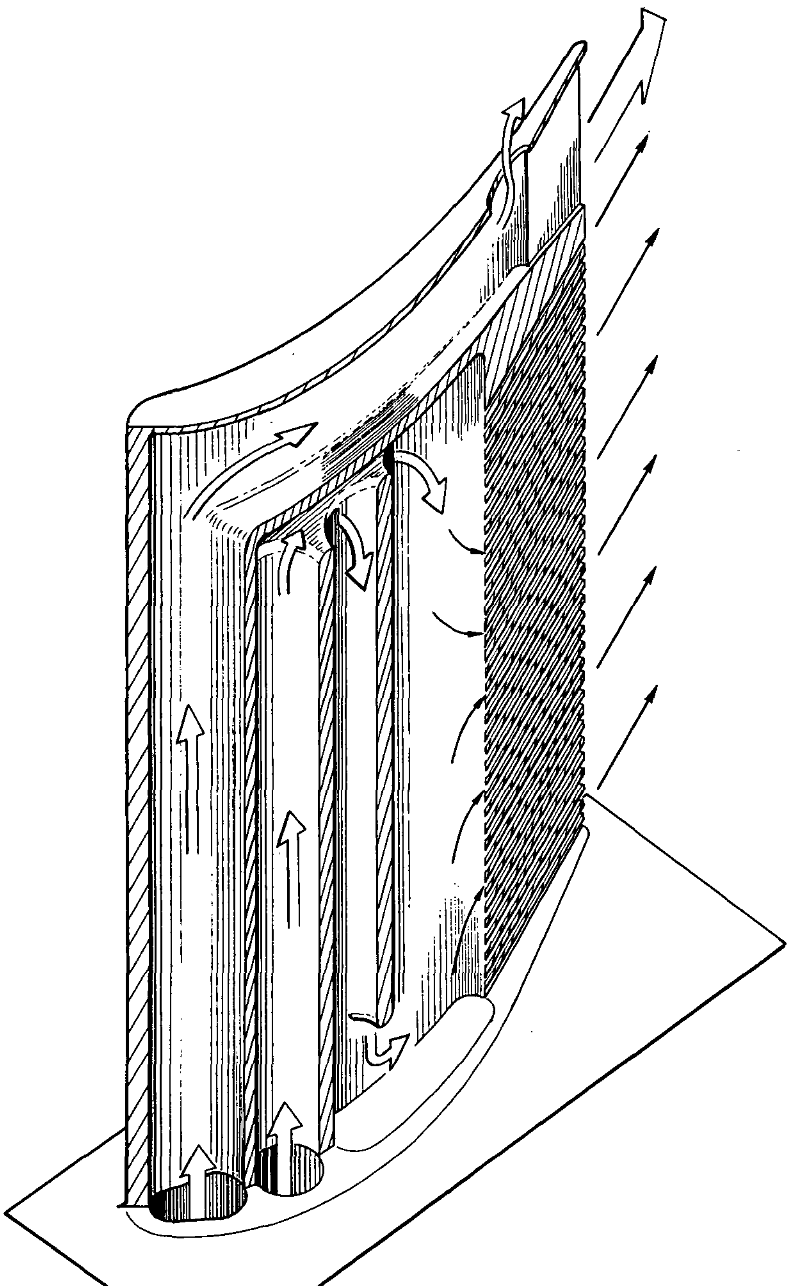

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(a) Cast or forged convection cooled blade design I.

Figure 5. - Blade designs.

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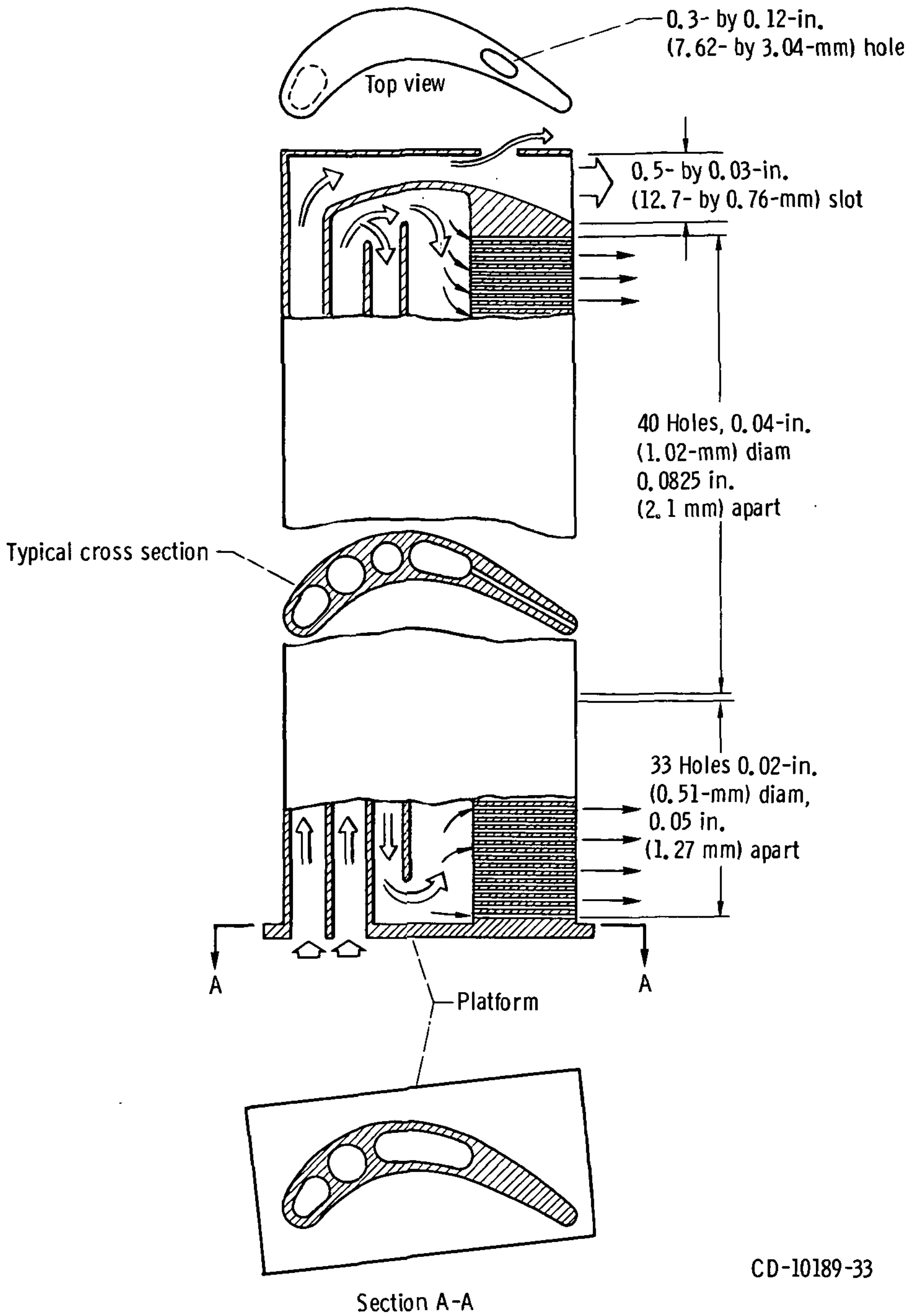
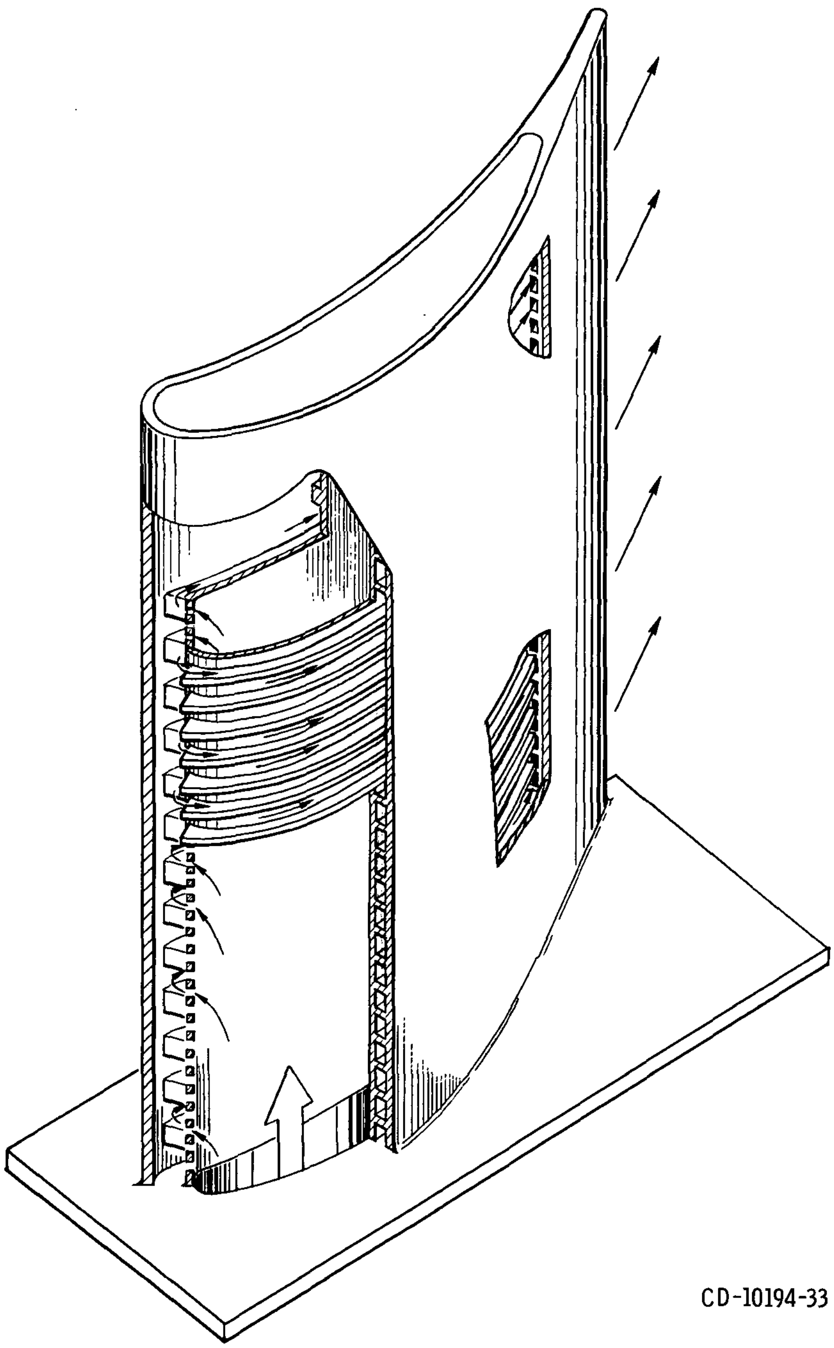


