Master

High-Reliability Gas Turbine Combined-Cycle Development Program
Advanced Compressor Design Study

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ABSTRACT

This report presents results of one task of the first phase of RP1187-2, High-Reliability Gas Turbine Combined-Cycle Development Program. The purpose of this task is to identify and quantify the potential benefits of applying advanced-design, low-aspect ratio, high-throughput compressor technology to heavy-duty power generation combustion turbine engines. Such technology is being used in the aircraft industry with satisfactory performance. This task was undertaken in view of this experience and with the expectation that the adaptation of this technology to electric utility combustion turbine power plants could offer substantial improvements in plant reliability, availability, and maintainability in comparison with units now in service. Included in this task is the development of conceptual designs and performance parameters based on the advanced compressor design approach to provide input to the selection of the best compressor configuration to meet the reliability and efficiency goals set forth for the new engine and combined cycle.

The work has progressed with the execution of a literature survey of advanced aircraft technology, low-aspect ratio designs, and the conceptual selection of aerodynamic stage parameters for first, middle, and last stages suitable for the 14:1 compressor pressure ratio requirement. Subsequent analyses identified key mechanical design requirements for blading and rotating elements of the design.

EPRI PERSPECTIVE

PROJECT DESCRIPTION

This final report of Phase I, Task 9 of RP1187-2, High-Reliability Gas Turbine Combined-Cycle Development, covers the conceptual work on an advanced compressor that uses low-aspect blading and other technology which has been developed in advanced aircraft compressor projects. The task consisted of a literature survey, primarily of advanced aircraft projects, followed by conceptual design work that resulted in the selection of stage parameters for the best compressor configuration to meet reliability and performance requirements for a high-reliability industrial combustion turbine. This task is closely coordinated with projects under the Department of Energy (DOE) program of better compressors for industrial engines (Contract No. DE-AD03-79ET-15332).

PROJECT OBJECTIVE

The major objective of Task 9 was to determine the benefits of using the technology developed in advanced aircraft compressor projects for the design of a low-aspect ratio blading compressor to reduce the number of parts while still maintaining a high efficiency. This work is intended to lead into the more-detailed design being investigated by DOE for contribution to the high-reliability combined-cycle concept.

RESULTS

The results of the Westinghouse study indicated that a utility combustion turbine could be obtained with a high-performance compressor with fewer stages and considerably fewer airfoils by adapting the low-aspect ratio blading technology developed in advanced aircraft engine projects. This type of compressor would contribute to the reliability of a new centerline-design combustion turbine. The reliability improvement is obtained both by the fewer number of parts and the more-rugged construction of the wider chords, which provide greater resistance to erosion and foreign-object damage. It is expected that the fewer parts will also result in a lower cost for the compressor. Further, the compressor efficiency is expected to be maintained at the previous high levels with no sacrifice in performance.

This Task 9 is only a conceptual study to indicate the potential of the low-aspect ratio technology. Additional efforts on this concept will be carried on (under DOE contract DE-AD03-79ET-15332) in order to obtain more-detailed design data. For further information, the reader is referred to the reports on that contract when they become available.

R. L. Duncan, Project Manager
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ACKNOWLEDGMENTS

The following people, in alphabetical order, provided vital contributions to the preparation of the work described by this report:

- Mr. S. F. Arlington performed the mechanical analysis of the conceptual design blading.

- Mr. C. A. Rohr performed the survey and evaluation of literature, conceptual aerodynamic design selection, and analysis of detail stage characteristics.

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SUMMARY AND CONCLUSIONS

This preliminary study was conducted under Task 9 of EPRI RP1187-2 to define a conceptual design of an alternate compressor for the high-reliability combustion turbine engine, based on the latest advances made in aeroengine compressor technology.

The design study resulted in the conceptual definition of a 13-stage compressor design providing a pressure ratio of 14:1 with an inlet airflow of 800 lbs/sec (360 kg/sec). This compressor concept, using low-aspect ratio blading, will be analyzed in further detail in a follow-on program funded by DOE under Contract No. DE-AC03-79E1-5332. This compressor will also be evaluated against a more aggressive approach with 11 stages and a more conservative design consisting of 15 stages as well as against the present W501 engine compressor. Finally, depending upon the results of their evaluation, a compressor configuration will be selected as a candidate for further study and possible development.

The task itself consisted of two major subtasks. The first was a data search of various experimental advanced aircraft compressors and an evaluation of the technology to determine the applicability of certain technological advancements which may be introduced in heavy industrial combustion turbine engines. This review also included a projection of the 1985 technology. The second part of the study covered the preliminary definition of an advanced industrial compressor, based on the latest aircraft engine compressor technology and advances projected to be available in 1985.

This study indicates that:

- A high-performance utility engine compressor with fewer stages, featuring low-aspect ratio blading based on aeroengine advanced compressor technology, can be developed to meet the high-reliability combustion turbine engine performance goals and reliability requirements.
- Low-aspect ratio compressor blading should provide larger flow range and, thus, greater stall margin and increased surge margin.

- Higher blade-tip speed is required to maintain an acceptable level of stage-work loading and, therefore, inlet and exit hub/tip ratio must be increased from the present values for the W501 compressor.

- Lower-aspect ratio blades, with their wider chords, may provide improved blade-erosion life, possibly resulting in engine performance deterioration.

- The reduced number of blades and vanes resulting from the fewer stages and low-aspect ratio blading should result in a lower compressor cost, in spite of the added complexity due to increased variable geometry. It is believed that reducing the number of stages and blades should also result in significant improvement of the compressor reliability.

- The higher hub/tip ratio may be beneficial from an overall engine design viewpoint by reducing the rotor-thrust balance problem associated with greater diameter of the three-stage turbine.

Section 1
INTRODUCTION

This report discusses the work carried out by the Westinghouse design team under Task 9 of Phase I of the RP1187-2 High Reliability Gas Turbine Combined Cycle Development Program. This particular task covers the conceptual definition of an alternate advanced compressor with a reduced number of stages using low aspect ratio blading and technology levels based on advanced design aircraft engine compressors.

Cycle Requirements

The High Reliability Combustion Turbine should have a low heat rate, be easy to maintain and should be reliable. Each of these desired characteristics is related to the performance and design of the engine major turbomachinery components. Design studies conducted under Phase I indicate that a minimum number of components, i.e., number of compressor stages with a corresponding reduction of the number of blades and vanes, would significantly contribute to the achievement of the reliability, availability and maintainability (RAM) goals for the combustion turbine.

