

NEW TECHNOLOGY IN TURBINE AERODYNAMICS

by

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ABSTRACT

This paper presents a cursory review of some of the recent work that has been done in turbine aerodynamic research at NASA-Lewis Research Center. Topics discussed include the aerodynamic effect of turbine coolant, high work-factor (ratio of stage work to square of blade speed) turbines, and computer methods for turbine design and performance prediction. An extensive bibliography is included.

Experimental cooled-turbine aerodynamics programs using two-dimensional cascades, full annular cascades, and cold rotating turbine stage tests are discussed with some typical results presented. Analytically predicted results for cooled blade performance are compared to experimental results.

The problems and some of the current programs associated with the use of very high work factors for fan-drive turbines of high-bypass-ratio engines are discussed.

Turbines currently being investigated make use of advanced blading concepts designed to maintain high efficiency under conditions of high aerodynamic loading.

Computer programs have been developed for turbine design-point performance, off-design performance, supersonic blade profile design, and the calculation of channel velocities for subsonic and transonic flow fields. The use of these programs for the design and analysis of axial and radial turbines is discussed.

INTRODUCTION

The Lewis Research Center of NASA has been involved in turbine research and technology for more than a quarter of a century. Application areas of interest have included propulsion and power systems of all types and sizes including airbreathing engines for both conventional and lift (VTOL) propulsion, rocket turbopump systems and space auxiliary power units driven by high energy propellants, and inert gas and metal vapor space power systems. Both axial- and radial-flow turbines have been studied, with sizes ranging from 3 to 30 inches in diameter, and with as many as 10 stages. Studies have been directed at developing new design and analysis techniques, determining fundamental effects of variables such as stage loading, reaction, solidity, Reynolds number, size, tip clearance, coolant addition, and fluid properties, screening advanced blading concepts for performance potential, and verifying the performance of turbines designed for various applications.

This paper covers work done at the NASA-Lewis Research Center in three areas that have been of interest in recent years. The first area is that of cooled turbine aerodynamics. Specific thrust or power output can be increased, with a consequent decrease in engine diameter and weight, by increasing turbine inlet temperature. Higher inlet temperatures require increasing amounts of blade-coolant air. Research is necessary if the turbine is to accept, without severe degradation in aerodynamic performance, the larger blade-coolant flows.

The second area is that of high work-factor (ratio of stage work to square of blade speed) turbines, which are often the outgrowth of the desire for lower fuel consumption and lower noise. These requirements result in higher bypass-ratio engines, with many turbine stages required to drive the high-flow low-speed fan. The work in this area is aimed at increasing the stage-work output of the low-speed turbine so as to be able to reduce the number of stages without severe penalties in turbine efficiency.

The third area to be discussed is the continuing effort that has been and is being devoted to developing and updating computer programs for flowpath design, blading design, and performance prediction. As computers become larger and faster, it becomes possible to

use more rigorous and complex design and analysis procedures.

The work being done in these three areas is discussed with emphasis on the new technology that is being generated. The work is briefly described and available results are presented along with a complete bibliography of published reports.

COOLED TURBINE AERODYNAMICS

The turbine inlet temperature of airbreathing gas turbine engines has continually increased over the past years and will continue to increase (Figure 1) in the years to come. This has been made possible by advances in both material technology and the use of advanced turbine blade cooling techniques. Normally, cooling air is bled from the discharge of the compressor and is directed through the stator and rotor blading as well as other parts to provide adequate cooling. The effect of the coolant on the aerodynamic performance of the turbine depends, among other things, on the type of cooling (e.g., convection, impingement, film, or transpiration) involved, the location and direction of coolant injection into the main stream, the temperature of the coolant relative to the main hot gas stream, and the amount of coolant required. A number of these factors are being studied by the use of various two-dimensional and annular cascades as well as rotating stage tests (refs. 1 to 13).

Single-Stage Turbine Tests

Tests were made of various types of stator cooling techniques in a 30-inch diameter turbine cold-air test facility where the temperature level of the coolant was the same as the turbine primary inlet air. The types of stator blades studied are shown in Figure 2. They are the slotted trailing-edge blade and two types of porous-skin blades. All three have the same outer-shell profile as the solid uncooled blade shown at the top of the figure. The trailing-edge-slot blades ejected all of the cool-

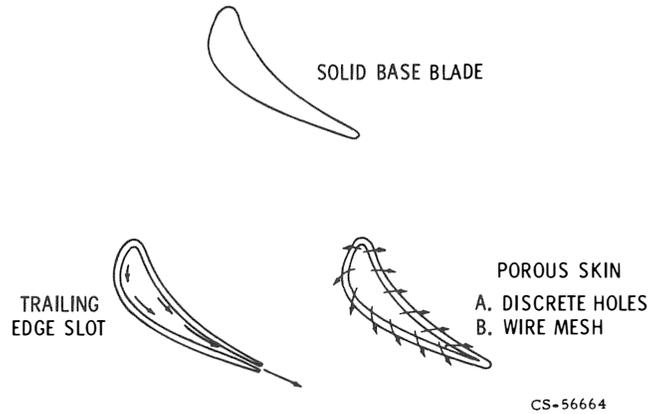


Figure 2. Cooled Turbine Blade Types.

ant through a slot in the trailing-edge in a direction generally the same as the adjacent primary air. The two types of porous-skin blades tested, the discrete holes and the wire mesh, ejected air around the entire periphery of the blading and in a direction generally normal to the adjacent primary air.

The three types of cooled blades tested are shown in Figure 3. The blade on the left had a slot extending along the entire length of the trailing edge. The variation in coolant flow through the discrete-hole blade was controlled by varying the size and spacing of the holes around the surface. The wire-mesh blade was made by wrapping and welding a wire mesh around internal struts that formed individual cooling chambers. The electron beam welds between the mesh and the internal strut can be seen in the figure. The distribution of the coolant around the periphery of the wire mesh blade was controlled by the size of the metering holes in the orifice

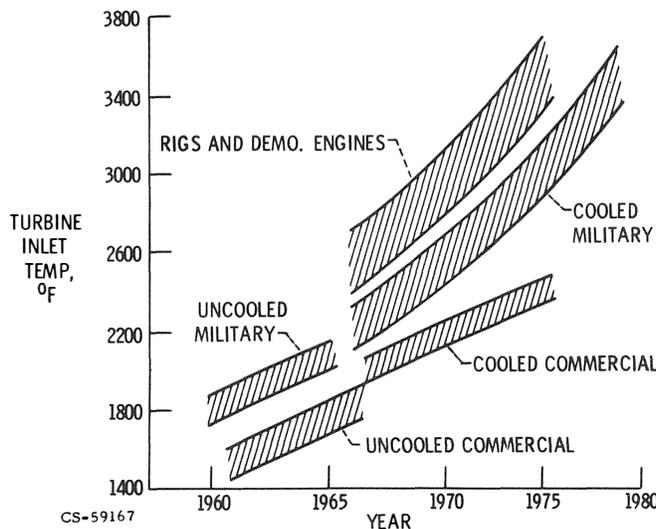


Figure 1. Trends in Turbine Gas Temperature.