(a) Concluded.

Figure 5. - Continued.

[redacted]

[redacted]



(b) Strut insert, convection cooled blade design II.

Figure 5. - Continued.

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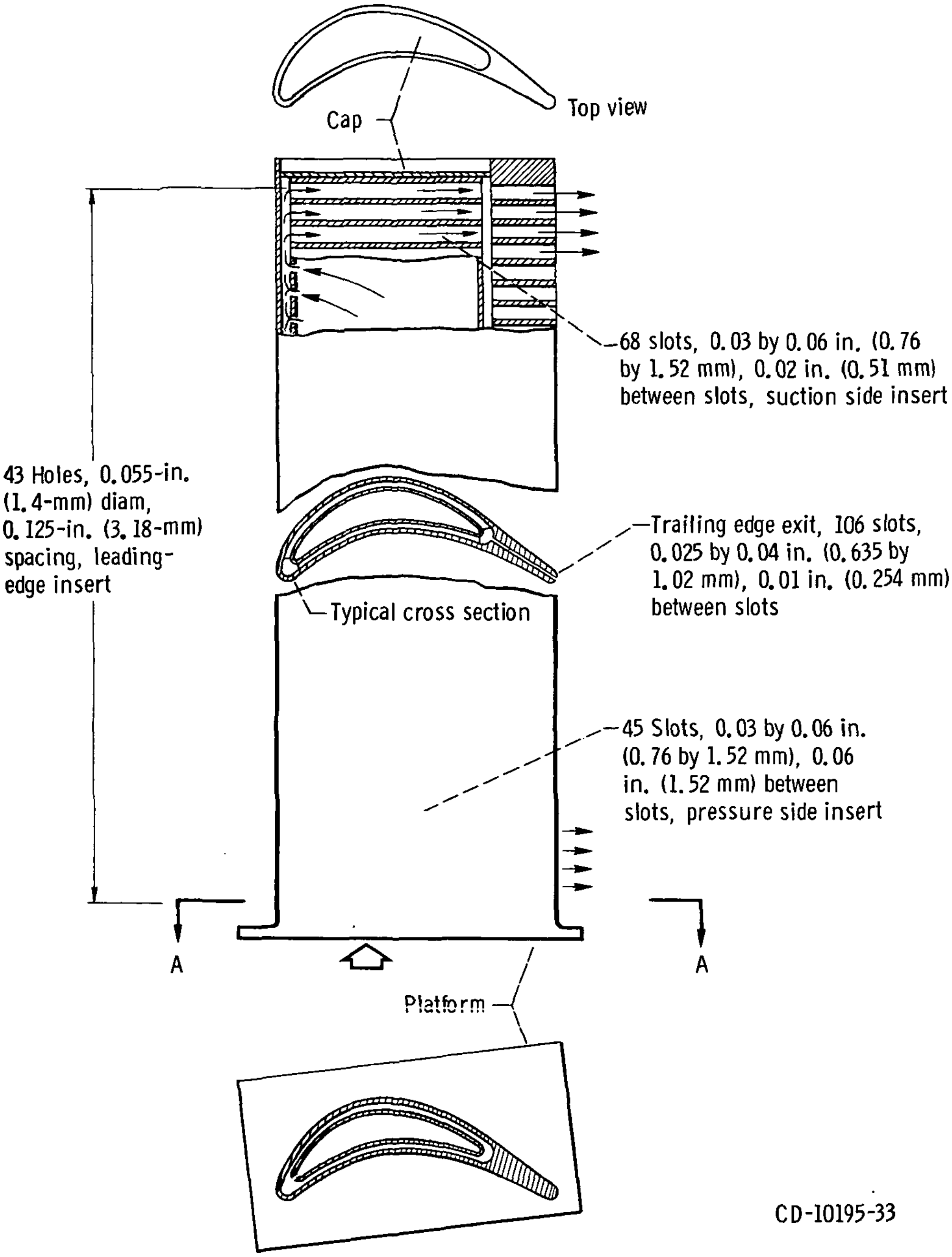

(b) Concluded.
Figure 5. - Continued.