The compressor performance requirements were established by an earlier study covered in Task 2 which showed that an overall engine cycle pressure ratio of 14:1 with an airflow of 800 lbs/sec (360 kg/sec) and an efficiency level reflecting 1985 technology would best meet the performance and heat rate goals as set forth by EPRI for the High Reliability Combustion Turbine. From a mechanical viewpoint, the life goal for the compressor was set at 100,000 hours. These goals were, therefore, used in the conceptual design of the compressor study described in the following sections.

Scope

This conceptual design study was divided into two subtasks. The initial subtask (9-1) of the study covered a survey of literature and available experimental test data and a consideration of basic compressor design parameters as well as a projection of the technology level attainable for industrial machines by 1985.

This survey was concentrated primarily on the results and work done in advanced aircraft engine compressor research and development, since this is the area in which the most promising results have been obtained to date and also where the most effort is being carried out today.

The following subtask (9-2) involved the definition of a conceptual design of a typical advanced industrial compressor based on these projected technological levels.

This basic configuration will be analyzed further and used as a reference design in the Comparative Evaluation and Trade-off Studies and Selection of Compressor for the New Centerline Engine to be covered under the DOE funded follow-on program, "Compressor Configuration and Design Optimization for the High Reliability Gas Turbine," Contract NO. DE-AC03-79ET15332.

Section 2

DATA SEARCH AND EVALUATION
(Subtask 9.1)

2.1 WORK STATEMENT

This task entails an in-depth data search for available information on low aspect ratio high through-put compressors. This investigation will include compiling data from such sources as the Department of Defense (Wright-Patterson Air Force Base), NASA and aircraft engine manufacturers with development experience in this area. A search of the literature on the subject will also be performed.

The data base established as a result of the search will be refined and evaluated to identify useful input to the design, performance and reliability analysis that will be performed under later tasks. For this purpose, an attempt will be made to compile and present the data in such a way as to yield the effects of such design parameters as stage loading, end-wall boundary layer treatment, tip clearance, tip speed, aspect-ratio, and number of blades per stage.

2.2 APPROACH

Current active development of high through-put, low aspect ratio compressors is most evident in the design of high performance, lightweight military aircraft engines where strong emphasis is also placed on reliable operation, although engine lifetimes are not comparable to those of utility/industrial machines.

Therefore the approach adopted for Subtask 9.1 was to contact two responsible agencies, Wright-Patterson AFB and NASA Lewis Research Center, for an update on the current status of advanced low aspect ratio compressor design technology.

As a result, a bibliography of advanced compressor technology literature was compiled. Representative reports from the bibliography were retrieved for subsequent analysis and evaluation as reported in the following section.

2.3 DISCUSSION OF CHARACTERISTICS IDENTIFIED FROM LITERATURE SURVEY

Data has been compiled from six advanced technology axial flow compressors from NASA and aircraft manufacturer sources. The overall referred flow, speed characteristics as well as individual first stage aerodynamic characteristics such as Mach numbers, stage flow and work coefficients, reaction, etc., and blading characteristics are identified in the following discussion.

Each of the pertinent stage aerodynamic design parameters has been reviewed in order to assess its value to the Advanced Compressor Design Study ACDS program and to identify useful direct input to the preliminary conceptual design performed under Task 9.2.

Overall Referred Performance

Overall referred performance of the six survey compressors is tabulated in Table 1. The first characteristic of importance is the referred inlet flow based on the ACDS referred speed of 3600 RPM. It shows that only two compressors have inlet flow near the cycle requirement of 800 lb/sec (360 kg/sec). This limitation has been imposed on the compressor flow selection due to aero/mechanical limitations of the last row turbine blade and not due to compressor aerodynamic design constraints. As a consequence of this flow constraint, the selection of detail aerodynamic design parameters for the first stage of the ACDS compressor will be largely influenced by the first two survey designs, configurations #1 and #2a. In addition, the overall pressure ratios for configurations #1, #2, and #2a of the survey compressors are equal to the ACDS design pressure ratio. These then will provide a good indication of advanced technology average stage loadings and aid in the selection of design parameters for the last stage. The other four surveyed designs were studies or tests of single front stages for much higher referred inlet flows and higher stage pressure ratio. Although not totally relevant to the ACDS program, they did provide further insight to technology levels projected for 1985 application.

Compressor Design Parameters

The compressor design parameters selected as independent variables that would define the compressor stage geometry and performance are:

- Hub-tip ratio
- Mean flow coefficient
- Reaction
- Mach number
- Stage loading
- Solidity
- Aspect ratio

Table 1

KEY CHARACTERISTICS OF COMPRESSORS SURVEYED IN
LITERATURE SEARCH AND RELATIONSHIP TO ACDS FLOW REQUIREMENTS

	Ref. Configuration			ACDS	
Configuration	Referred Inlet Flow $Wa\sqrt{\theta}/\delta$	Referred Speed $N/\sqrt{\theta}$	Overall Design Pressure Ratio	Referred Inlet $Wa\sqrt{\theta}/\delta$	Referred Speed $N/\sqrt{\theta}$
1	88.1	11384.	14.0	881.	3600.
2	68.4	14325.	14.0	1083.	3600.
2a	68.4	12678.	14.0	848.	3600.
3	96.5	13818.	18.0	1422.	3600.
4	104.2	12210.	--	1198.	3600.
5	44.5	17188.	--	1014.	3600.
6	44.5	17188.	--	1014.	3600.

The detail design parameters tabulated on Tables 2 and 3 display the overall compressor characteristics, where they were available, plus first stage rotor and stator data. These data represent the predicted compressor technology levels for 1985 aircraft engines. The emphasis in the survey studies has been primarily on reduction of weight through fewer number of stages, as a result of increased work per stage, increased Mach number levels, and increased tip wheel speeds. These criteria did not, however, generally describe the highest efficiency compressor for the given performance duty. The survey reveals that compressor efficiency estimates generally increase with trends to lower wheel speeds and increased number of stages because of reductions in airfoil losses. Thus these data will only be treated as probable trends for future design technology and not necessarily the best selection for the ACDS conceptual preliminary design.

Many gaps in information exist in the source material of the compressors surveyed. Some data shown has been filled in by estimates, based on typical stage design calculations and matching the available source data. These figures are denoted by an asterisk. The shortage of published data, however, did not impair the objective of obtaining trends of pertinent stage design information.