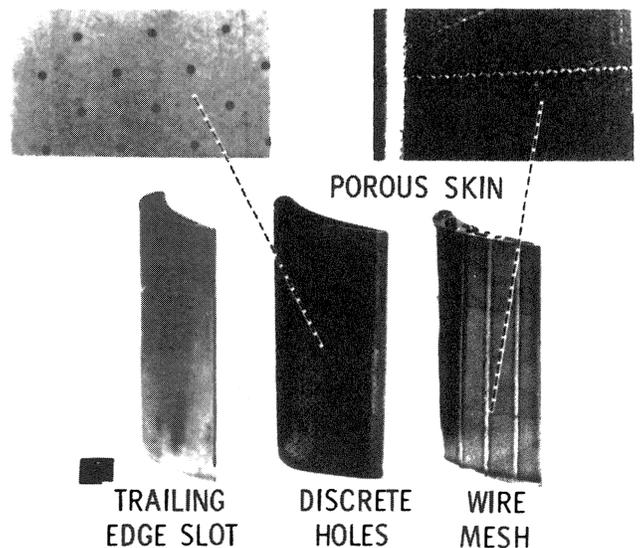


Figure 3. Stator Blades Tested.

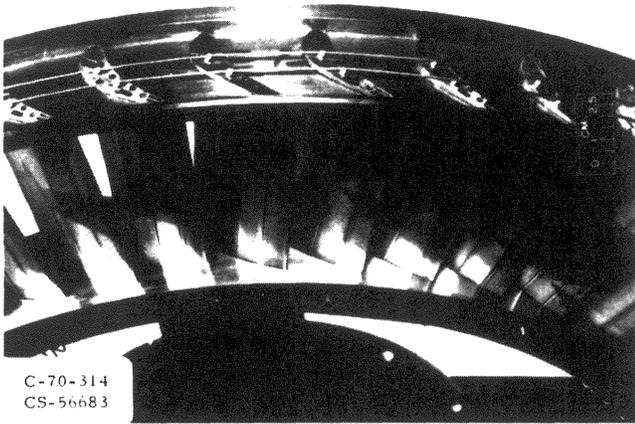


Figure 4. Wire Mesh Stator Assembly.

plate that can be seen at the top of the blade. A photograph of the wire-mesh stator assembly is shown in Figure 4.

Total pressure surveys in both the radial and circumferential directions were made downstream of each stator assembly with the rotor removed to determine the relative effect of the coolant on stator loss. Figure 5 shows a comparison of wake traces made at a constant radius downstream of the stator with trailing edge slots and the stator with discrete holes. The traces are made across one blade wake. The depth and width of the total pressure drop traces are an indication of the total frictional loss up to the plane of measurement and indicate a considerable difference in loss pattern and magnitude as a function of the type of cooling utilized. For the porous-skin blade, the wake thickens and loss increases continually as the coolant flow increases. For the trailing-edge-slot blade, the loss starts to increase with coolant flow, but the wake then becomes energized with further coolant addition. Higher coolant pressures are required to obtain the higher coolant flows, and this accounts for the energization of the wake.

Figure 6 is a photograph of the rotor used for rotating stage tests of each type of stator blade tested. For these tests, solid rotor blades were used with thick pro-

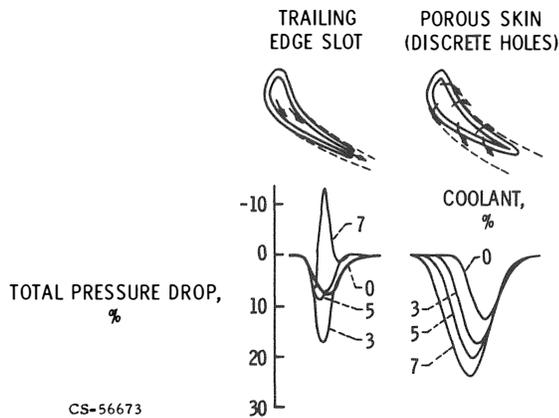


Figure 5. Stator Wake Traces.

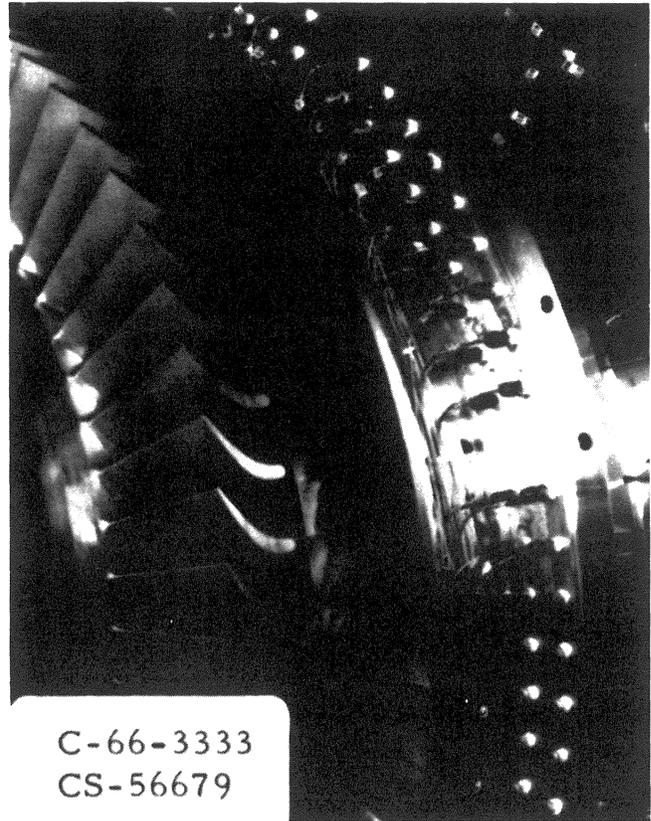


Figure 6. Rotor Installed.

files and trailing edges, such as would be required to allow adequate room for coolant flow inside the rotor blades. Subsequent tests will utilize this same profile with hollow blades for rotor coolant tests. A comparison of the stage efficiencies for the four turbines with different stator configurations is shown in Figure 7. As indicated, the efficiency of the turbine with trailing edge ejection was very close to that of the solid blade turbine over the range of coolant flows tested. This means that the energy of the coolant was nearly equal to that of the primary airstream and results from (1) ejection in a direction essentially the same as the primary air (Figure

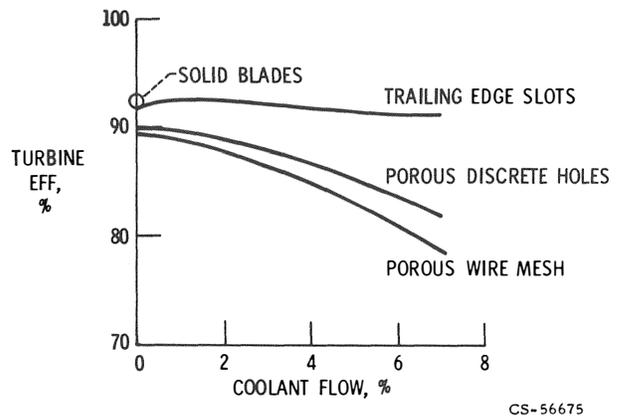


Figure 7. Turbine Stage Efficiencies.