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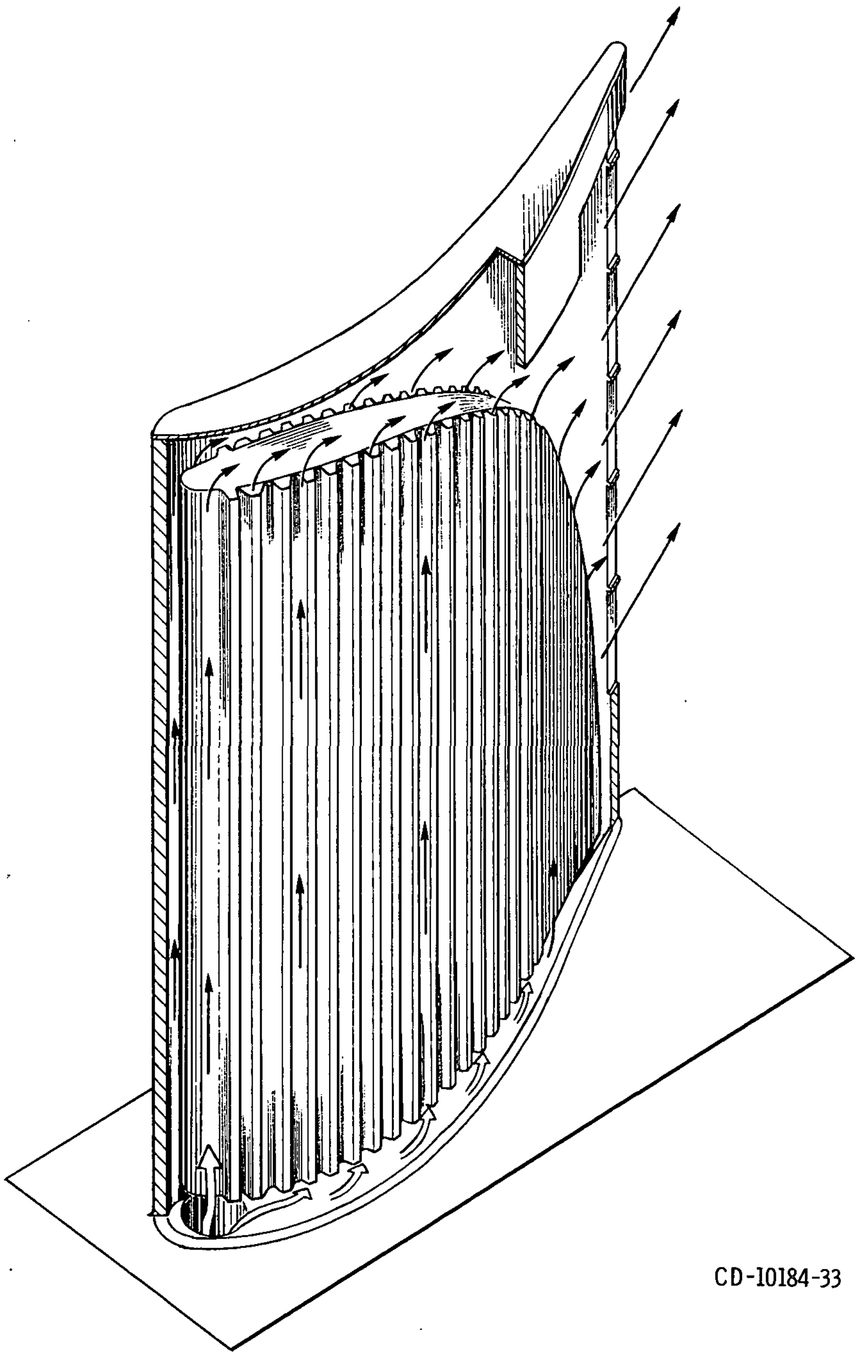

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(c) Corrugated sheet metal insert; convection cooled blade design III.
Figure 5. - Continued.

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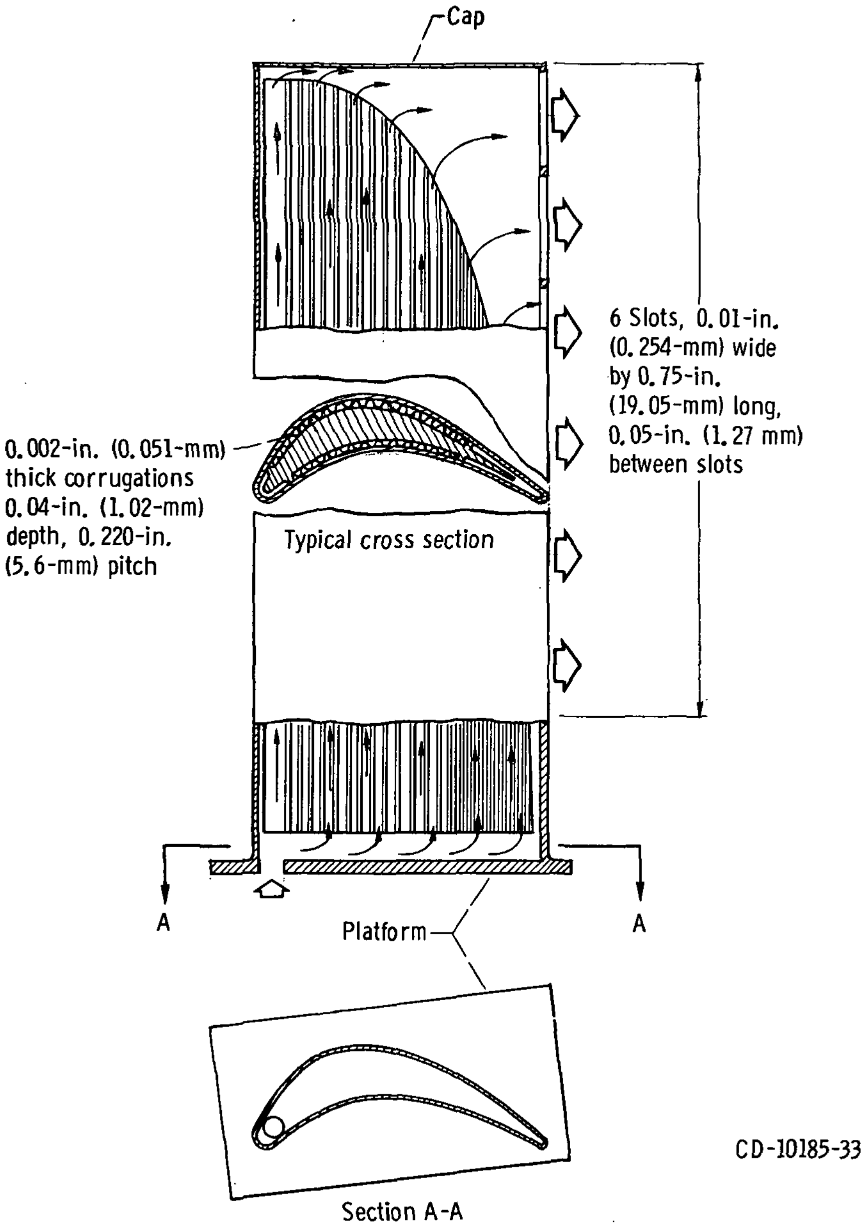

(c) Concluded.
Figure 5. - Continued.

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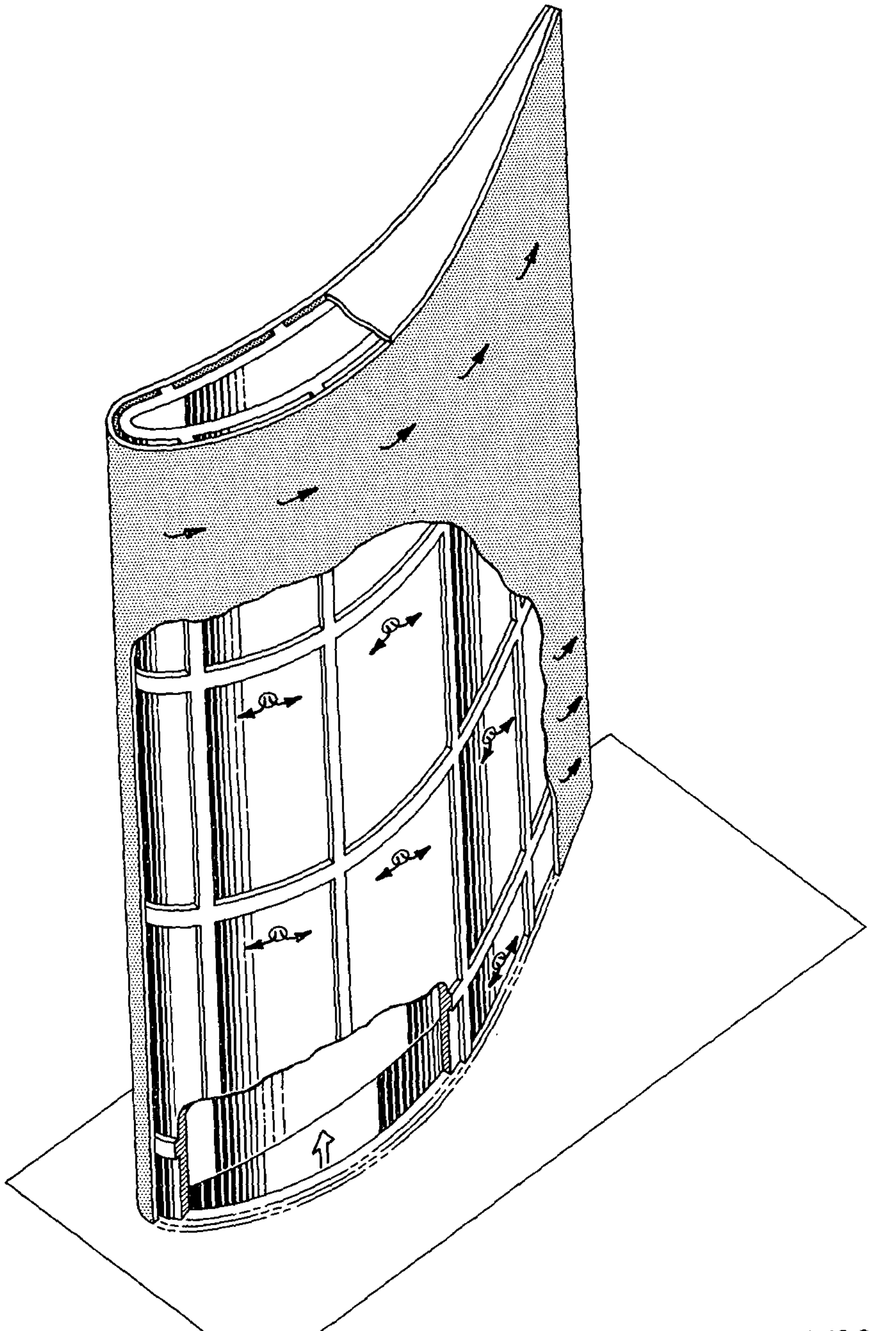

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(d) Transpiration cooled blade design IV (not to scale).