Evaluating each compressor design variable identified above, the following are ranges selected for further study and use for the ACDS first stage conceptual design:

Inlet hub-tip ratio	0.566 to 0.700
Mean flow coefficient	0.500 to 0.600
Reaction at hub	0.500 to 0.700
Mach Number	
Rotor Tip Relative	1.13 to 1.48
Stator Hub	0.74 to 0.96
Average Stage Pressure Ratio	1.25 to 1.82
Solidity	
Rotor Mean	1.4 to 1.5
Stator Mean	1.1 to 1.4
Blade Aspect Ratio	1.19 to 1.65

Data for the design of intermediate stages and the last stage was very minimal. Selection of design variable ranges would be largely influenced by the first stage. A more detailed evaluation is presented in Subtask 9.2.

Table 2

COMPARISON OF SELECTED FIRST STAGE CHARACTERISTICS OF ADVANCED COMPRESSORS SURVEYED

Configuration	$W_a\sqrt{\theta}/\delta$ lb/sec.	$N/\sqrt{\theta}$ RPM	D_H/D_T	Blades Per Row	Vanes Per Row	Avg. Stage Press. Ratio	Stage Eff. Percent	Number Stages	Compr. Press. Ratio	Compr. Eff. Percent	Shaft Speed RPM
1	88.1	11,384	0.630	--	--	1.302	--	10	14.0	88.2	13,495
2	68.4	14,325	0.566	28	--	1.341	--	9	14.0	86.8	15,983
2a	68.4	12,678	0.566	--	--	1.246	--	12	14.0	86.8	14,145
3	96.5	13,818	0.580	--	--	1.435	--	8	18.0	87.9	16,120
4	104.2	12,210	0.597	24	30	1.805	88.5	1	--	--	--
5	44.52	17,188	0.700 (Inlet)	48	62	1.822	82.2	1	--	--	--
6	44.52	17,188	0.700 (Inlet)	36	46	1.820	82.8	1	--	--	--

Table 3

COMPARISON OF RADIAL VARIATION OF FIRST STAGE PARAMETERS FOR ADVANCED COMPRESSOR SURVEYED

CONFIGURATION	AIRFOIL	RADIUS	DIAMETER IN	# FLOW COEFFICIENT	V WORK COEFFICIENT	REACTION*	DT*	MACH NO* REL.	WHEEL SPEED FT SEC	SOLIDITY	T/C	CHORD IN	ASPECT RATIO MEAN	ROTOR PRESS RATIO	ROTOR EFF
				STAGE QUANTITIES											
1	R	H	16.766	0.855*	1.181*	0.600*	.379*	0.914*	788						
		M	21.690	0.600*					1273						
		T	26.613	0.446*	0.469*	0.685*	.319	1.128*	1568						
	S	H					.427*	0.883*							
		M													
		T					.428*								
[illegible]	R	H	12.995	0.780*	1.367*	0.550*	.464*	0.892*	906*	1.870	.090	2.855			
		M	17.978	0.500*			.400		1254			2.855	1.600		
		T	22.960	0.349*	0.438*	0.745*	.320*	1.22*	1601	1.119	.025	2.855			
	S	H						0.80*							
		M					.420								
		T													
2A	R	H													
		M													
		T													
	S	H													
		M													
		T													
3	R	H	14.684*	0.846*	1.443*	0.500*	.431*	1.012*	1034	2.111					
		M	20.000*	0.550*					1408	1.550			1.52	1.839	
		T	25.316*	0.386*	0.485*	0.710*	.340*	1.307*	1782	1.225					
	S	H					.504*	0.965*		1.392					
		M					-			1.100			1.51		
		T					.519*	–		0.869					
4	R	H	16.585	0.796*	1.681*	0.600*	.599	0.970	883	1.693	.085	3.676			
		M	22.080	0.550*	-		.571	1.168	1178	1.414	.063	4.087	1.30	1.845	.921
		T	26.875	0.409*	0.599*	0.712*	.543	1.317	1419	1.254	.035	4.411			
	S	H	17.940				.586	0.879	-	1.426	.050	2.680			
		M	21.895				.481	0.839	–	1.169	.060	2.680	1.42		
		T	25.560				.552	0.692	-	1.001	.070	2.680	-		
5	R	H					.440	1.13		1.80		1.65		–	
		M	–	–	–	–	.500	1.37		1.48	–	1.65	1.63	1.863	.858
		T					.480	1.48		1.29		1.65	–	–	
	S	H					.410	0.740		1.49		1.12	–		
		M	–				.370	0.690	–	1.38	–	1.19	1.78		–
		T					.420	0.700		1.30		1.25	–		
6	R	H	14.198	0.635*	1.244*	0.700	.460	1.13	1065	1.800		2.200	–	–	
		M	16.994	0.500*	–	–	.480	1.37	1276	1.480	–	2.200	1.190	1.865	0.865
		T	19.789	0.408*	0.609*	0.790	.460	1.48	1485	1.290		2.200	–	–	
	S	H					.340	0.76		1.490		1.510	–		
		M	–				.320	0.73	–	1.380	–	1.600	1.260	–	–
		T					.320	0.73		1.300		1.680	–		

*ESTIMATED

In summary, the findings of the survey show, at best, trends which must be recognized as representing a specific aircraft, optimum duty based on overall aircraft economics. For use in a power generation application specific values of optimum design variables will be selected which primarily would tend to maximize overall compressor efficiency.

2.4 1985 TECHNOLOGY

The conceptual configuration of the alternate advanced axial compressor for the High Reliability Combustion Turbine has been based on the expected level of compressor technology available by 1985.

This technology is essentially based on the present advanced aircraft engine compressor technology and the projected improvements and refinements currently under development.

The technological areas covered in this conceptual design included mainly the compressor aerodynamics, however, attention was also paid to mechanical design considerations, materials and manufacturing processes.

Major conceptual design effort was concentrated on a substantial reduction of the number of stages while trying to maintain the same overall pressure ratio of 14:1 and the same high level of efficiency as currently demonstrated by the W-501 engine compressor.

It is believed that reducing the number of stages and blades should result in significant improvement of the compressor reliability as well as a lower unit cost. To achieve this objective, it is necessary to obtain high pressure ratio per stage with high efficiency. This goal imposes more demanding aerodynamic requirements than current industrial subsonic compressor technology, since higher pressure ratio per stage usually requires higher tip speed as well as higher blade loadings. As blade speed is increased, the airflow relative velocities to the blades tend to become supersonic, resulting in shocks within the blade passages and therefore increased losses.