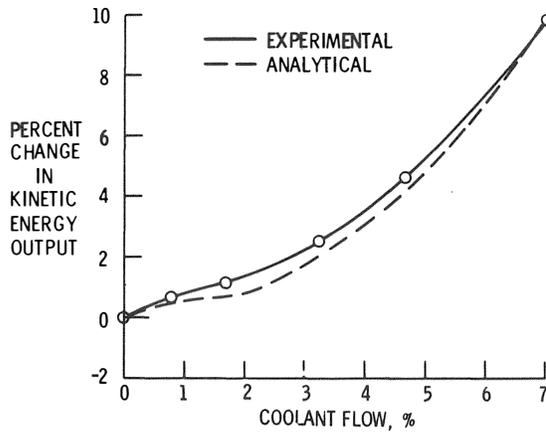


Figure 8. Comparison of Experimental and Analytical Performance of Stator Blading with Trailing-Edge Coolant Ejection.

2) and (2) a small required pressure drop inside the blade. The efficiency of both turbines using porous stator blading decreased considerably as coolant flows increased. It should be noted that the blading tested only simulated that required for an actual engine. Heat transfer considerations would necessitate a more complicated blade internal structure, with associated added pressure drops.

Comparison of Experimental with Predicted Results

Analytical models are being used to predict the effect of coolant on blade row efficiency (Refs. 14 to 16) as a function of the variables previously described. Reference 16 uses a model that considers the effect of the relative velocity of the coolant at the exit of the blade on the kinetic-energy output of the blade row relative to the solid uncooled blade. The predicted change (Ref. 16) in kinetic-energy output of a stator blade row with trailing edge ejection is compared to experimental results in Figure 8. Good agreement is seen to exist. Such analytical treatments will be refined as more experimental data become available. Currently, cascade testing is in progress to investigate the aerodynamic performance of blades with coolant holes systematically placed in different locations and to evaluate the effect of coolant-to-primary air temperature ratio.

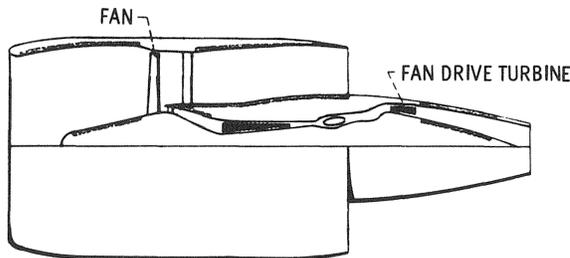


Figure 9. High Bypass Ratio Fan Engine.

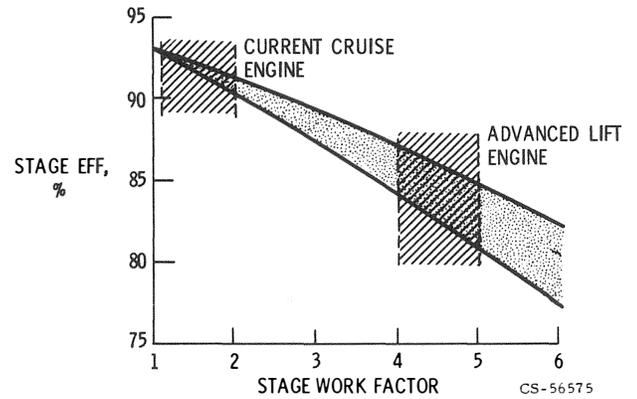


Figure 10. Effect of Turbine Stage Work on Efficiency.

HIGH WORK-FACTOR TURBINES

The trend in many fan engines is towards higher bypass ratios. As bypass ratio increases, the relative diameter of the direct fan-drive turbine (Figure 9) decreases, which results in much lower turbine blade tip speeds than presently encountered. This in turn results in a large increase in the number of turbine stages to develop the required power at high efficiency. Consequently, it is desired to reduce the number of stages, which in turn increases the loading, or "work factor," of the remaining stages. Stage work factor (ratio of stage work to square of blade speed) relates the work extracted to the blade energy, and for current cruise engines (Figure 10) ranges in value from 1 to 2. However, for advanced high-bypass-ratio engines, values of stage work factors as high as 5 will be required. As indicated by Figure 10, stage efficiency drops markedly as work factor increases. The shaded area between the two curves indicate the uncertainty in efficiency level due to the lack of turbine data at work factors over 2.

A program is in progress to study three turbines having average stage work factors of 3, 4, and 5. The objective is to establish the level of efficiency at the higher work factors using the best design tools available and utilizing unconventional blading where appropriate. Figure 11 shows a flow-path schematic of the turbine designed (Refs. 17 and 18) for an average stage work factor of 3. It is a 3 stage turbine designed for nearly axial flow leaving the turbine. It is this axial-flow requirement that makes the last-stage rotor blade most criti-

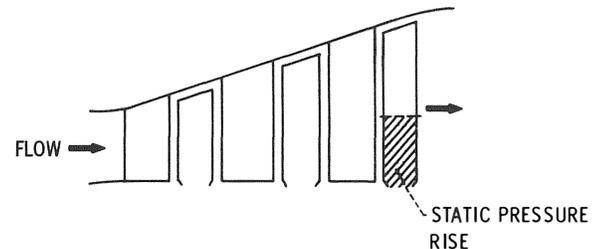


Figure 11. Three-Stage Fan Drive Research Turbine.



Figure 12. Advanced Blade Concepts.

cal because of the resultant static pressure rise across the bottom third of the blade (shaded area).

Programs (Refs. 19 to 23) investigating advanced concepts (tandem and jet flap shown in Figure 12) for highly loaded blades indicated that for such conditions tandem blades have a potential advantage over plain blades in preventing flow separation off the blade provided that the Mach number is not too high. This is implied by wake traces (Figure 13) taken downstream of tandem and plain blades. The area deficit indicated by the wakes are an indication of blade row losses, and it can be seen that the tandem blade has a higher efficiency. The turbine shown in Figure 11 has been tested with various combinations of plain and tandem blading. A photograph of the plain rotor assembly is shown as Figure 14. The results of this program will be published as NASA contractor reports.

The other two turbines being fabricated for this program (work factors of 4 and 5) will have $3\frac{1}{2}$ and $4\frac{1}{2}$ stages, respectively. The additional $\frac{1}{2}$ stage for both turbines are outlet guide vanes used to turn the flow from the last-stage rotor to axial. This allows a more equal distribution of work split between the stages and a more conservative last-stage rotor at the expense of an extra $\frac{1}{2}$ stage. These two turbines are being fabricated for experimental tests.