Figure 5. - Continued.

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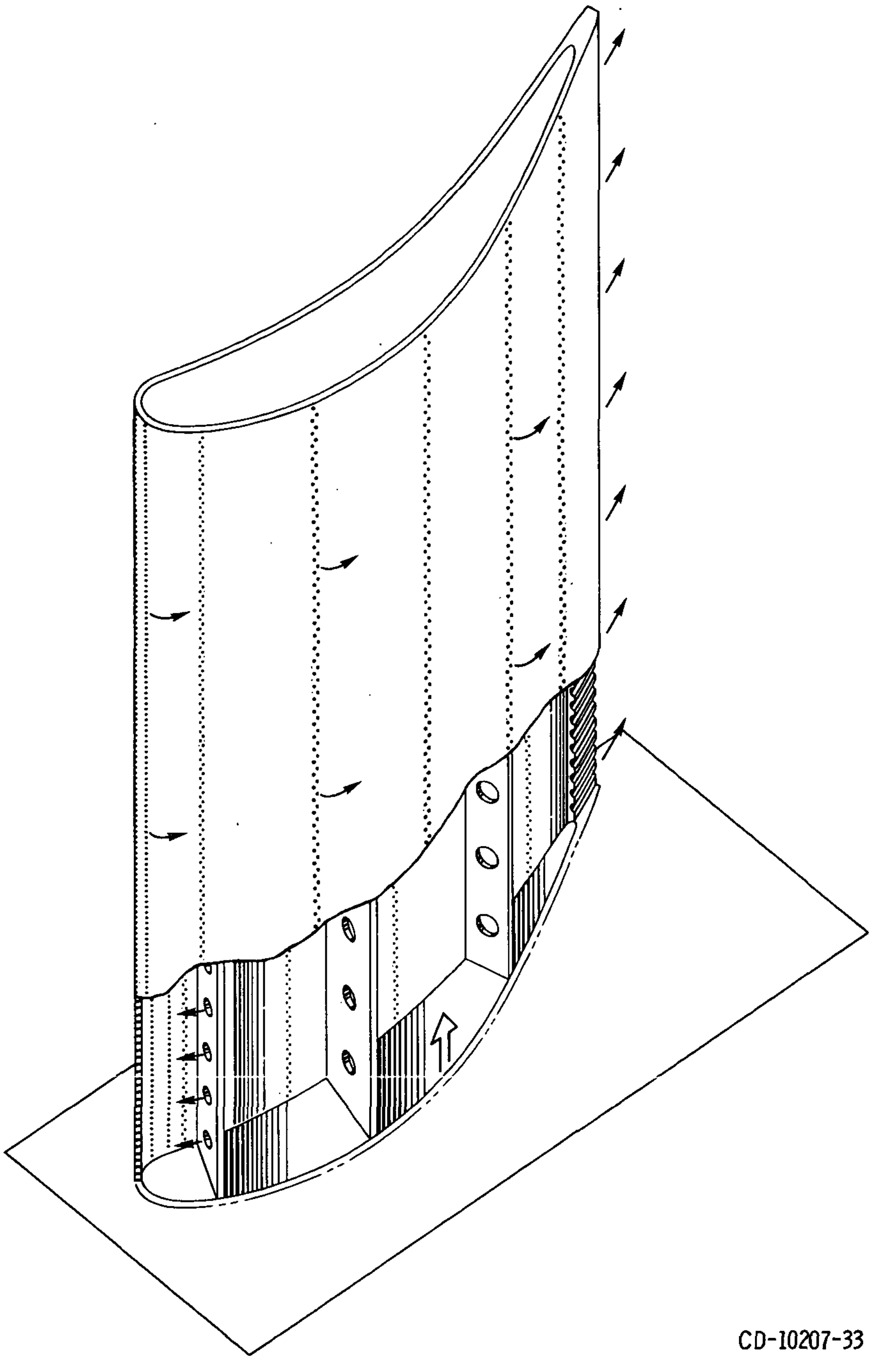

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(e) Film cooled blade design V.

Figure 5. - Continued.

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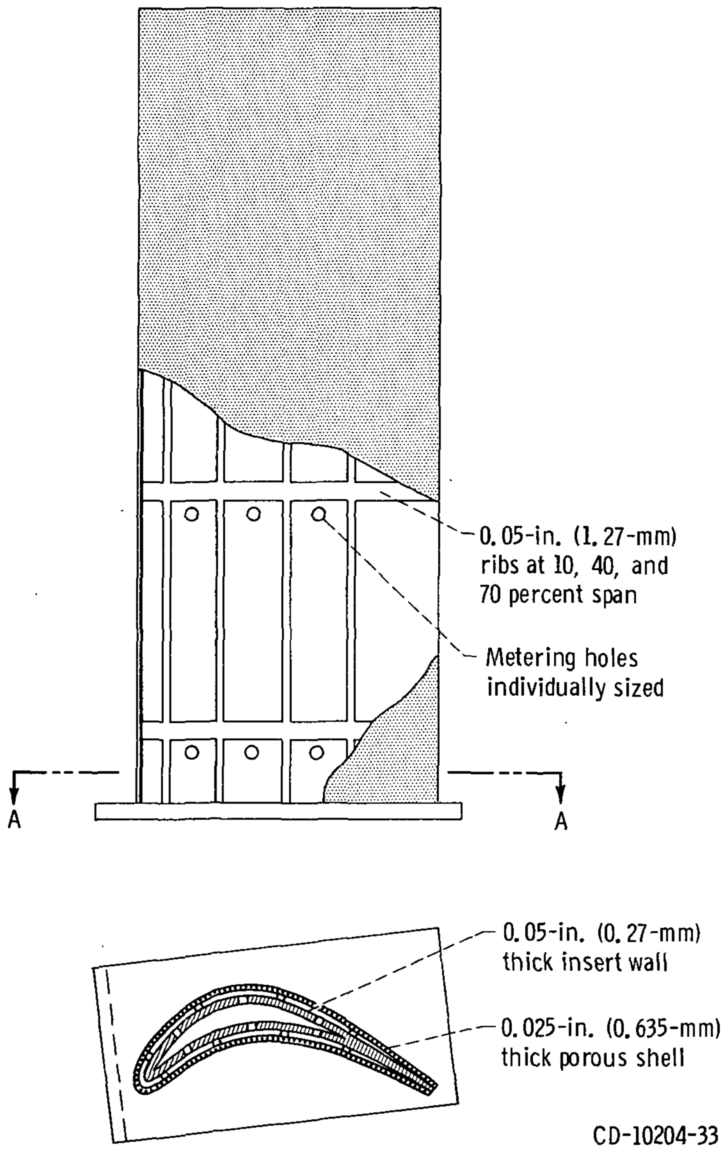

(d) Concluded.

Figure 5. - Continued.