Therefore to achieve high efficiency it is necessary to develop blade profiles and a flow path that minimizes shock losses. Current advanced aircraft engine compressors have demonstrated very high pressure ratio per stage with good efficiency with tip speeds up to approximately 1500 ft/sec (460 m/sec). Above 1500 ft/sec (460 m/sec) tip speed, the maximum efficiencies that have been obtained are somewhat lower and would not meet the high performance level required for power generation combustion turbines.

In addition, these machines have an inherent very high airflow per unit of annulus area which, when scaled up for a utility combustion turbine running at 3600 RPM, would be excessive due to the last turbine stage blade design restrictions. This results from the well known relationship between blade stress, RPM and annulus area commonly referred to as $\sigma = f(AN)^2$ where

σ = Stress

A = Blade flow annulus area

N = RPM

Based on that relationship, at a fixed RPM, for any given compressor pressure ratio and turbine inlet total temperature, there exists a limiting annulus exit area that corresponds to a blade metal temperature equal to the allowable operation temperature. Corresponding to this limiting annulus area, there is a limiting flow rate dependent on exit axial velocity.

Blade Losses

To reduce blade losses in transonic and supersonic stages of a high performance advanced aircraft engine compressors, multiple circular arc blading has been developed. These airfoils provide a means of controlling blade losses resulting from viscous and shock effects.

The alternate advanced compressor is not expected to reach the high supersonic Mach numbers encountered in aircraft engine compressors. Therefore, it is believed that double circular arc (DCA) blading should be quite satisfactory for this application where required. These blade profiles would result in lower losses than if the present W-501 conventional W65 series airfoils were used.

There is also evidence that further improvements in efficiency and flow range can be achieved through the use of lower aspect ratio blading. In addition to performance benefits, low aspect ratio blading will result in a further reduction of the number of blades in each stage. Low aspect ratio blades with longer chords and thicker profiles should also be more resistant to erosion and retain their performance over a longer period of time.

End Wall and Blade Loadings

Limits of stable operation of a compressor blade row depends on the aerodynamics of the blading and annulus endwalls. Loading limits exist for both blade and endwall boundary layers beyond which the flow becomes unstable.

Boundary layers which form the inner and outer flow path boundaries build up on the endwalls or casings. Losses can become quite severe, particularly in the corner regions where the blade surface and endwall boundary layers come together. In addition, because low velocity flow in the boundary layer areas cannot withstand the pressure gradients due to high mainstream velocities, secondary flows are set up within the boundary layers, adding losses within tip and corner areas of the blading.

Data from current research work and predicted improvements indicate that through special end wall treatment, end wall losses are likely to be reduced by a very substantial amount.

Tip Clearances

Improved performance should also be achieved by adopting closer running blade tip clearances as well as stator seal clearance. Tip clearance losses result from vortices created when the flow passing through the tip clearance space meets an adverse pressure gradient from the next blade. Reductions of running clearances should therefore reduce flow leakage and thus contribute to efficiency improvement. Clearance reductions could be achieved through the use of abradable casings or stationary shrouds and perhaps by the use of active clearance control in the latter stage rows.

In addition, a shorter rotor due to the reduced number of stages should contribute to a greater rotor dynamic stability, reducing the possibility of blade tip rubs which increase the tip clearance through wear.

Stage Matching

The higher pressure ratio per stage makes the matching problem more difficult. This is, however, compensated by the fewer number of stages and the use of low aspect ratio stages which have wider flow stability ranges. More extensive use of variable vane geometry will be needed with additional interstage bleed for engine starting.

From these considerations it is believed that a high efficiency compressor with a reduced number of stages and with a wide stable flow range can be developed by 1985 using higher loaded stages with lower aspect ratio blading and designed to account for end wall boundary layer effects.

Mechanical aspects from literature survey indicate fatigue life etc., applicable to predicted 1985 aircraft engine state-of-the-art but do not compare with the more conservative mechanical design life factors used in the design of utility size heavy duty combustor turbines.

Section 3

CONCEPTUAL DESIGN
(Subtask 9.2)

3.1 WORK STATEMENT

Based on initial data available, a conceptual design of the advanced compressor design will be defined. This conceptual design will provide a basic flow path description and preliminary values of general aerodynamic, thermodynamic and mechanical design parameters. The design will be compatible with the new-centerline engine conceptual design being prepared for EPRI under contract RP1187-2.

3.2 APPROACH

As required, both aerodynamic and mechanical conceptual design analysis have been applied in the Subtask 9.2 Conceptual Design work. The aerodynamic study effort was done by a systematic approach to identify and select candidate configurations for the first, middle, and last stages of a 13 stage, 14:1 total pressure ratio compressor. This effort was divided into three steps. First, the technology levels used in the parametric screening methods were selected on the basis of design variable ranges established in Subtask 9.1. Second, based on the results of the parametic study, more detailed aerodynamic studies were made for each of the three stages using 'meanline' and axisymmetric 'streamline' calculations. Finally, rotor and stator airfoil blade section shapes were developed which would provide adequate stall and choke margins consistent with the design Mach number and loading requirements.

The approach for mechanical analysis has concentrated on the conceptual definition of blading design features. The key stages, first, seventh, and last, of the reference design thirteen stage machine have been considered in this work. These and vibratory characteristics analysis of the rotating blades for these key stages has constituted the central area of study in this portion of Task 9.2.

3.3 AERODYNAMIC DESIGN SELECTION

State Parametric Screening Studies

First stage parametric screening studies were made for the full range of values for the inlet hub-tip ratio (0.5 to 0.7), mean flow coefficient (0.5 to 0.6) and hub reaction (0.5 to 0.7). Stage work was selected at 15 Btu's/lb. (34.9 kJ/kg) which is equivalent to a stage pressure ratio of approximately 1.4. The results of this study are shown by Figures 1,2, and 3. Stage reaction, which is the percentage of rotor static enthalpy rise for the stage, is a significant parameter in establishing both rotor and stator inlet Mach numbers as well as the row wall loading $\Delta P/q'$. It has no apparent leverage in determining the stage tip diameter which is largely dependent on hub-tip ratio and mean flow coefficient. Rotor and stator solidity have not been a consideration at this point in the study but will be selected in the stage detail design phase based on blade airfoil section stall and choke margin and the attainment of acceptable levels of airfoil section diffusion factors.