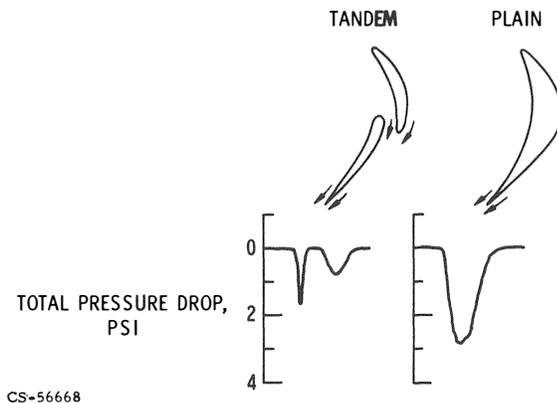


Figure 13. Wake Traces from Tandem and Plain Blades.

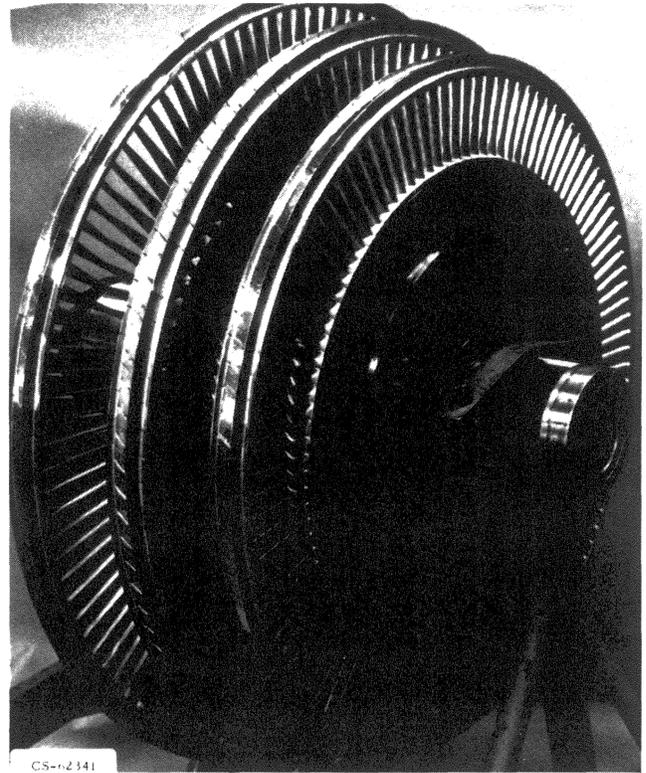


Figure 14. High Work-factor Turbine Rotor.

COMPUTER PROGRAMS FOR DESIGN AND ANALYSIS

During the past decade, NASA-Lewis has developed, or funded the development, and published many computer programs for the design and analysis of turbines, both axial and radial. There are programs for design geometry and performance of axial-flow turbines, off-design performance of axial and radial turbines, analysis of flow in axial, radial, or mixed flow turbomachine blade channels, compressible-flow boundary layer calculations, and blade profile design for supersonic-flow blade rows. Each of the types of programs is briefly discussed in this section.

Design Geometry and Performance

The first phase of any turbine analysis or design is the determination of number of stages, flowpath annulus dimensions, velocity diagrams, and an associated estimate of turbine efficiency. For a preliminary analysis, complete design accuracy and detail are not necessary; therefore, approximate and rapid generalized procedures are sufficient to yield the desired turbine overall geometry and performance. For the design of a particular turbine, more detail and accuracy are desired and, consequently, more complex procedures are required. Programs for both types of analyses for axial-flow turbines are available.

A program for the preliminary design analysis of axial-flow turbines is presented in Reference 24. This program is based upon an analysis of the flow at the

turbine mean diameter, and radial gradients of flow properties are not considered. For an given turbine, all stages are specified to have the same shape velocity diagram, the particular shape depending on the stage work-factor value and the specified type of velocity diagram. Three basic types of velocity diagrams can be considered: symmetrical (50 percent reaction), zero exit swirl, and impulse. These three types of velocity diagrams are illustrated in Figure 15 for three values of stage work factor.

Input design requirements for the program of Reference 24 include power or pressure ratio, mass flow rate, inlet temperature and pressure, and rotative speed. The design variables include number of stages, inlet and exit diameters, and stator-exit angle or turbine-exit radius ratio. The program output includes inlet and exit annulus dimensions, exit temperature and pressure, total and static efficiencies, flow angles, and velocities. The turbine efficiency correlation used in the program is based on the methods of References 25 and 26, where stage work factor is the primary determinant of stage efficiency.

For the design of a turbine, it is desirable to analyze the entire flow field from hub to tip rather than just the mean section. This is important because of the large radial variations in flow angles and velocity that can occur. These variations are illustrated in Figure 16 by an example set of velocity diagrams for the hub, mean, and tip sections of a blade with a radius ratio of 0.6. The radial variation in diagram shape is considerable. The mean-section diagram is symmetrical (50 percent reaction). The associated hub diagram is nearly impulse while the tip diagram is very conservative with high reaction. It is, therefore, important to be able to analyze the flow at all radii in order to assure satisfactory conditions at all sections.

An axial-flow turbine design analysis program that includes radial gradients in flow is presented in Reference 27. This program is based on an axisymmetric streamline analysis of flow from the inner to the outer wall of the flow passage. Non-free vortex turbine designs as well as free-vortex designs can be analyzed. The program is capable of analyzing both single and multi-

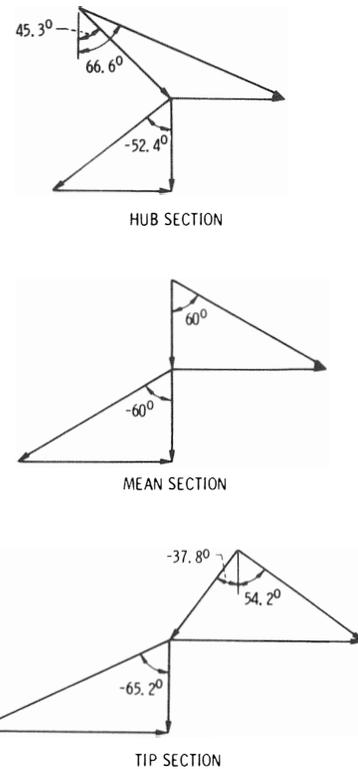


Figure 16. Radial Variation in Velocity Diagrams (Radius Ratio, 0.6).

spool units. A maximum of three spools, each with up to eight stages, is allowed. Radial variation effects of the following quantities are taken into account: inlet conditions, streamline angle and curvature, loss coefficient, and meridional velocity. Losses can be input or calculated internally on the basis of a pressure-loss-coefficient correlation whose development is reported in Reference 28.

Off-Design Performance

A turbine is designed for a single operating condition called the design point. In many cases, particularly in airbreathing propulsion engines, the turbine is required to perform at many conditions other than the design point. The turbine work output and flow rate can be varied by adjusting the rotative speed and/or pressure ratio. Under these different running conditions, the turbine is said to be operating off-design. To completely predict a turbine's characteristics, it becomes necessary to compute the off-design performance over a wide range of operating conditions.

