Design of a High Performance Axial Compressor for Utility Gas Turbine

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ABSTRACT

An aerodynamic design study to configure a high efficiency industrial-size gas turbine compressor is presented. This study was conducted using an advanced aircraft engine compressor design system. Starting with an initial configuration based on conventional design practice, compressor design parameters were progressively optimized. To further improve the efficiency potential of this design, several advanced design concepts (such as stator end-end bends and velocity controlled airfoils) were introduced. The projected polytropic efficiency of the final advanced concept compressor design having 19 axial stages was estimated at 92.8 percent, which is 2 to 3 percent higher than the current high efficiency aircraft turbine engine compressors.

The influence of variable geometry on the flow and efficiency (at design speed) was also investigated. Operation at 77 percent design flow with inlet guide vanes and front five variable stators is predicted to increase the compressor efficiency by 6 points as compared to conventional designs having only the inlet guide vane as variable geometry.

NOMENCLATURE

- Δ - Change in parameter
- δ - Inlet pressure/standard day pressure
- δ = P_T/14.696
- η - Efficiency, %
- θ - Inlet total temperature/standard day temperature, θ = T_T/518.7
- H - Boundary layer momentum thickness, in.
- σ - Airfoil solidity, σ = a/S
- w - Total pressure loss coefficient

I INTRODUCTION

There have been great improvements in axial compressor design techniques for compressors used in aircraft gas turbine engines. In addition to increasing the efficiency, these improvements have been aimed at increasing the specific flow, the surge margin, and the pressure ratio per stage. This has allowed development of compact, light weight high performance compressors and engines that are of primary importance for flight economics.

The purpose of the design presented in this paper was to apply the knowledge and design techniques developed for aircraft engine axial flow compressors to the compressor for a large utility turbine engine. With weight and number of stages being of secondary importance, the goal was to develop an aerodynamic

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design with a level of efficiency potentially higher than that of the large utility turbine engine compressors that are now in operation.

This paper presents the results of an investigation to develop a 14:1 pressure ratio, 800 lb/sec utility compressor aerodynamic design.

The design study for this Electric Power Research Institute (EPRI) compressor was conducted in four phases. The first phase was concerned with a detailed parametric study to evaluate, with the objective of maximizing overall compressor efficiency, the influence of key aerodynamic and geometric design variables on compressor performance. Phase II refined the Phase I design to consider (1) selection of airfoil profiles, (2) off-design performance, and (3) blade and vane dynamics. Phases I and II employed conventional compressor design techniques in that circular arc and series airfoil profiles were incorporated in the design. The basic design tool used was based on the radial equilibrium with in-built semi-empirical loss-loading correlations. The Phase II design should only be viewed as an optimized detailed aerodynamic design.

Phase III examined the application of advanced blading concepts that had the potential for increasing design point efficiency. Two and three-dimensional (3-D) end-wall and airfoil surface boundary layer analysis codes, along with secondary flow and blade tip clearance analysis models, were employed to develop the advanced concept airfoil design loss models. These advanced concept airfoils were then integrated into the Phase II design to produce the final advanced concept compressor.

The final phase (Phase IV) of this study was directed at investigating the influence of variable geometry on the compressor efficiency during the part power operation of the utility turbine.

II AXIAL COMPRESSOR DESIGN SYSTEM

Figure 2-1 shows the overall logic of the axial compressor design system used for the design study reported in this paper. Also shown in this figure are the major computer modules used for EPRI compressor design. Several other modules which are specifically used for highly loaded transonic compressor stages (for example, three-dimensional viscous codes) are not included here.

The basic design philosophy centers around iteratively updating design parameters (such as, work distribution) and airfoil shape until a "minimum loss" design is established. In this iterative optimization for minimum loss (or maximum efficiency), it is ensured that the other design requirements such as surge margin and structural integrity are not sacrifices.