A typical first stage design, with approximate levels of rotor and stator inlet Mach numbers determined so far would lead to stage efficiencies as shown in Figure 4. A range of rotor total loss coefficient from 0.05 to 0.13 was chosen with a constant value of stator total loss coefficient of 0.05. Using some typical rotor and stator Mach numbers for radial positions of hub, mean and tip, and estimated overall first stage efficiency would be in the range of 0.820 to 0.880. This representative of stage efficiency as denoted by survey configurations #4, #5, and #6.

The screening of survey compressors indicated a trend for last stage inlet hub-tip ratio of 0.900 to 0.930. Figures 5 and 6 show the results of last stage parametric screening studies with survey compressor configurations #1, #2, and #3 superimposed for comparison on Figure 6. The survey indicates that last stage inlet hub-tip ratios for advanced technology compressors are in the range of 0.908 to 0.923, however, compressor configurations #1 and #2 are the only ones relevant to the ACDS program because the exit referred flow for each of these closely matches the exit conditions for the ACDS compressor. Stage hub reaction and mean flow coefficients values were selected for parametric screening which were consistent with the inlet stage results. As can be seen from the rotor and stator Mach number levels, they should not represent any aerodynamic or blade design problems.

The major last stage problem is a trade-off between lowering hub-tip ratio to minimize tip leakage losses and simultaneously not exceeding acceptable aerodynamic loading limits of wall loading $\Delta P/q$ and diffusion factor as wheel speed comes down. Because of the low Mach number encountered in this stage, solidity can

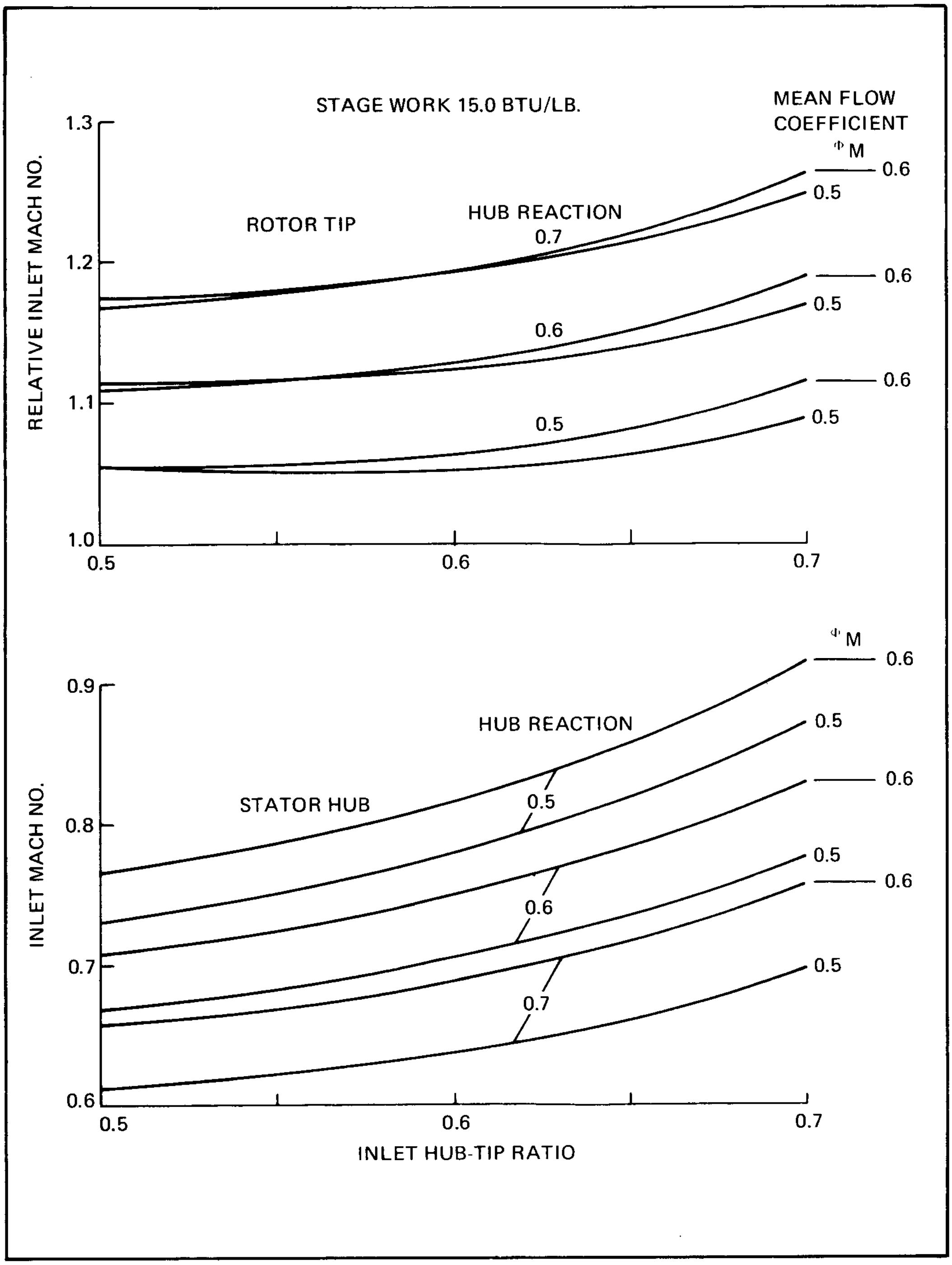


Figure 1. First Stage Parametric Study; Relative and Absolute Inlet Mach Number vs. Inlet Hub-Tip Ratio