Computer programs for calculating off-design performance of radial-inflow and axial-flow turbines are reported in References 29 and 30, respectively. The radial turbine program is based on a meanline analysis of the flow and is for a single-stage machine. The axial turbine program is applicable to turbines having any number of stages up through eight. The program can be run as a meanline analysis or can allow for radial

STAGE WORK FACTOR	DIAGRAM TYPE		
	ZERO EXIT SWIRL	IMPULSE	SYMMETRICAL
1			
2			
4			

Figure 15. Effect of Diagram Type and Stage Work Factor on Velocity Diagram Shape.

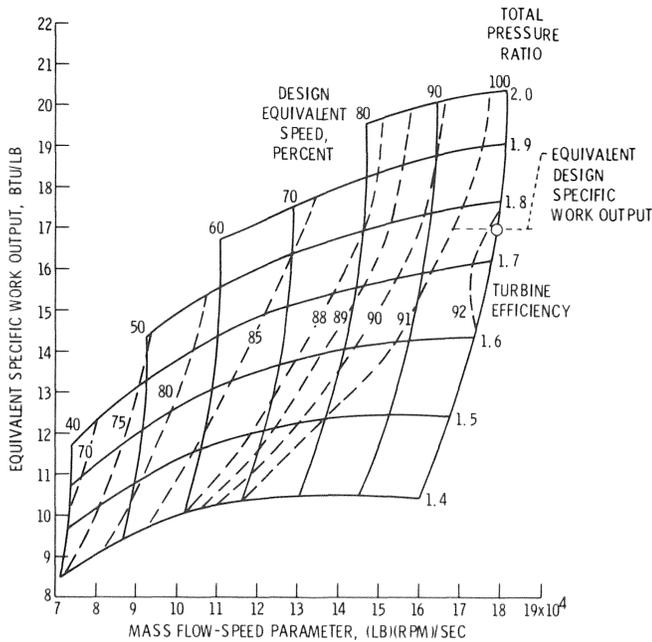


Figure 17. Turbine Performance Map.

variations in loss and flow conditions. Two loss options are provided with a kinetic energy coefficient-inlet recovery coefficient method and a total-pressure loss coefficient method. The analysis is applicable from zero speed

to high speed, and the work done may vary up to the maximum as limited by discharge annulus area choking.

The value of an off-design performance program depends upon its ability to accurately predict performance over a wide range of conditions. An experimentally determined performance map, as reported in Reference 3, for a single-stage axial-flow turbine is shown in Figure 17. Data was obtained over a range of speed from 40 to 100 percent of design and for pressure ratios of 1.4 to 2.0. The program of Reference 30, with coefficients selected to match design-point performance, was used to predict the turbine work and flow for the same range of conditions. Over the entire map, the computed performance was within 1 percent of the experimental values. Thus, the validity of the program was demonstrated. It should be noted, however, that such a good simulation cannot be expected for all turbines.

Blading Flow Analysis

In order to insure good performance from turbine blades, it is necessary to determine and control the flow distribution throughout the blading flow passages. A typical flow passage for an axial-flow turbine is shown in Figure 18. Velocity gradients occur across the channel from the suction to the pressure surface as a result of the static pressure difference required to turn the flow. Radial variations in velocity occur as a result of radial equilibrium considerations.

The determination of a proper blade profile depends upon our ability to calculate the blade-row flow field and, thereby, determine the velocity at the blade surfaces. The actual velocity distribution cannot be calculated because of the extreme complexity of nonsteady, viscous, three-dimensional flow through geometrically complex passages. To calculate a theoretical velocity distribution,

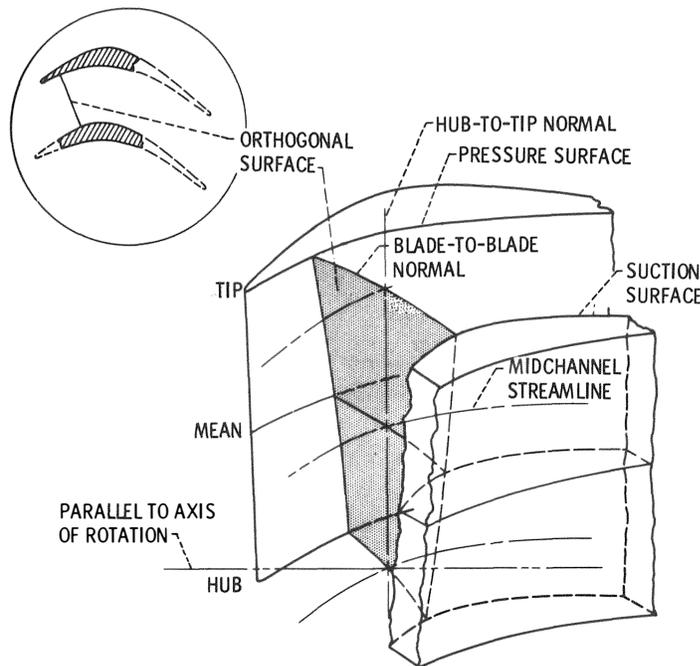


Figure 18. Pair of Typical Turbine Blades with Three-Dimensional Orthogonal Surface Across Flow Passage.