The first major step in this iterative design process is the solution of the meridional (axisymmetric) flow in conjunction with blade generation process. This provides the designer with a reasonable approximation of the flowpath, number of stages, airfoil solidity, aspect ratio, and also an initial guess of the blade shape. Built into this core compressor design module (CCDM) are semi-empirical correlations for airfoil profile and shock losses, deviation, and tip clearance effects with associated mixing losses. This design is now refined by conducting two-dimensional blade-to-blade flow analysis (BBFA) and more accurate estimate of the end wall boundary layer secondary flow losses (SECOND). Before a design is finalized, three-dimensional inviscid (or viscous, if highly loaded) analyses is carried out to evaluate three-dimensional effects that cannot be predicted by the above quasi-three-dimensional system.

Further relevant details of the design process and computer modules are included in appropriate sections of this paper.

III PHASE I - PRELIMINARY DESIGN CONFIGURATION

The first phase of the program was to define a conceptual design configuration for the axial flow compressor with the following design point conditions:

1. corrected flow, \( \dot{W}/\dot{V} \) 800 lb/sec
2. overall pressure ratio, \( P_1/P_2 \) 14.0:1
3. corrected speed, \( N/\dot{V} \) 3600 rpm

This conceptual design was configured by conducting a detailed aerodynamic and geometric parametric study where key design parameters were varied to evaluate their influence on compressor performance with the design objective of maximizing compressor efficiency.

Parametric Study. The parametric study was divided into two steps, an initial screening phase followed by fine tuning of each design parameter.
During the initial screening phase, the range of each design parameter was based on the conventional aircraft engine compressor design practices. Table 1 gives various design parameters and their ranges that were investigated in this phase.

<table>
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<tr>
<th>TABLE 1</th>
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<tr>
<td><strong>Flow path shape</strong></td>
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<tr>
<td><strong>Number of stages</strong></td>
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<tr>
<td><strong>Aspect ratio</strong></td>
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<tr>
<td><strong>Solidity</strong></td>
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<td><strong>Inlet specific flow</strong></td>
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<tr>
<td><strong>Stage reaction</strong></td>
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<td><strong>Exit Mach No.</strong></td>
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<td><strong>Inlet hub/tip radius ratio</strong></td>
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The general approach to the initial phase of the parameter study was to develop a series of designs for constant hub, mean, and tip flow path shapes with the number of stages varying from 19 to 8 for each flow path shape. If during the study, airfoil diffusion factor, \( D_f \), exceeded 0.5 then stage pressure ratio distributions were varied to limit \( D_f < 0.5 \). Because of the low wheel speed (3600 rpm) several instances occurred where the stage loadings were excessive and could not be corrected by alterations in stage axial pressure ratio distribution. The condition was particularly evident for the constant hub designs.

Figure 3-1 illustrates the effect of number of stages and inlet specific flow for the constant hub, constant mean, and constant tip configurations on the overall adiabatic efficiency of the compressor. Rotor 1 inlet hub/tip for this study was 0.5.

The adiabatic efficiency potential of this configuration is computed to be 87.0 percent (polytropic efficiency = 90.8%) which is of the same level as the as GE E-3 compressor.

In the next phase of the parameter study, a more detailed optimization was carried out with design variables extended well beyond the conventional levels used in the aircraft engine compressors. One such design variable was the tip solidity which was progressively reduced below the conventional aircraft compressor design practices. To counter the effect of increased diffusion factor associated with reduced solidity, the stage number was increased thereby reducing the average stage pressure ratio. Particular attention was also paid to the stage-wise and span-wise distributions of work, diffusion factor, and swirl.

Results of this fine tuning indicated that the maximum efficiency potential is offered by a 19-stage configuration having a relatively high specific flow in the front stage and very low airfoil solidity. Increasing the number of stages beyond 19, shows a sudden drop in efficiency (see Figure 3-2).