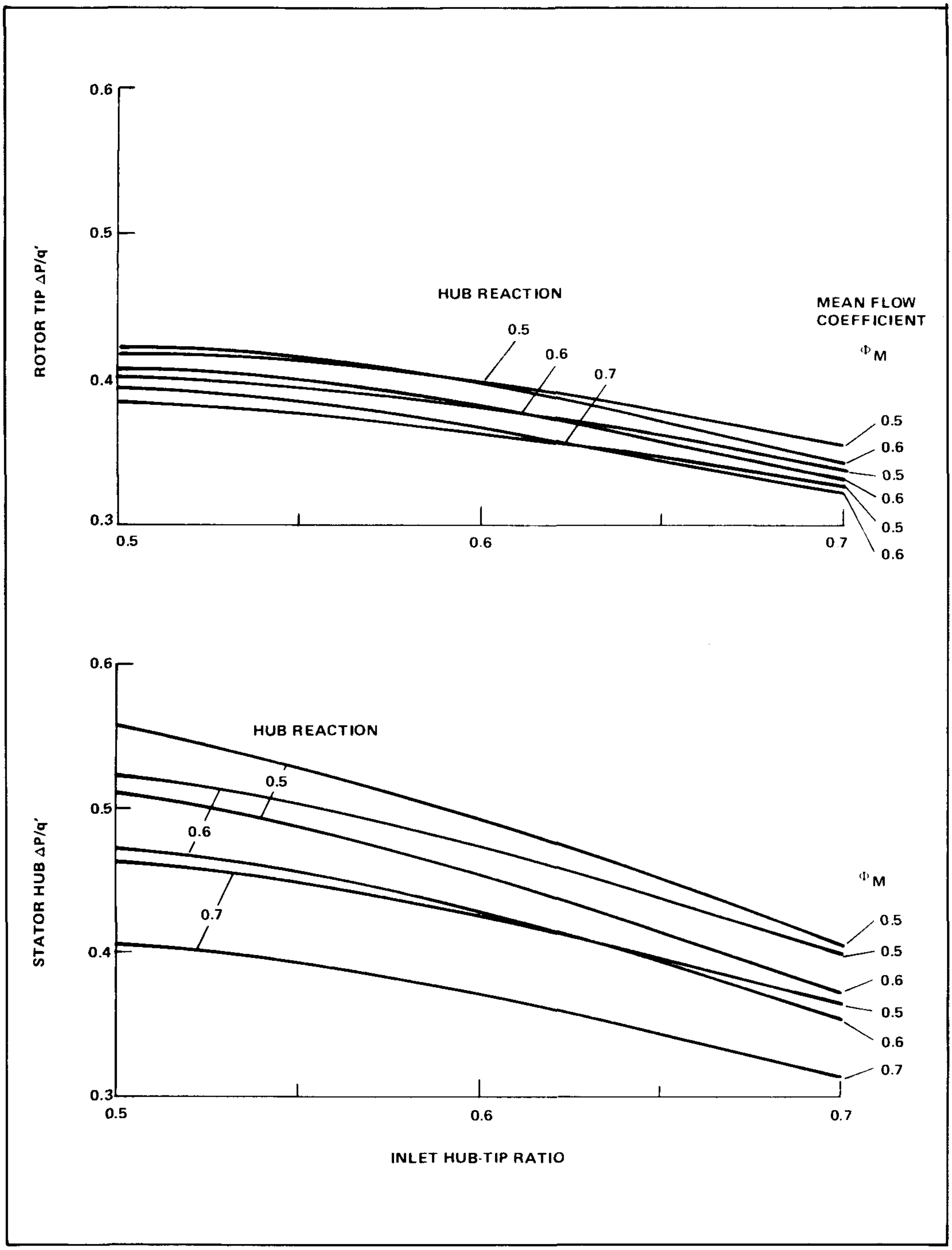


Figure 2. First Stage Parametric Study; Rotor and Stator Loadings vs. Inlet Hub-Tip Ratio