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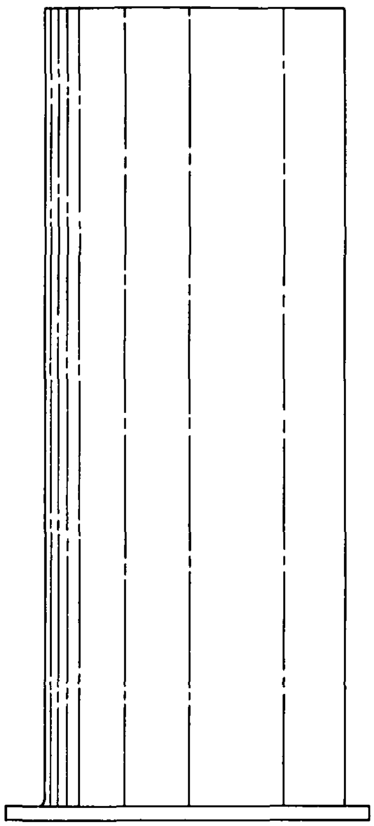
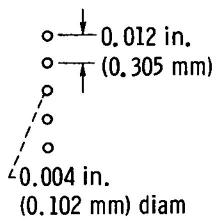
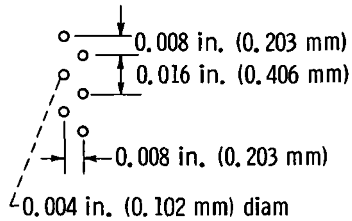
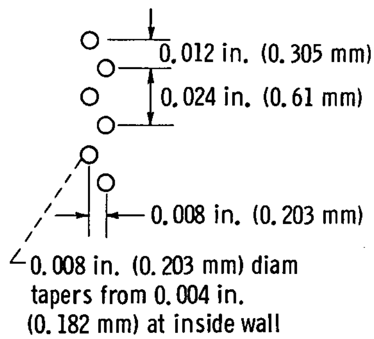
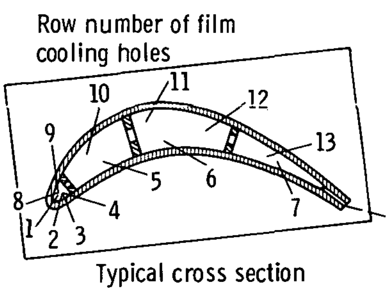


Row	Hole configuration
1 - 3 and 8	A
4 and 9	B
5 - 7 and 10 - 13	C









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(e) Concluded.

Figure 5. - Continued.

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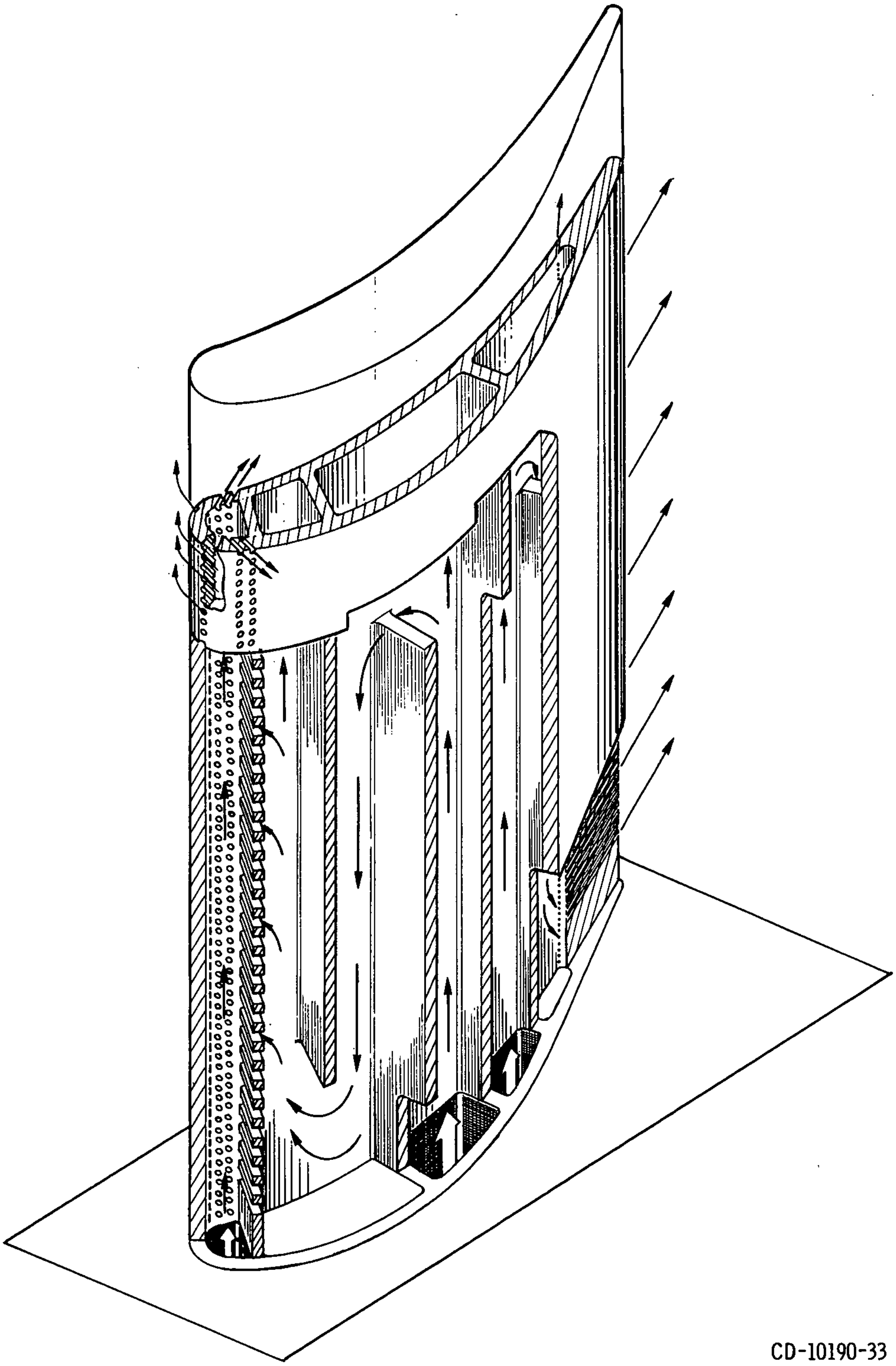

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(f) Film and convection cooled blade design VI.

Figure 5. - Continued.

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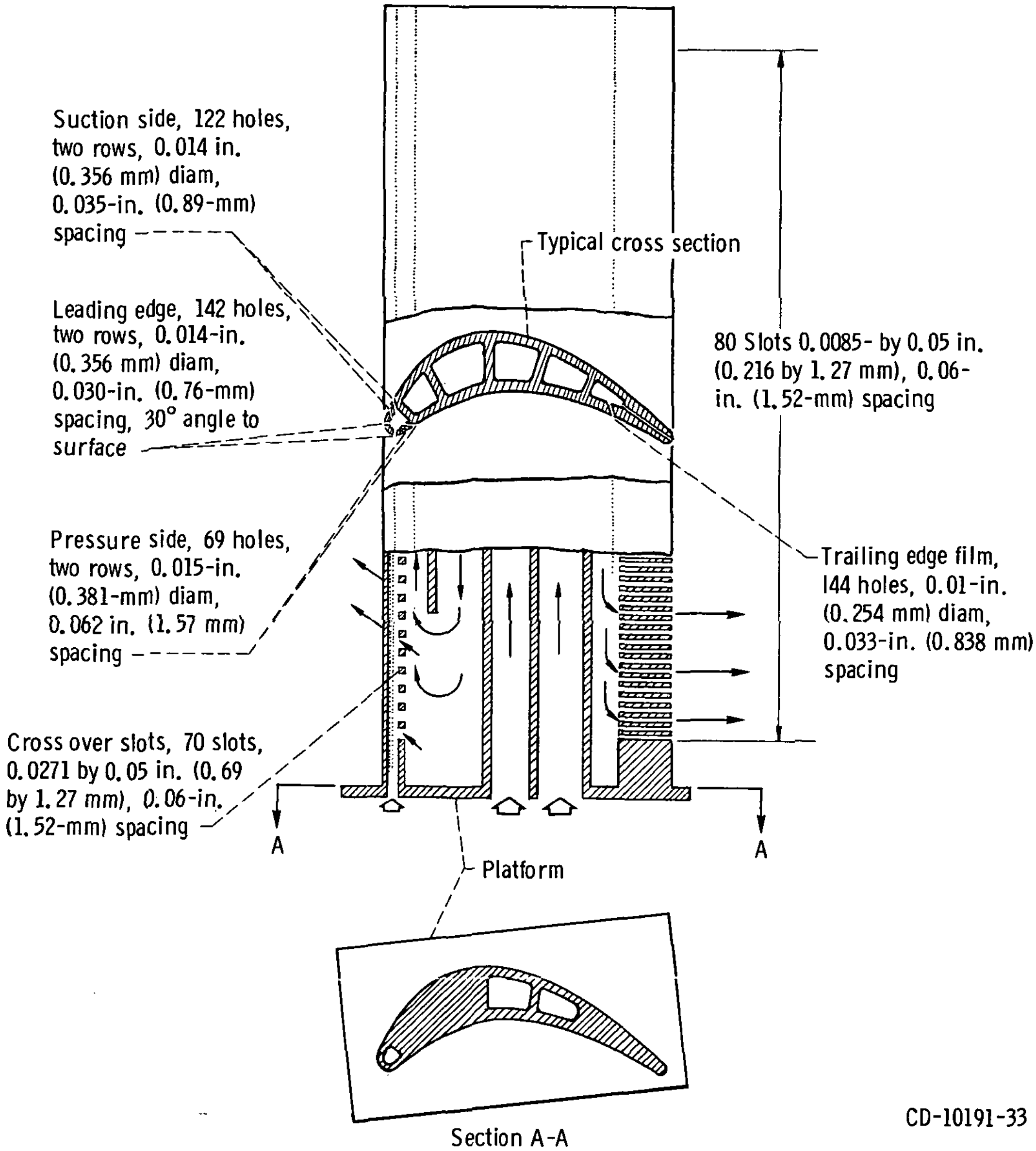