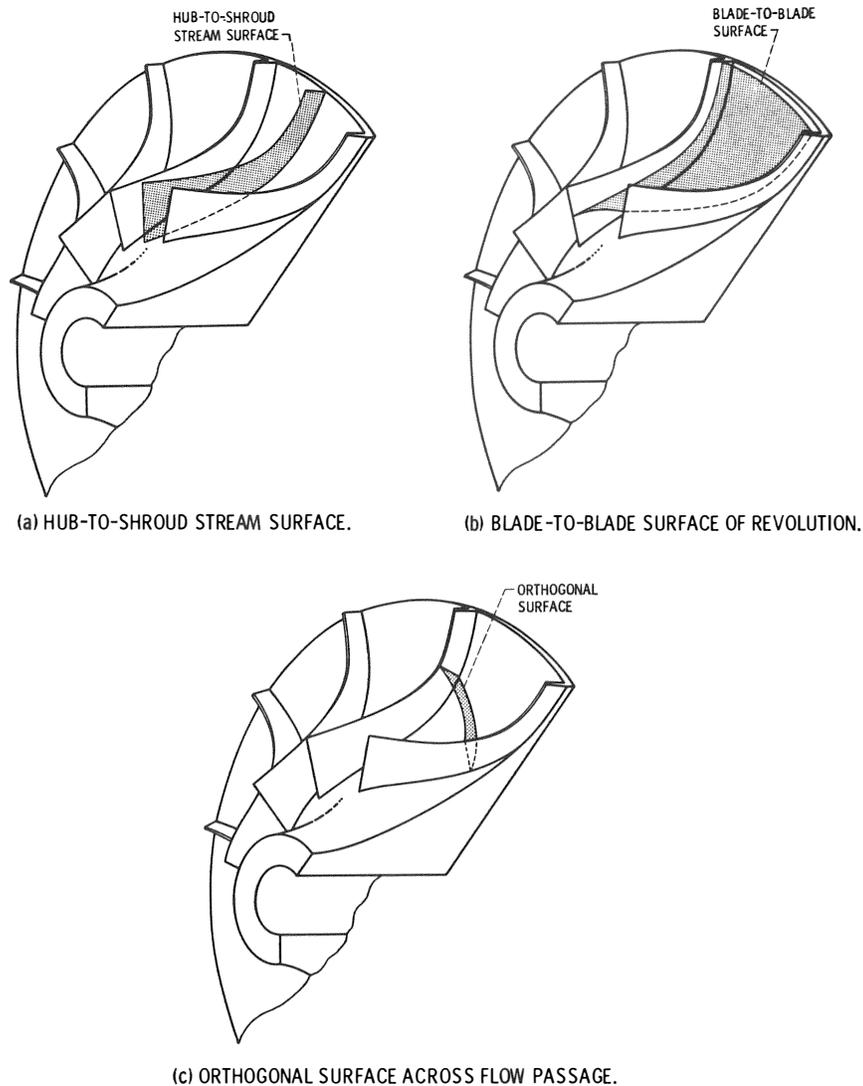


Figure 19. Two-Dimensional Surfaces for Flow Analysis in a Radial-Inflow Turbine.

therefore, certain simplifying assumptions must be made. First of all, three-dimensional flow is simplified to two-dimensional flow to provide solutions in the meridional, blade-to-blade, and orthogonal surfaces of blade passages. These surfaces for a radial-inflow turbine are illustrated in Figure 19. In addition, the flow is assumed to be steady relative to the blades, isentropic, inviscid, and ideal. Two basic computation techniques are used to obtain the desired flow solutions: a stream function method, which gives results for the entire passage, and a velocity gradient method, which is good only for the guided part of the passage.

Several computer programs have been written at NASA-Lewis for the analysis of flow through turbomachine blading by stream function methods. Most of these programs are for blade-to-blade (surface of revolution) analysis. A program described in reference 31 can be used to analyze axial, radial, or mixed flow. In accordance with the constraints associated with the stream function method, the flow must be subsonic

throughout the entire solution region. The program described in Reference 32 supersedes that of Reference 31 in that it performs all the same calculations and, in addition extends the solution to transonic (local supersonic velocities) flow problems. Transonic solutions are obtained by using a velocity gradient equation to extend a preliminary subsonic stream-function solution. Shown in Figure 20(a) are the blade and channel profiles for the mean section of a stator blade row investigated in Reference 1. The experimentally-determined surface Mach numbers, some of which are in the transonic region, are compared with the Reference 32 computer program results in Figure 20(b). The experimental and calculated Mach numbers agree quite well over most of the surface length.

Flow in tandem or slotted blade rows or in blade rows with splitters (partial blades between each pair of full blades) can be analyzed using the program described in Reference 33. Experimental surface Mach numbers obtained from static pressure measurements made on a

turbine tandem blade cascade, having the profile shown in Figure 21(a), are compared with the computer program results in Figure 21(b). There is close agreement between computed and experimental values on the surfaces of both the front and rear blades. A program described in Reference 34 can be used to obtain a detailed solution in the leading or trailing edge regions of any blade or in the slot region of tandem or slotted blades.

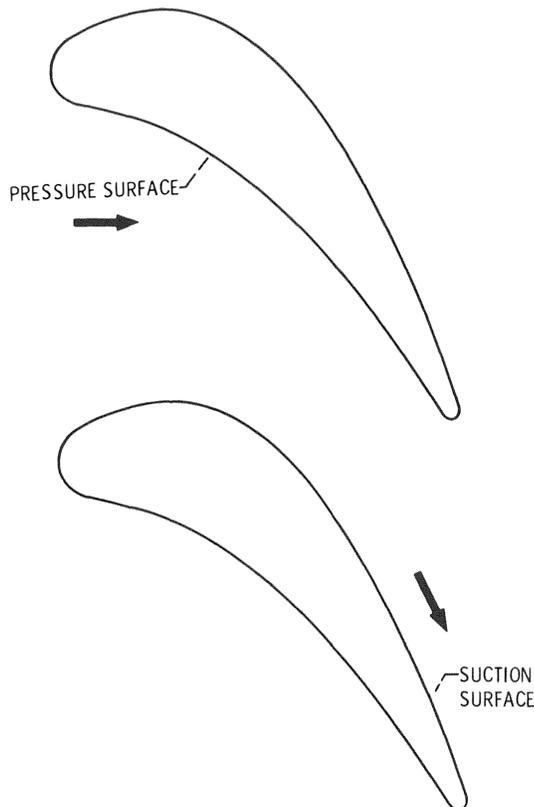
Flow in the meridional plane (mean radial-axial flow surface) of any axial, radial, or mixed flow turbomachine can be analyzed by the program described in Reference 35. Here, too, transonic solutions can be obtained by the use of a velocity gradient equation to extend a preliminary subsonic stream-function solution.

It is possible to use a velocity gradient method of analysis alone to obtain subsonic, transonic, or supersonic solutions. The velocity gradient analysis is often called a stream-filament analysis because the velocity gradient equation involves the streamline, or stream filament, curvature and position. A velocity gradient method of analysis can only give solutions within a guided passage; that is, a passage where both ends of all streamline orthogonals intersect a solid boundary. Therefore, the usefulness of this method depends on the degree of flow guidance provided by the turbine blades.

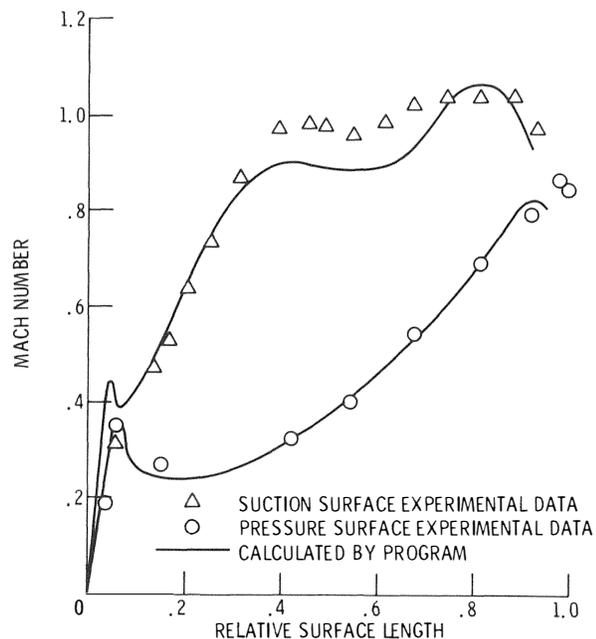
Several computer programs for the analysis of flow through turbomachine blading by velocity gradient methods have also been written at NASA-Lewis. One pro-

gram that was used for many years is described in Reference 36 and is limited to axial-flow turbines. This program has now been superseded by the more general and easier to use program described in Reference 37. This program can be used to analyze axial, radial, or mixed flow turbines or compressors. Velocity gradient equations are used to determine velocity variations both from hub to tip along meridional streamline orthogonals and from blade to blade along hub, mean, and tip streamline orthogonals. This results in a flow solution for an orthogonal surface, which is illustrated in Figure 18 for an axial turbine and Figure 19(c) for a radial turbine. Computations are made for a number of these surfaces along the blade passage. This program not only yields surface velocities, but also yields a two-dimensional estimate for design and choking mass flows.