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Preliminary Design Configuration. Based on the results of the detailed parametric study, the compressor configuration that produced maximum efficiency incorporated the following features:

- 19 stages plus IGV
- 40 lb/sec-ft\(^2\) inlet specific flow
- 0.45 inlet hub/tip radius ratio
- constant mean section flow path
- swirl in all stages except the last stator
- very low solidity

It was predicted that the proposed compressor design would have the following design point performance:

- compressor exit Mach number, \( M_e \) 0.3
- overall adiabatic efficiency, \( \eta_{ad} \) 88.8%
- overall polytropic efficiency, \( \eta_{poly} \) 92.1%

As seen from Figure 3-1, the initial screening study favors a 15-stage, constant meanline configuration with an inlet specific flow of 40 lb/sq ft-sec.
Figure 3-3 presents the flow path of the optimized preliminary design configuration. Important design variables for this configuration are presented in Figures 3-4 through 3-7. Also shown in these figures are the corresponding design parameters of contemporary aircraft engine compressors. The compressor selected for this comparison are the T56-A427 and five other unnamed compressors. This comparison highlights the similarities and differences of the EPRI design with these existing compressors.

Figure 3-3 19 Stage Preliminary Design Compressor

Figure 3-4 Comparison of EPRI Proposed Design Rotor Inlet Specific Flow Distribution with Other Contemporary Designs

Figure 3-5 Comparison of EPRI Proposed Design Stage Pressure Ratio Distribution with Other Contemporary Designs

Figure 3-6 Comparison of EPRI Proposed Design Rotor Solidity with Other Contemporary Designs

Figure 3-7 Comparison of EPRI Proposed Design Diffusion Factors with Other Contemporary Designs

IV PHASE II - AIRFOIL GEOMETRY

Phase I of this study optimized global design parameters, using conventional design techniques, to produce a preliminary design. In Phase II, preliminary geometry of rotor and stator airfoils was defined. This involved selection of incidence, chord, maximum thickness/chord, and interblade row gaps. The criteria was to select airfoil geometric parameters that offer the best compromise between the design and off-design performance without compromising the structural integrity of the airfoil. Some adjustment of solidity and swirl distributions were also carried out to improve stage-wise and span-wise distributions.
of airfoil loading. No attempt was made at this point to customize the airfoil shape. All the airfoils had circular arc meanline with the exception of the first rotor multiple circular arc (MCA) configuration.

Incidence. In an effort to maximize the performance at the design point conditions by keeping the airfoil diffusion requirements reasonable and still to limit the swing in incidence angles from design point to a near-stall operating point (20% surge margin), near-zero incidence angles were selected for all the rotors. For stators, incidence angles were iteratively selected in the forward portion of the compressor; in the aft stages, the stator incidence angles were set at negative values. The negative incidence angles were limited to the aft stages, since in the multistage environment, it is principally the aft stages of the compressor that react to the increases in loading and incidence angle change as the compressor moves from design point operation to stall.

The design process began with the basic airfoil geometry of the preliminary design configuration whose flow path was shown in Figure 3-3. Incidence angles were then selected based on the above criteria, and the resulting airfoils were configured for design operating point using a deviation rule based on the NASA SP36 (Reference 1) cascade correlation with corrections for endwall effects.

A detailed blade-to-blade flow analysis was also conducted on selected rotor and stator airfoils, with the resulting surface velocity distributions examined for possible flow separation and unacceptably large values of the equivalent diffusion parameter, $D_s$ (suction surface $V_{max}/V_c$, which is a measure of suction surface diffusion).

The criteria used during this phase of study for the prevention of incipient flow separation was to keep the value of the surface incompressible boundary layer shape factor $H_s \leq 3.0$ at the airfoil trailing edge. It is recognized that the value of $H_s = 3.0$ is somewhat higher than the more generally accepted value of $H_s > 2.2$ for incipient separation. It was anticipated that $H_s$ could be substantially reduced later in the study when the advanced concept customized airfoil profiles were employed in the design process.