(f) Concluded.

Figure 5. - Continued.

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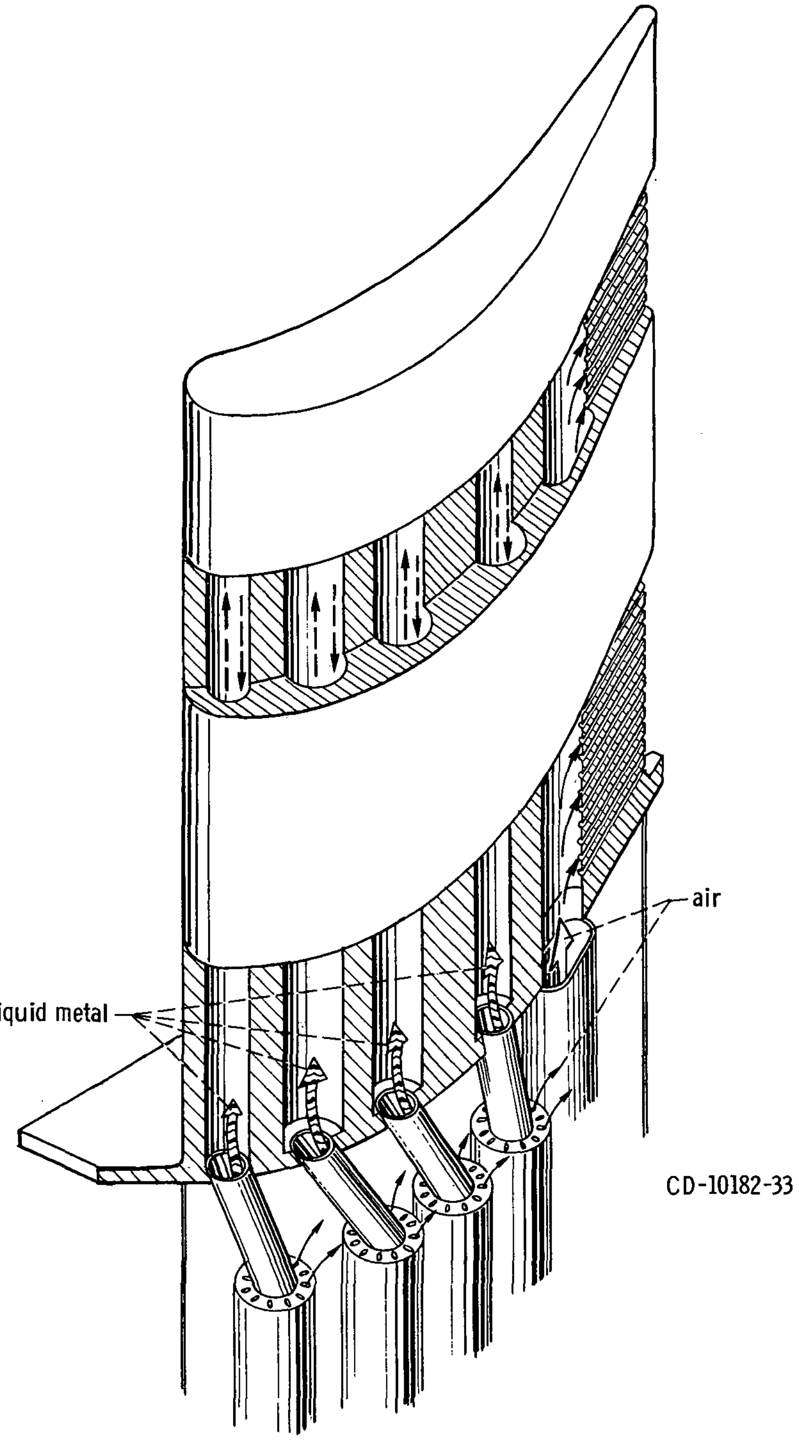

(g) Liquid metal and convection cooled blade design VII.

Figure 5. - Continued.