Velocity gradient methods have also been used to obtain meridional-plane and blade-to-blade plane solutions. The basic method for a meridional-plane analysis for mixed-flow centrifugal impellers is presented in Reference 38, which uses the velocity gradient equation along streamline orthogonals. Since the orthogonal lengths are not known in advance, it was more convenient to base a computer program on the use of the velocity gradient equation along fixed straight lines, which were called quasiorthogonals. Such a program for a meridional-plane analysis of a radial-inflow turbine impeller is presented in Reference 39. A program for a blade-to-blade



(a) BLADE AND CHANNEL PROFILES.



(b) SURFACE VELOCITIES.

Figure 20. Comparison of Experimental and Computed Surface Velocities on Plain Blade.

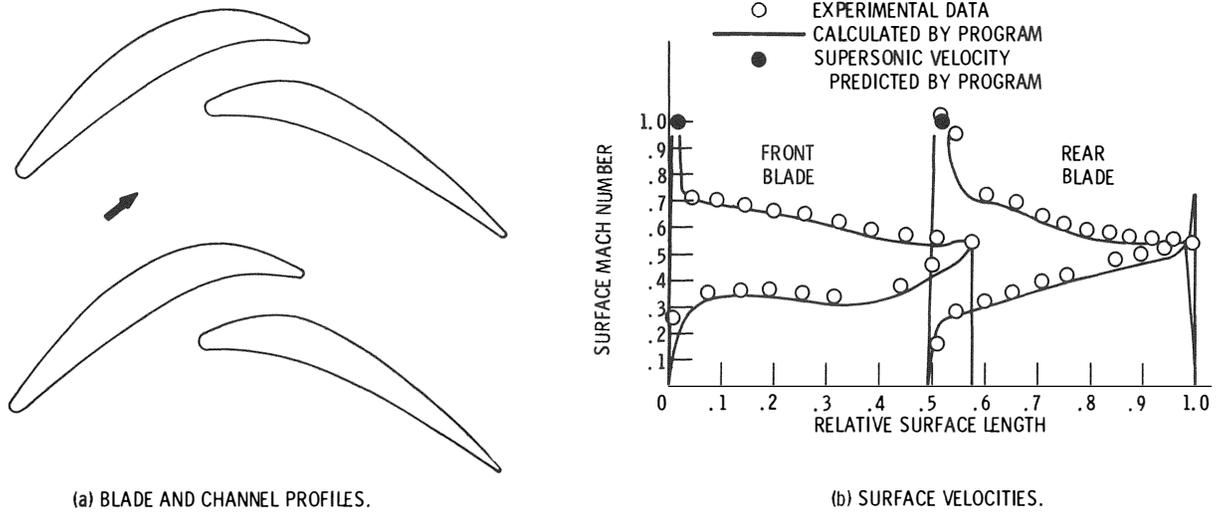


Figure 21. Comparison of Experimental and Computed Surface Velocities on Tandem Blade.

plane analysis using quasi-orthogonals for a radial-inflow turbine impeller is described in Reference 40.

Blading Boundary Layer

The primary cause of losses in a turbine is the boundary layer that builds up on the blade and end-wall surfaces. There is a basic viscous loss associated with the boundary layer itself, and then there are associated trailing edge and mixing losses. In order to analytically predict these losses, it is necessary to be able to simulate

the boundary layer growth on the blade and therefrom compute the displacement and momentum thicknesses.

A computer program that gives the solution of the two-dimensional, compressible laminar and turbulent boundary-layer equations in an arbitrary pressure gradient is reported in reference 41. The methods of references 42 and 43 are used for calculation of the laminar and turbulent cases, respectively. Transition from laminar to turbulent boundary layer can be predicted by the program or specified by the user. Separation is also

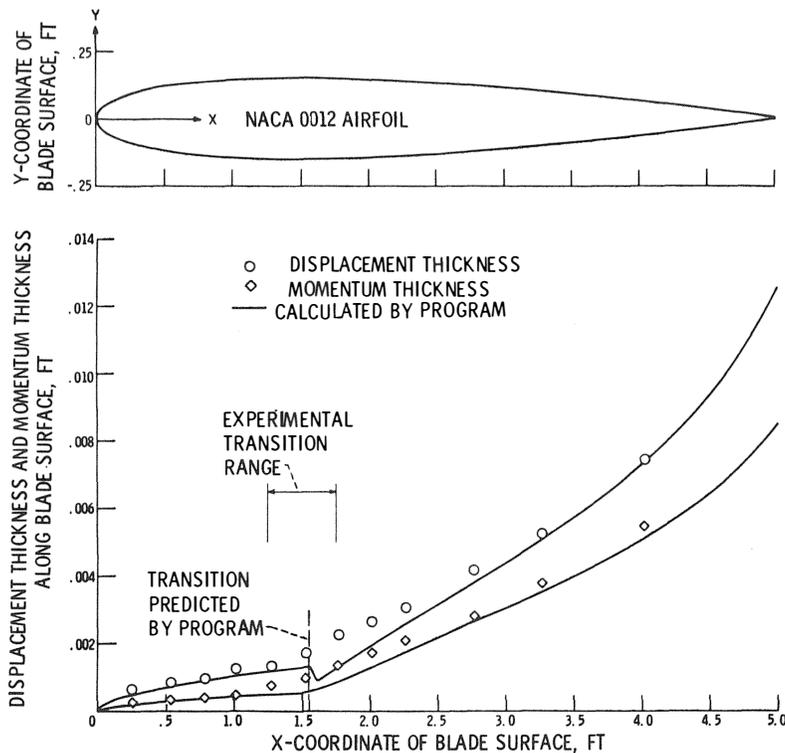


Figure 22. Comparison of Analytical Results with Experimental Data on NACA 0012 Airfoil.

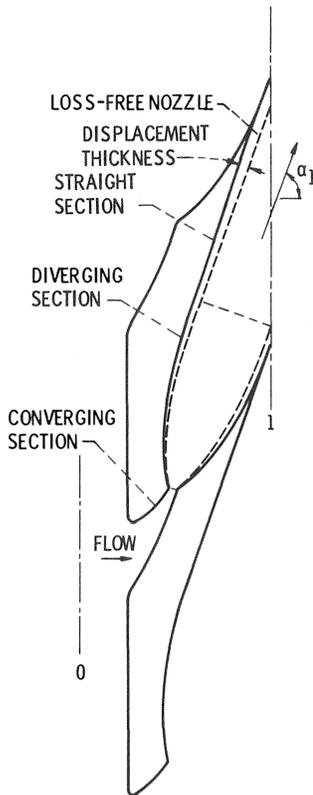


Figure 23. Supersonic Nozzle Design.

predicted by the program. The program input consists of surface geometry, a pressure or velocity distribution external to the boundary layer (this can be obtained from one of the computer programs discussed in the previous section), wall temperatures, and initial values of displacement and momentum thickness, if any. The output includes all the principal boundary-layer parameters, such as displacement thickness, momentum thickness, form factor, skin friction, heat transfer, and velocity profiles. The program will handle any two-dimensional case with subsonic or supersonic Mach numbers.