Once the design point airfoil flow conditions were considered acceptable, an off-design model of the compressor was developed, and the above computations were repeated at the predicted 20 percent surge margin operating point. The increases in incidence angles, $\Delta \alpha$, and potential flow separation were examined at the 20 percent surge margin point. If the resulting design/off-design point results were deemed unsatisfactory, the entire process was repeated with another assumed axial/radial incidence angle distribution. This design process led to the incidence angle distributions illustrated in Figures 4-1 and 4-2 for the rotors and stators, respectively. It can be seen that the incorporation of negative incidence angles at the design point limited the swing in incidence from about $-1$ to $+6$ deg for the rotors and $-5$ to $+6$ deg for the stators for the projected range of operation at design speed. Figure 4-3 presents the predicted design speed pressure-flow characteristic associated with the airfoil geometry resulting from the incidence angle distributions of Figures 4-1 and 4-2.

**Figure 4-1** Variation of the Rotor Incidence Angle from Design to 20 Percent Surge Margin Operating Conditions

**Figure 4-2** Variation of the Stator Incidence Angle from Design to 20 Percent Surge Margin Operating Conditions

**Figure 4-3** Flow Characteristics at 100 Percent Design Speed Pressure

Figures 4-4 and 4-5 show the stage wise radial distribution of $D_s$ for the rotors and stators, respectively. Forward-stage rotor and aft-stage stator values are aggressive, but, as stated earlier, it is anticipated that these levels of high diffusion can be accommodated by customizing the airfoil contours. Also shown in Figures 4-4 and 4-5 are the projected increased levels of $D_s$ at the 20 percent surge margin operating point. The improved diffusion capability of the customized airfoil contours will be beneficial in attaining these off-design $D_s$ levels and providing the compressor with the capability to achieve 20 percent surge margin.

**Dynamic Analysis.** To ensure that the detailed design configuration was structurally sound, a dynamic analysis (using the state-of-the-art finite element code) was performed on blade rows throughout the compressor. The objective was to identify maximum...
airfoil aspect ratio that is acceptable from the point of view of structural integrity as well as the passage diffusion capacity. The criteria for dynamic acceptability was that all rotors first bending-mode natural frequencies be above second engine order. For stators, the criteria was that the first bending frequency not be near the first engine order. The criteria for the limiting passage diffusion capacity was similar to that presented in Koch's paper (Ref. 2).

Figure 4-6 Detailed Design Configuration Flow Path

Comparing the compressor flow paths of the preliminary design configuration of Figure 3-3 with that of Figure 4-6 shows that a substantial reduction in overall compressor length has been effected in the detailed design configuration. Stage-wise distributions of key design aerodynamic and geometric parameters for the detailed design configuration are illustrated in Figures 4-7 through 4-9.

Figure 4-4 Variation of the Rotor Equivalent Diffusion Factor from Design to 20% Surge Margin Operating Conditions

Figure 4-5 Variation of the Stator Equivalent Diffusion Factor from Design to 20% Surge Margin Operating Conditions

This part of the study helped to define airfoil maximum thickness/chord ratios and the minimum acceptable length of the compressor. Minimizing the length reduces endwall skin friction drag, shafting dynamics problem, and probably, overall compressor cost. Details of this study are presented in Reference 3.

Detailed Design Configuration. Relative to the preliminary design configuration reviewed in Section III, all of the previously mentioned design refinements of airfoil incidence, thickness/chord ratio, solidity, aspect ratio, and axial gaps along with minor changes in airfoil swirl distributions, produced the detailed design configuration presented in Figure 4-6.