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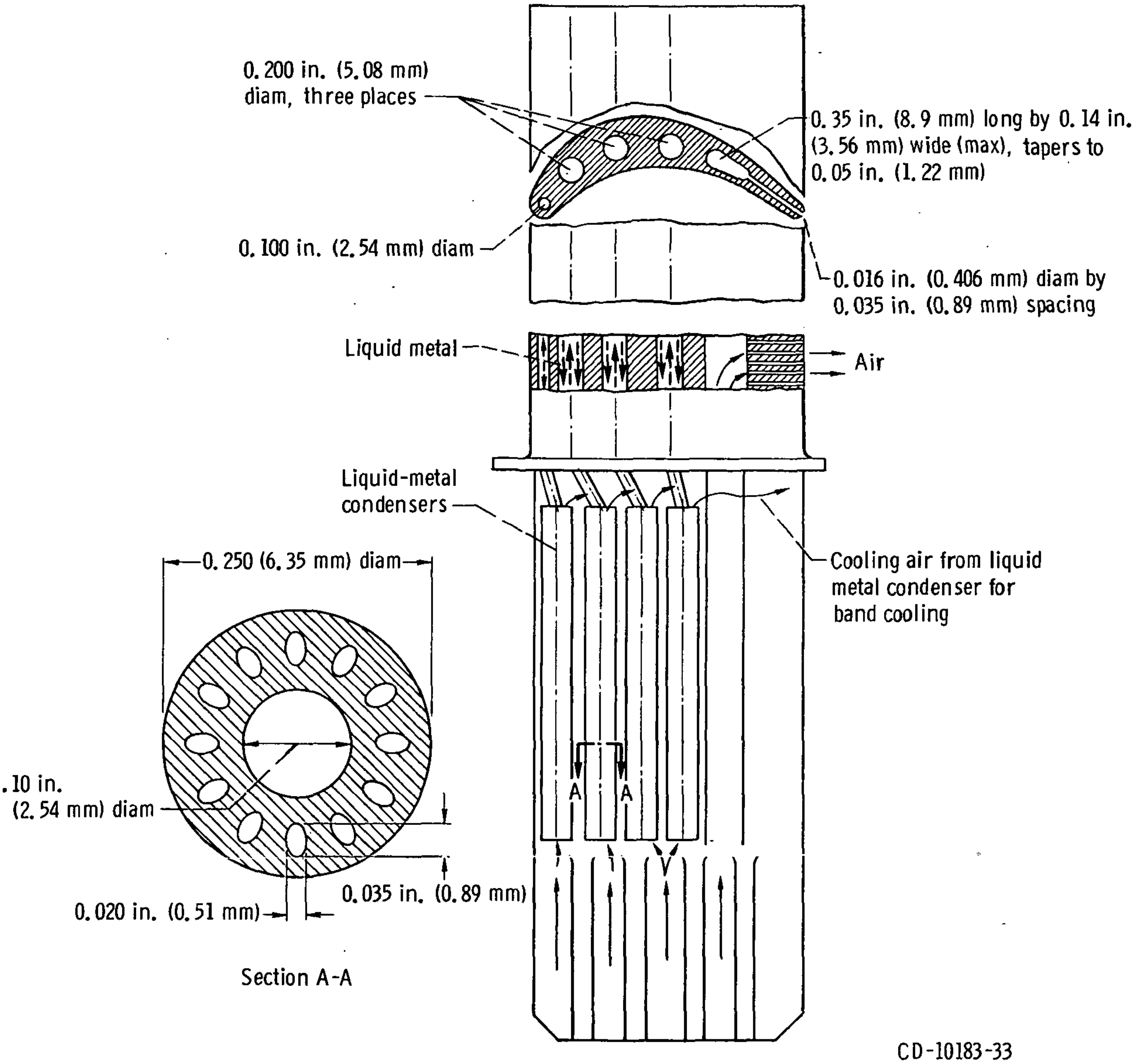
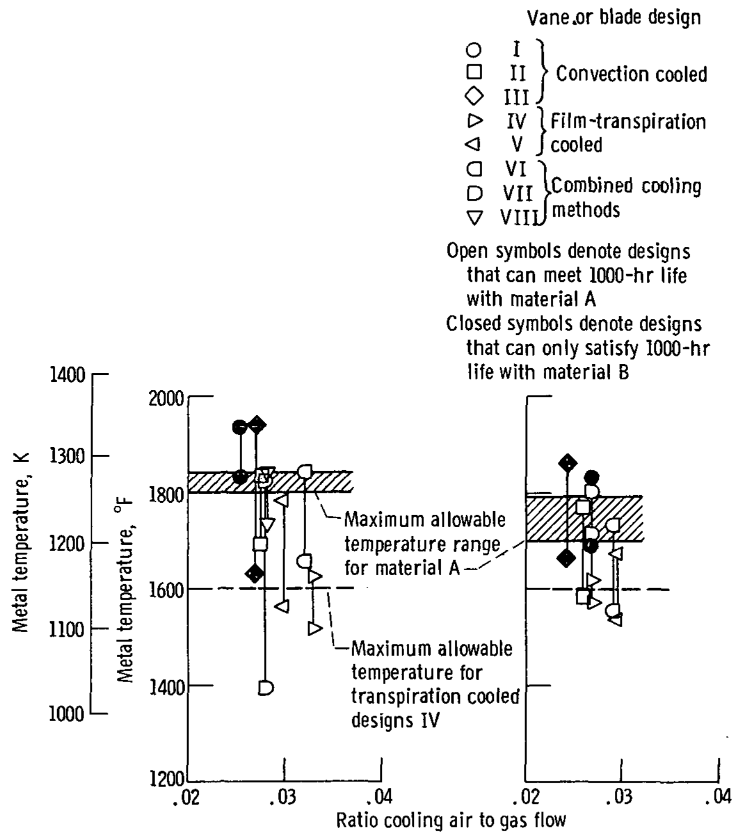
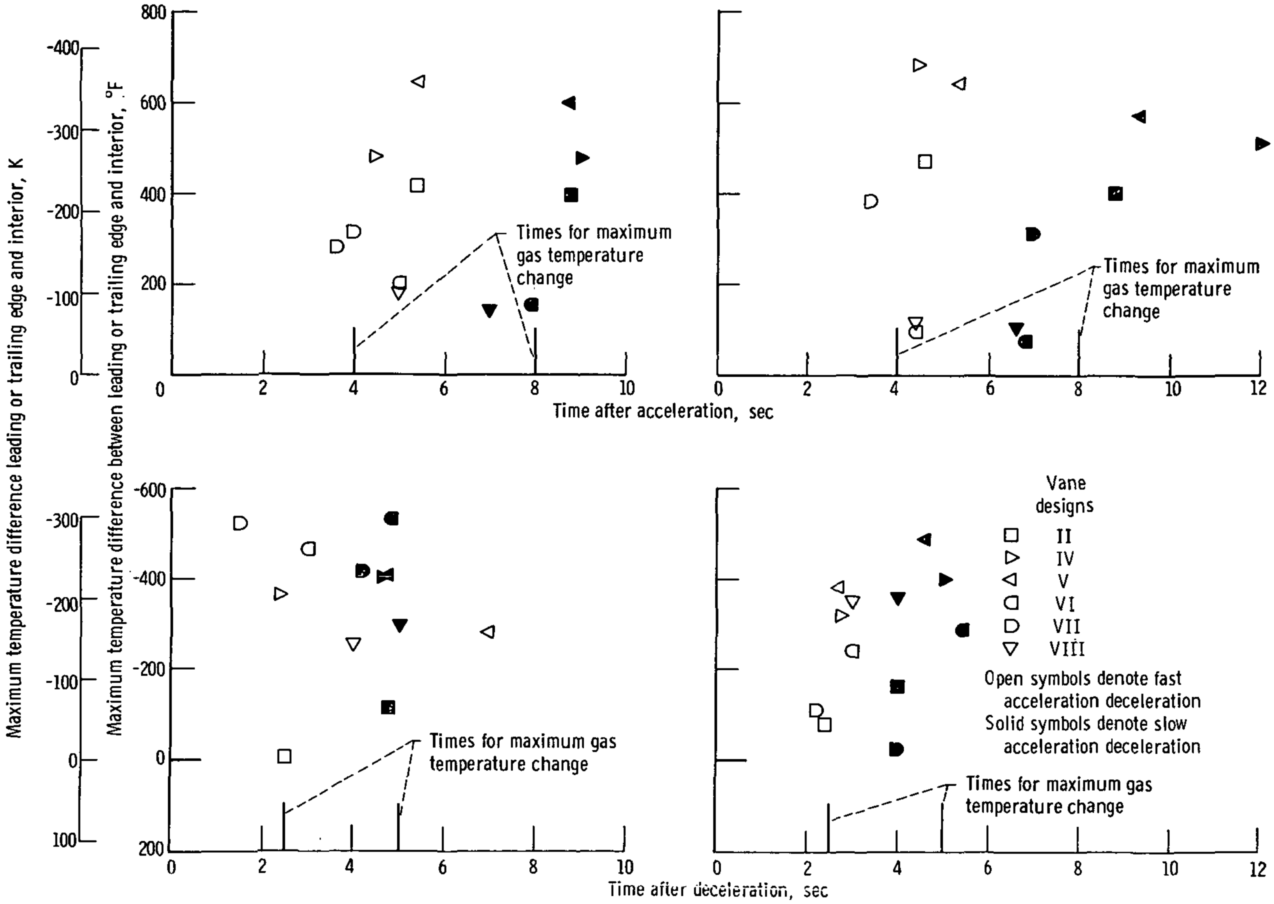


(g) Concluded.
Figure 5. - Concluded.



(a) Vanes. (b) Blades.

Figure 6. - Comparison of extremes in metal temperatures at approximately midspan of cooled turbine airfoil designs. Conditions: Cruise speed, Mach 3; cruise altitude, 75 000 feet (22.8 km); turbine rotor inlet temperature, 2200° F (1478 K).