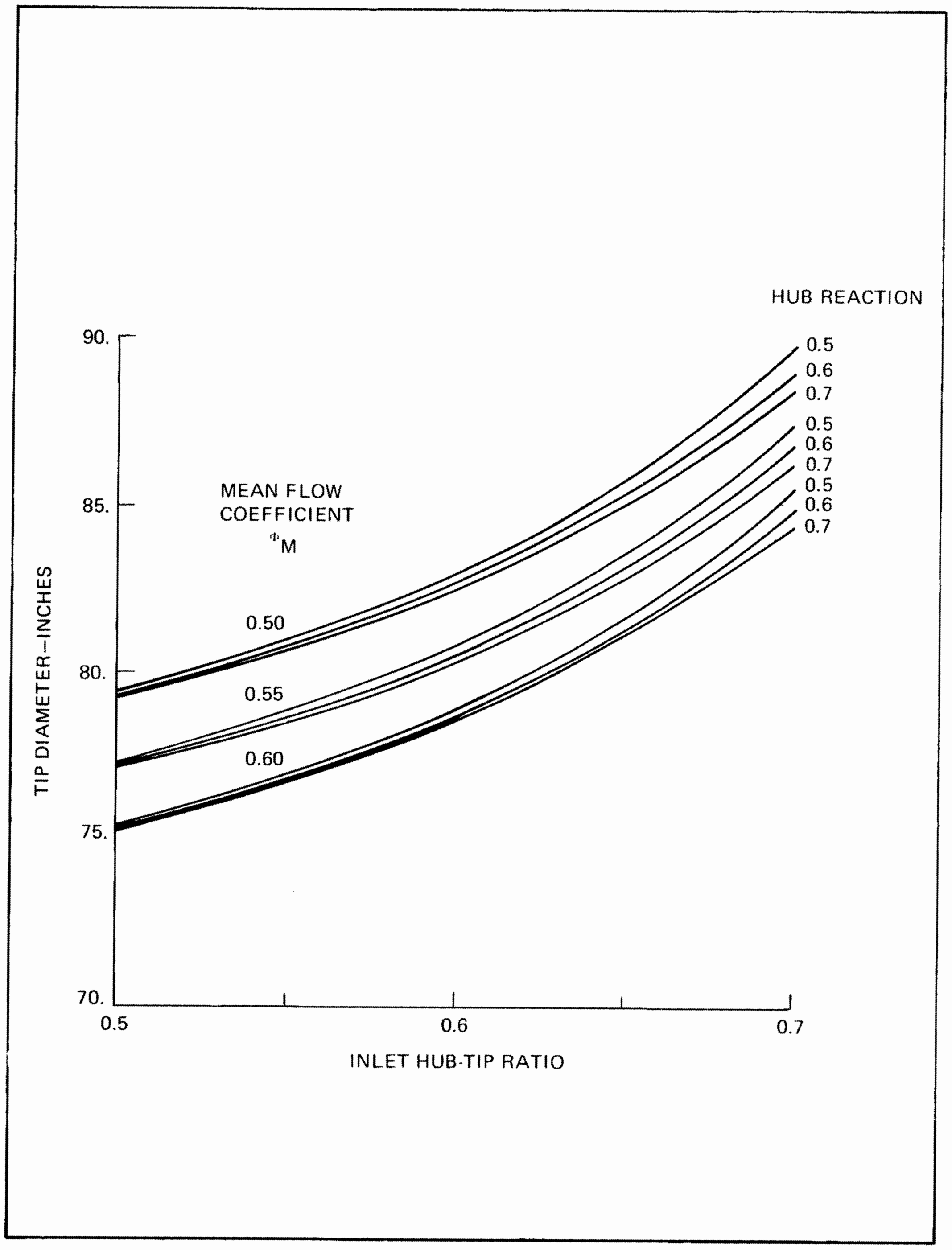


Figure 3. First Stage Parametric Study; Tip Diameter vs Inlet Hub-Tip Ratio

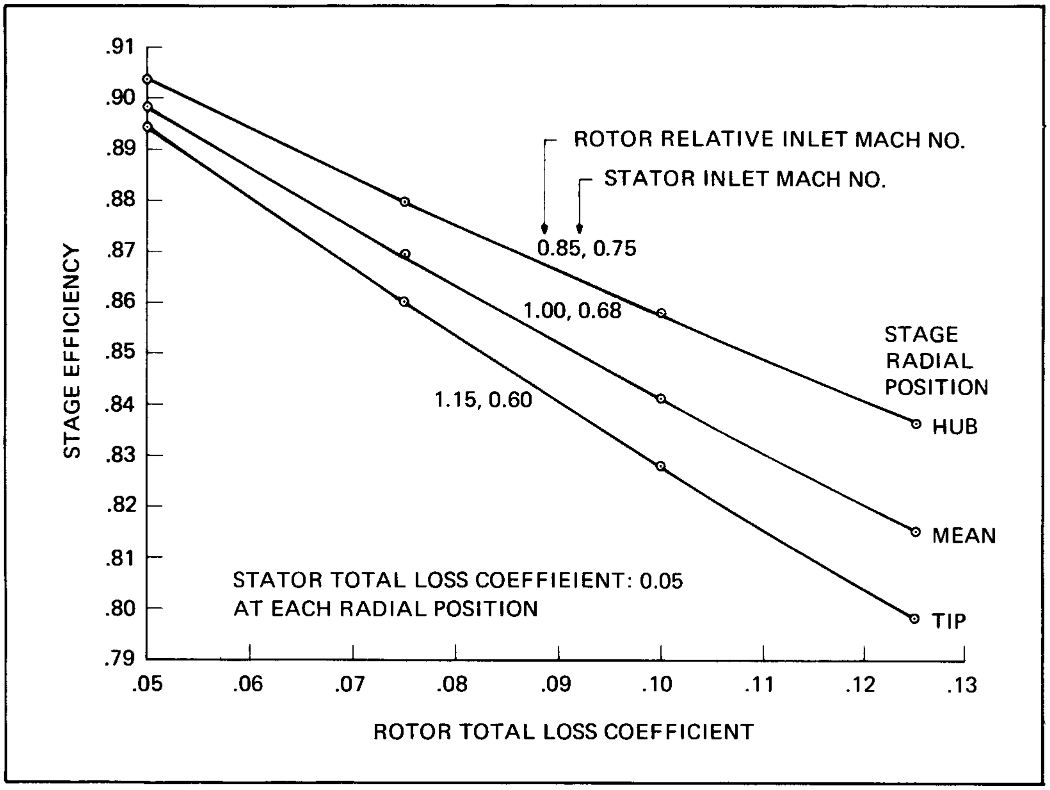


Figure 4. First Stage Estimated Range of Efficiency

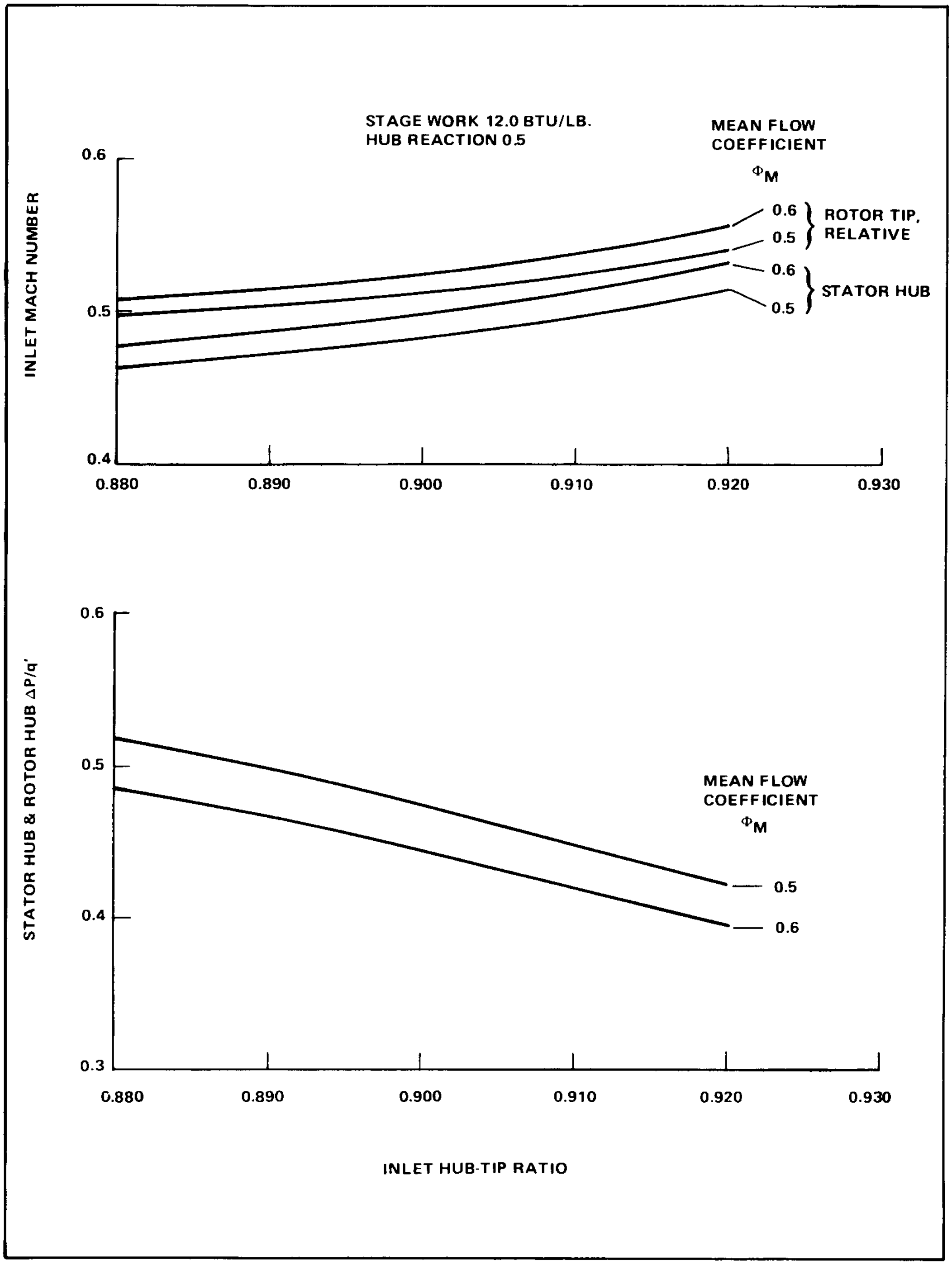


Figure 5. Last Stage Parametric Study; Inlet Mach Number and Stator and Rotor Hub Wall Loading vs. Inlet Hub-Tip Ratio