Figure 4-7 Detailed Design Configuration Stagewise Distribution of Rotor and Stator Aspect Ratio

Figure 4-8 Detailed Design Configuration Stagewise Distribution of Diffusion Factors

Figure 4-9 Detailed Design Configuration Stagewise Distribution of Solidity
The detailed design configuration can be categorized as being a conventionally designed compressor with relatively high specific flow, subsonic, high aspect ratio, fairly lightly loaded, very low solidity design features. Results of these design studies show that the optimum values of design parameters such as stage loading, hub/tip radius ratio, and swirl are similar to industrial compressors in service today. The predicted design point performance for the detailed design configuration is:

- overall adiabatic efficiency 88.8%
- overall polytropic efficiency 92.1%
- surge margin, SM 20%

This projected performance level is considered to be that which would result from experimental development to fine-tune stage matching, high quality airfoils, very good surface finishes, and a blade running tip clearance of 0.02 inches.

V PHASE III - ADVANCED DESIGN CONCEPTS

The performance results described in the previous sections have shown that if any significant improvements in efficiency are going to be realized, they are going to be the result of integration of advanced airfoil design concepts into the aerodynamic design process. In any compressor blade row design, the principal contributors in total pressure loss are the following:

- Cascade Profile Loss--due to viscous effects on airfoil surfaces
- Endwall Loss--due to viscous three-dimensional boundary layer flows resulting in pressure loss on the endwalls, formation of energy dissipating secondary corner vortices, and high incidence on airfoil leading edges submerged within the wall boundary layer
- Tip Clearance Loss--losses associated with the blade tip clearance leakage flow along with the casing boundary layer fluid contained and dissipated within the tip leakage vortex core
- Shock Loss--total pressure losses associated with deceleration through a shock positioned in front of or within airfoil passage

The detailed design configuration is essentially a subsonic compressor design; consequently, the impact of shock loss on overall compressor efficiency was not a major consideration. On the other hand, the means to control or reduce profile, endwall, secondary, and tip clearance losses are a major concern if the efficiency level predicted for the EPRI detailed design configuration is to be improved.

Although several advanced airfoil concepts were reviewed (see Reference 3), only the following three concepts were considered potentially beneficial for reducing aerodynamic losses:

- Customized or Velocity controlled Airfoils--These airfoil shapes minimize suction surface diffusion loss and maximize the static pressure rise capability of an airfoil.
- Stator End Bends--This concept has shown promise by unloading the stator endwall region and by improving the incidence on the portion of the airfoil submerged in the wall boundary layer.
- Swept-Forward Leading Edge Stators--This concept reduces endwall region loss by inducing a migration of high-momentum fluid to the stator endwall region.

The performance potential of velocity-controlled airfoils (VCA) and stator end bends was estimated with the aid of Allison's advanced design tools. A detailed description of the design methodology and analytical procedures adopted for incorporating the VCA and stator end bends into the detailed design configuration is presented in the following text.

Velocity Controlled Airfoils. The VCA concept, which aims at optimizing the streamwise distribution of suction surface diffusion, was applied to the airfoil sections not subjected to strong three-dimensional effects (i.e., away from the endwall regions).

This concept could be extended to the endwall region but that would require a fully three-dimensional viscous analysis. To generate VCA sections, the customized blade generation program (CBG) was combined with two-dimensional blade-to-blade flow analysis code (BBFA) and an integral-momentum boundary layer analysis. The BBFA is a fast analysis capable of handling variation in streamtube height, boundary layer blockage, loss, and local supersonic bubbles. The scheme, based on the streamline curvature technique, has been validated against cascade data and other computational schemes and is extensively used to design axial compressor airfoils. The boundary layer analysis, which is merged with BBFA, is based on the well-known Head's momentum-integral technique (Ref 4).

The design methodology adopted for the present study involved iteratively updating the airfoil shape to minimize the boundary layer skin friction loss while ensuring that the suction surface boundary layer does not separate. The following design criteria were used:

- The trailing edge shape factor, H, is less than or equal to 2.2. This is to ensure that the boundary layer remains attached to the airfoil surface.
- The computed exit flow angle for the customized airfoil remained equal to that of the baseline circular arc meanline airfoil. This design requirement ensured that the inlet swirl to the downstream rotor would remain unchanged.