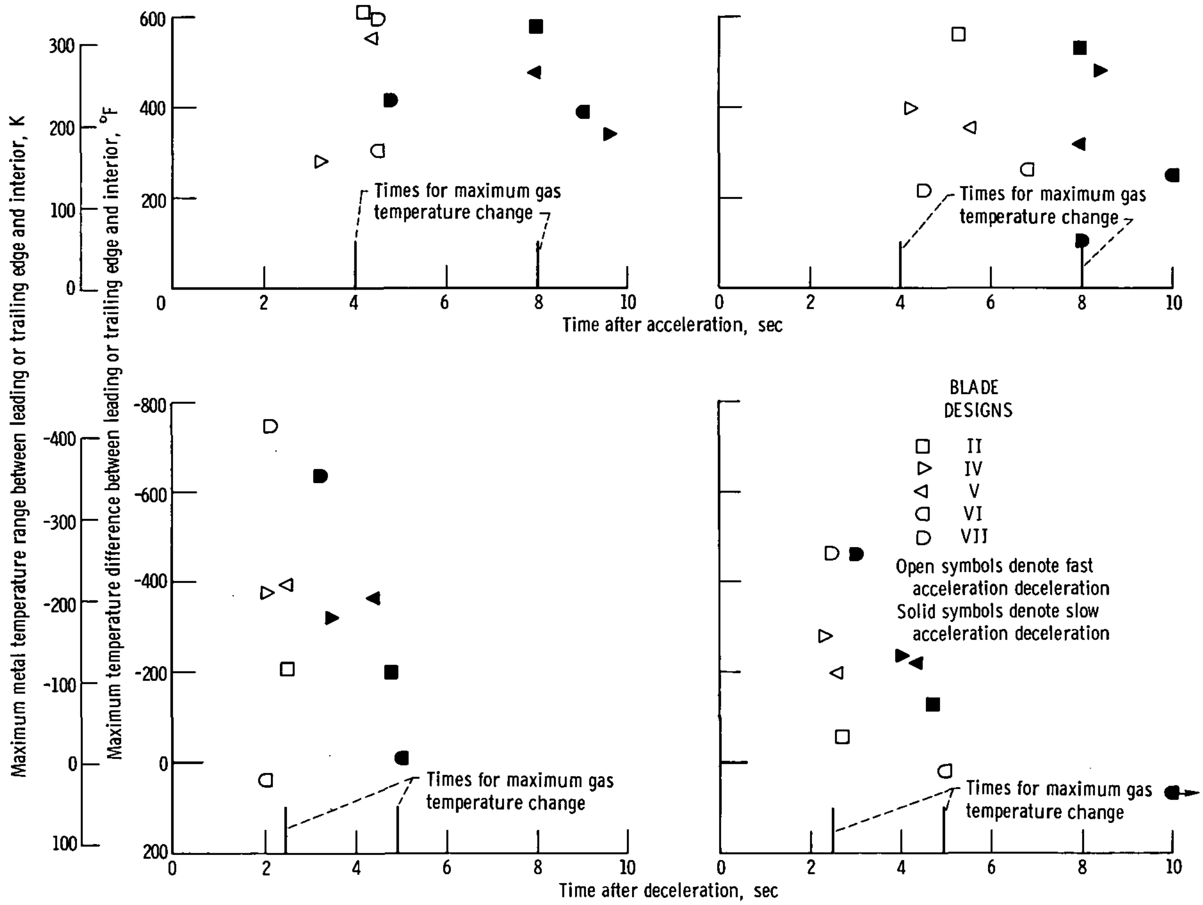


(a) Differences between leading edge and interior of vane designs.

(b) Differences between trailing edges and interior of vane designs.

Figure 7. - Maximum transient metal temperature difference during engine accelerations and decelerations.

[redacted]



(c) Differences between leading edge and interior of blade designs.

(d) Differences between trailing edges and interior of blade designs.

Figure 7. - Concluded.

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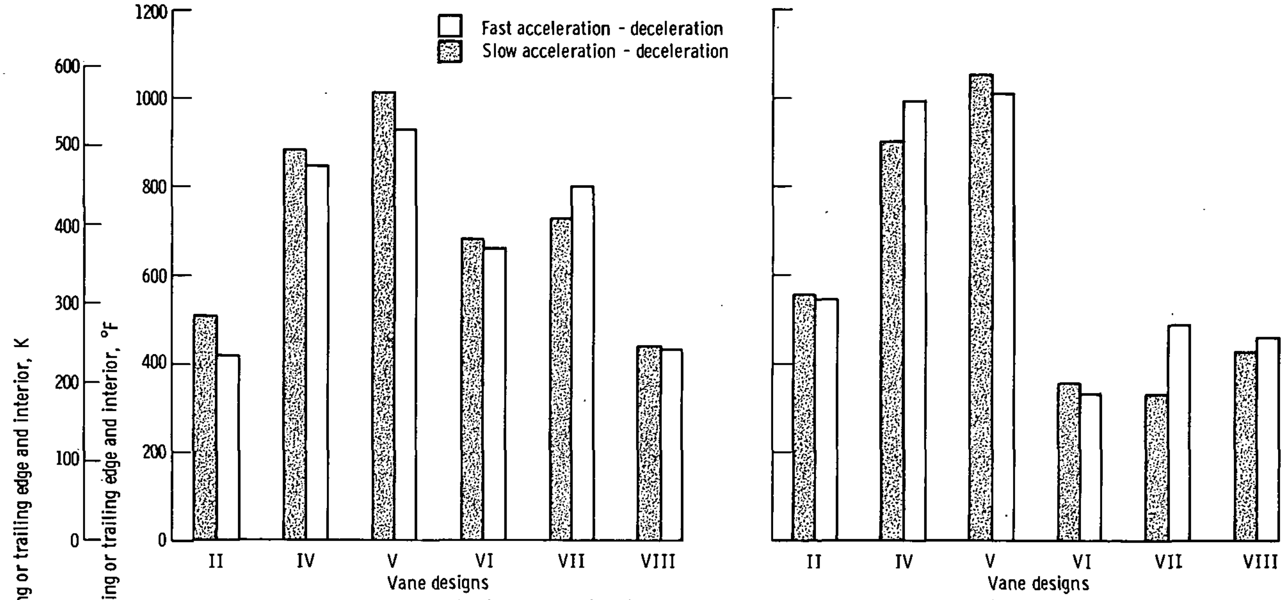
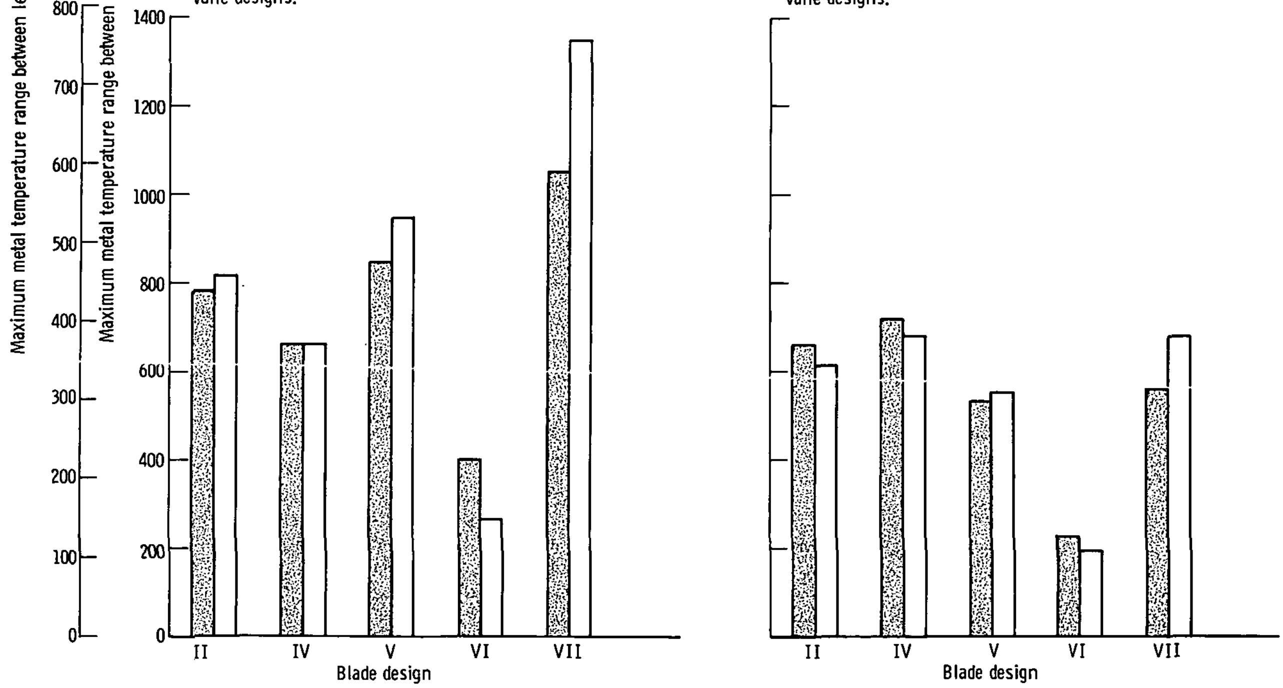


(a) Temperature difference between leading edge and interior of vane designs.

(b) Temperature differences between trailing edge and interior of vane designs.



(c) Temperature difference between leading edge and interior of blade designs.

(d) Temperature difference between trailing edge and interior of blade designs.

Figure 8. - Maximum metal temperature difference range experienced by vane and blade designs during combined engine acceleration and deceleration cycles.

[redacted]

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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