A comparison of analytical results with experimental data is presented in Figure 22 for an airfoil section. The boundary layer was assumed to begin at a stagnation point at the leading edge of the blade. Transition was predicted by the program and occurred within the range in which it was measured experimentally. The agreement between experimental and analytical displacement and momentum thicknesses is good except for a region around the transition point. This is understandable since transition occurs gradually in the actual flow, but is forced to occur at a single point in the program.

Blading Profile for Supersonic Flow

A supersonic turbine stage is one in which the absolute velocity at the nozzle exit and the relative velocity at rotor inlet are supersonic. The design of both the stator and rotor blading is accomplished in

a similar manner. First, the ideal (based on inviscid flow) passage (stator or rotor) is designed by the method of characteristics as applied to the isentropic flow of a perfect gas. Boundary layer parameters are then calculated for the ideal passage and the final profile is obtained by correcting the ideal profile for the displacement thickness.

The design of the ideal stator blading is based on establishing uniform parallel flow at the blade exit in the minimum possible distance. The computer program described in reference 44 is used for this purpose. A typical sharp-edged-throat nozzle of this type (shown in Fig. 23) consists of three sections: (1) a converging section, (2) a diverging section, and (3) a straight section. The converging section produces the flow turning with small losses and is not designed by the computer program. The symmetric diverging section accelerates the flow to the desired freestream Mach number at the blade exit. The straight line segment, parallel to the flow direction, completes the nozzle profile.

The boundary layer parameters (displacement and momentum thicknesses) are calculated using the methods of the previously discussed computer program described in reference 41. A complete description of the computer program for the design of supersonic nozzles corrected for boundary layer displacement thickness is given in reference 45. The program input consists primarily of the nozzle exit Mach number, nozzle angle, specific heat ratio, and total flow conditions. The program output gives the corrected nozzle profile.

Experimental pressure distributions were obtained at design and off-design conditions in a nozzle designed by this computer program and are shown in Figure 24. The theoretical pressure distribution as predicted by the program for the design condition is also shown. At the design pressure ratio, the agreement between theory and

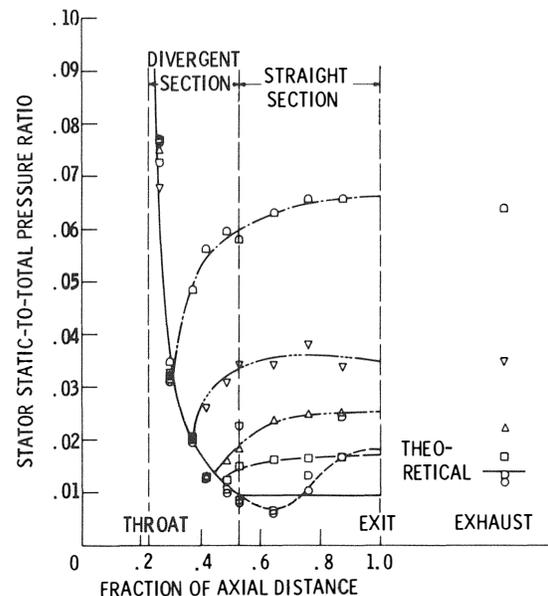


Figure 24. Comparison of Experimental and Analytical Stator Pressure Distributions.

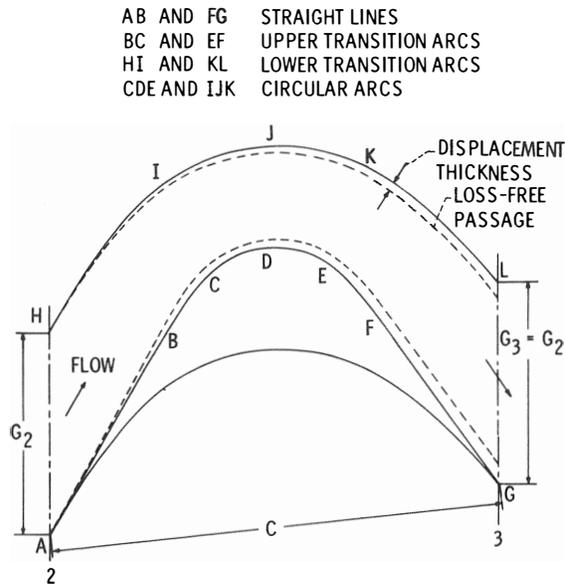


Figure 25. Supersonic Rotor Design.

experiment is good for the divergent portion of the nozzle. Along the straight section, where the pressure should be theoretically constant, the pressure first decreases below design and then increases. This behavior is apparently caused by expansion and shock waves forming on the straight section.

The design of the ideal rotor blading is based on establishing vortex flow within the blade passage. The computer program described in reference 46 is used for the calculation of the ideal passage. A typical passage (shown in Figure 25) consists essentially of three major parts: (1) inlet transition arcs, (2) circular arcs, and (3) outlet transition arcs. The inlet transition arcs (upper and lower surfaces) are required to convert the uniform parallel flow at the passage inlet into vortex flow. The concentric circular arcs turn and maintain the vortex flow condition. The outlet arcs reconvert the vortex flow into uniform parallel flow at the passage exit. Straight line segments, on the suction surface, parallel to the inlet and outlet flow directions complete the passage.

A complete description of the computer program for the design of supersonic rotor blades corrected for boundary layer displacement thickness has been given in reference 47. The program input consists essentially of the inlet and exit Mach numbers, flow angles, specific heat ratio, circular arc Mach numbers, and total flow conditions. The program output consists of the corrected rotor passage and the boundary layer parameters.

CONCLUDING REMARKS

In recent years, turbine aerodynamics research at NASA-Lewis Research Center has primarily been directed toward allowing the turbine to efficiently accept higher inlet temperature and blade loading. This is being accomplished by such means as studying the interaction between coolant and primary flows so as to be

able to obtain maximum cooling with minimum aerodynamic loss, studying the use of advanced blading concepts designed to efficiently increase stage work output, and improving computerized design techniques to yield more complete flow analyses and thus allow the designer to avoid aerodynamic problem areas. The discussion in this paper has covered primarily these areas.

There are still additional gains to be made by further work in these areas. In addition, other areas requiring study include (1) unsteady flow effects and associated stator-rotor interactions, (2) noise generation in the blading, and (3) supersonic flow analyses including shocks. These latter areas are expected to receive increased emphasis in the future.

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