Figure 5-1 shows a diagram of the overall logic employed to arrive at the optimized VCA sections.

Figure 5-2 shows, for stator 6, the impact of (1) customizing the midspan profile to optimize diffusion and (2) the influence of chord (solidity) reduction on key boundary layer and aerodynamic parameters relative to the baseline circular arc midspan sections. These results illustrate that a 20 percent reduction in chord is possible for the customized airfoil and produces a substantial decrease in trailing edge shape factor and momentum thickness relative to the baseline midspan section. In other words, customizing has produced a lower solidity, slightly more heavily loaded airfoil that is less susceptible to flow separation and should perform more efficiently than a baseline midspan section.
Customizing moves the loading forward on the airfoil, thus effecting a more gradual diffusion to the trailing edge with attendant reduction in skin friction loss.

Midspan DCA and the 20 percent reduced chord customized sections are illustrated in Figure 5-4.

The reduction in skin friction or profile loss resulting from the use of velocity controlled airfoils was estimated by computing the suction and pressure surface boundary layer characteristics at the stator trailing edge. The expression, developed in Reference 5 for this purpose, is as follows:

\[
\Delta \mu = \left( \frac{\nu}{\nu} \right) \left( \frac{1 + \frac{1}{2} \frac{V}{c} \cos\left(\frac{\theta}{c}\right)}{1 + \frac{1}{2} \frac{V}{c} \cos\left(\frac{\theta}{c}\right)} \right) - \left( \frac{1 + \frac{1}{2} \frac{V}{c} \cos\left(\frac{\theta}{c}\right)}{1 + \frac{1}{2} \frac{V}{c} \cos\left(\frac{\theta}{c}\right)} \right)
\]

Figure 5-5 Influence of Midspan Chord Reduction of Customized Airfoils on Midspan Loss Reduction

The foregoing loss model demonstrated that a 0.0035 reduction in stator midspan loss coefficient could be realized by the implementation of reduced midspan chord customized stator airfoils. This design concept and its associated reduction in loss was applied to all stator rows.

Stator End Bends. The stator end bend configurations were evolved from an iterative computation that included a new secondary flow loss model formulation.
This subsection contains highlights of this loss model, followed by a brief description of the procedure that optimized the stator reset angle geometry.

The method adopted for computing the change in endwall losses and deviation angle resulting from the incorporation of stator end bends was based on a secondary flow analysis (SECOND) similar to the classical secondary flow theory (Reference 6). However, unlike Reference 6, the analysis employed here takes into account the viscous flow effects on the development of secondary vorticity through the blade row.

An implicit finite difference scheme is used to integrate the secondary vorticity equation along an inviscid mid passage streamline passing through the edge of the boundary layer. The integration is carried out twice, once for the hub region and once for the casing region. The numerical procedure for the secondary flow computation needs the following input:

- Inlet boundary layer profile—This is computed with the aid of Allison's three-dimensional endwall boundary layer code (shroud) described in Reference 7.
- Inviscid blade passage velocity distribution at the edge of the boundary layer—BBFA was used for this computation.
- Spanwise distribution of exit flow angle—This is available from the meridional plane flow analysis (MFA).

The secondary vorticity distribution computed in the exit plane of the stator was determined by solving the Poisson equation for the secondary flow stream function as formulated in Reference 8. By integrating secondary velocity in the circumferential direction, deviation angles associated with endwall boundary layer secondary flow can be evaluated.

To compute the total pressure loss associated with the secondary flow, it was assumed that all of the secondary flow kinetic energy is dissipated and converted into total pressure loss. This heuristic assumption needs some refinement and validation for computing the secondary flow loss in a cascade; however, it is considered to provide a reasonable approximation for estimating the change in secondary flow loss between the geometries. Based on this assumption, the reduction in total pressure loss coefficient, due to the influence of reset closed stator end bends, was evaluated.

Results of the loss reduction computation for a representative stator (stator 10) is shown in Figure 5-6. Also, for all of the geometries evaluated, the change in deviation angle was of the order of one degree. The loss computation shows a fairly large reduction in the endwall region, with levels depending on the amount of reset and where the subject stator was positioned in the compressor. Also, the region located a small distance from the hub and tip wall experienced a sizable reduction in loss coefficient.

Loss contours of the type shown in Figure 5-6 were developed for all 19-stage stators and used in the iterative process that produced the final stator reset geometry for all stages. The following stepwise procedure was used to optimize the reset angles and define the end bend geometry to be factored into the detailed design configuration:

1. With the aid of the three-dimensional endwall boundary layer code (SHROUD), stator inlet flow angle was calculated inside the endwall boundary layer.
2. The endwall sections (hub and tip) of each stator were reset to ensure that the incidence in the endwall boundary layer is less than or equal to zero.
3. The secondary flow analysis code (SECOND) was then used to estimate the endwall secondary flow including losses and flow angles for reset and baseline blades.
4. The computed endwall region losses and exit flow angles were then input into the meridional flow analysis (MFA), and the program was run in the predict performance mode. Reduced losses for the customized stator midspan sections were also included in this calculation; thus, the results of this analysis gave the total pressure ratio and efficiency of the complete advanced concept design.
5. If the pressure ratio of advanced design was different from the design intent (14.0:1), the resets were adjusted, and steps (1) through (4) were repeated until the pressure ratio matched the design intent.

A three-dimensional view of the advanced concept compressor design final stator-6 geometry, including end bends in addition to the reduced chord velocity-controlled midspan region, is compared with the baseline DCA airfoil in Figure 5-7.

Incorporation of the airfoil reset geometry and the respective reduced loss coefficient profiles for the stator end bends and reduced chord customized midspans produced a computed overall advanced concept compressor design adiabatic efficiency of 89.3 percent.

Although no separate investigation was carried out to customize rotor airfoils in the midspan and endwall region, it is expected that incorporation of customized rotor airfoils will produce higher levels of efficiency improvements than that estimated for customized stator airfoils. This is because, for the same change in the boundary layer momentum thickness, the change in the total pressure loss is larger in a rotor than in a stator, due to the higher inlet dynamic head and exit flow angle. Based on this
argument, the overall adiabatic efficiency of the advanced concept compressor design is projected to be at least one percent higher than the baseline design. This translates to an adiabatic efficiency of 89.8 percent or a polytropic efficiency of 92.8 percent.

![Diagram](image1)

**Figure 5-7** Impact of Incorporating Reduced Chord Customized Midspan Sections with Hub and Tip End Bends on Stator-6 Baseline DCA Airfoils—Advanced Concept Compressor Design Final Stator-6 Design

Final Advanced Concept Compressor Design. The design point projected performance of the final advanced concept compressor design is as follows:

- corrected inlet flow, \( \dot{W}_i \), 800 lb/sec
- pressure ratio, \( R_{c} \), 140:1
- corrected wheel speed, \( N/\sqrt{V} \), 3600 rpm
- surge margin, SM, 20%
- adiabatic efficiency, \( \eta_{TT} \), 89.8%
- polytropic efficiency, \( \eta_{poly} \), 92.8%

![Diagram](image2)

**Figure 5-8** Advanced Concept Compressor Design Configuration

A performance map was developed for the advanced concept compressor design. The map, presented in Figure 5-9, illustrates the impact of speed variation on overall performance. The off-design characteristics were estimated with the aid of extensive in-house data base on multistage, fixed-geometry, high-pressure ratio compressors. The EPRI compressor design presented in this paper would be more suitable for a variable-geometry design. For a fixed-geometry situation, it would have been more desirable to unload the front stages of the compressor to allow good surge margin at lower corrected speed.

![Diagram](image3)

**Figure 5-9** Fixed Geometry Performance Map for EPRI Advanced Concept Compressor Design

VI PHASE IV - OPTIMIZATION OF PART POWER OPERATION WITH VARIABLE GEOMETRY

A variable inlet guide vane (IGV) is commonly used in utility gas turbine compressors in order to 1) improve starting flexibility and 2) improve the heat rate at part-power in combined cycle. In aircraft gas turbines, it is common to have a number of variable stator stages (in addition to the IGV) to further improve compressor operability at part-power and under changing flight conditions. In a utility gas turbine compressor, the power is reduced at the design rpm (3600) by reducing the inlet flow and pressure ratio through the closure of the IGV and stator vanes.

In order to find the possible improvement in part-flow efficiency, the following three configurations of variable geometry were considered in this study:

1. Variable IGV only.
2. Variable IGV and stators 1 through 5 in a ganged mode (all vanes reset same number of degrees).
3. Variable IGV and stators 1 through 5 in a graduated mode (in this configuration, for each degree rest of IGV, stators 1 through 5 were reset 5/6, 4/6, 3/6, 2/6, and 1/6 deg. respectively.)

To do this study, the IGV reset was varied from 5 deg to 30 deg. The operating point of each reset was on a constant \( \dot{W}_i/V \) line (which corresponds to constant turbine inlet temperature) prescribed by the design point value. An iterative procedure using the meridional flow analysis code (MFA) was adopted for estimating the mass flow and efficiency corresponding to each reset. Extra losses were added to account for separation due to high incidence; at the same time,
the flow was dropped to ensure that the compressor was operating in an unchoked mode.

The primary results of this analysis are presented in Figure 6-1, which compares the efficiency decrement as a function of flow for the above three variable geometry configurations. It may be seen that either of the two variable stator configurations is highly superior to the IGV-only configuration. At 75 percent flow, there is approximately one-third as much decrement to efficiency as for the IGV-only, while the flow can be reduced to about 60 percent before the efficiency decrement is the same as that for the IGV-only at 75 percent.

Figure 6-1 Efficiency Decrement versus Flow Reduction for Three Variable-Geometry Configuration

VII CONCLUSIONS

Based on the results of this three-phase design study effort to develop an aerodynamic design for a large utility gas turbine engine compressor, the following conclusions can be drawn:

1. Using conventional design techniques, a preliminary compressor design was developed and refined to produce a polytropic efficiency of 92.1 percent. This level of efficiency potential was achieved by having a large number of lightly loaded stages with low solidity airfoils.

2. Several advanced design concepts were investigated, and stator midspan velocity-controlled airfoils, stator end bends, and stator leading edge sweep were shown to hold promise for increasing overall compressor efficiency.

3. Analytical studies on the velocity-controlled stator midspan sections showed that the customized airfoils would operate with less loss and be less susceptible to flow separation, even with a 20 percent reduction in solidity, than a baseline DCA airfoil.

4. Incorporation of geometries and losses associated with stator end bends and stator customized midspan sections resulted in a 0.5 percent improvement in adiabatic efficiency. An additional improvement of 0.5 percent in adiabatic efficiency is expected to result from the customizing of rotor airfoils.

5. Estimated performance of the 800 lb/sec, 14.0:1 pressure ratio EPRI advanced concept compressor design was:
   - surge margin, SM >20%
   - adiabatic efficiency, $\eta_{TT}$ 89.8%
   - polytropic efficiency, $\eta_{poly}$ 92.8%

6. Incorporation of variable geometry in the IGV and the first five stators of the advanced concept compressor design demonstrated that the compressor flow could be reduced by 25 percent with a 4 percent efficiency penalty. The corresponding efficiency penalty for the variable IGV-only configuration was estimated to be in excess of 10.7 percent.

REFERENCES