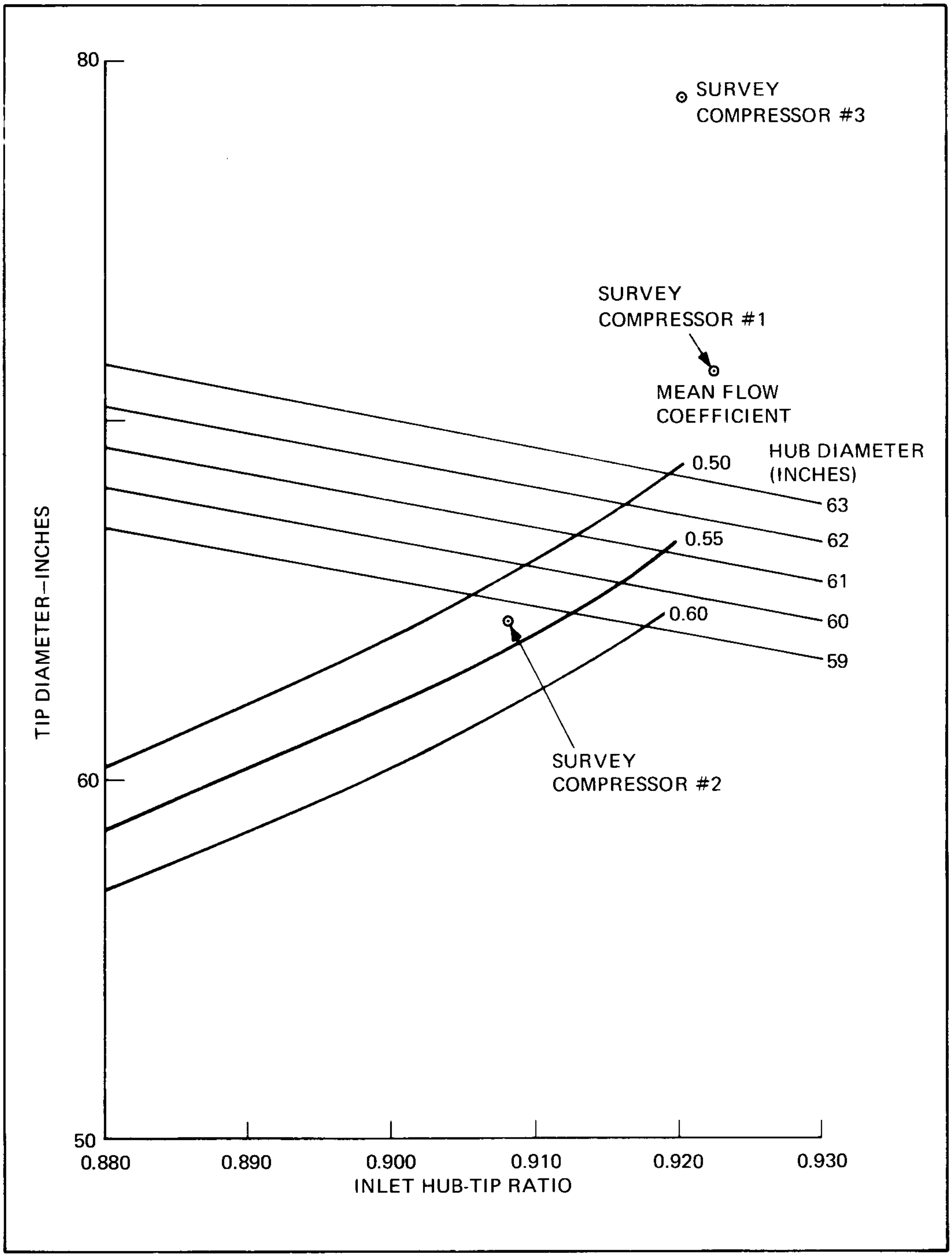


Figure 6. Last Stage Parametric Study; Tip Diameter vs. Inlet Hub-Tip Ratio

be kept high, to keep diffusion factors within tolerable limits without any detriment to blade passage choking, however, some degree of caution must be exercised such that blade element airfoil efficiency is not seriously jeopardized.

The characteristic of the middle stage is a more moderate level of critical stage Mach numbers than the first stage without the hub-tip ratio constraints of the last stage. This permits a higher level of stage work than for either the first or last stages. Figures 7 and 8 indicate inlet Mach numbers to the rotor and stator of between 0.6 and 0.7. If some overall consideration is given to maintaining an approximately constant mean diameter compressor, an initial selection of hub-tip ratio can be made for the middle stage. From reviewing the survey compressors, a trend toward constant mean diameter designs was evident. To complete the parametric screening, stage mean flow coefficient and reaction were selected, again consistent with first stage results.

Preliminary Compressor Design

The design point requirements of overall pressure ratio, inlet referred flow and RPM were established from cycle requirements. The reference first, middle and last stage configurations that meet these requirements were selected from the stage parametric screeening studies. The final three reference stage designs for the 13 stage compressor are:

Stage	Hub-Tip Ratio	Stage Work Btu/lb	Mean Flow Coefficient	Hub Reaction
First	0.600/0.650	12.0	0.55	0.5/0.6
Middle	0.852	13.5	0.55	0.5
Last	0.920	12.0	0.55	0.5

Using preliminary first, middle and last stage configurations, a preliminary overall compressor flow path aerodynamic design, using 'meanline' calculation procedures, was determined. This design system consists of a 'meanline' analysis of all stages based on estimated values of stage efficiency, end-wall blockage, and stage exit swirl angle generally consistent with previously designed Westinghouse high efficiency compressors. A stage work distribution was chosen which would satisfy the overall pressure ratio of the 13 stage compressor including

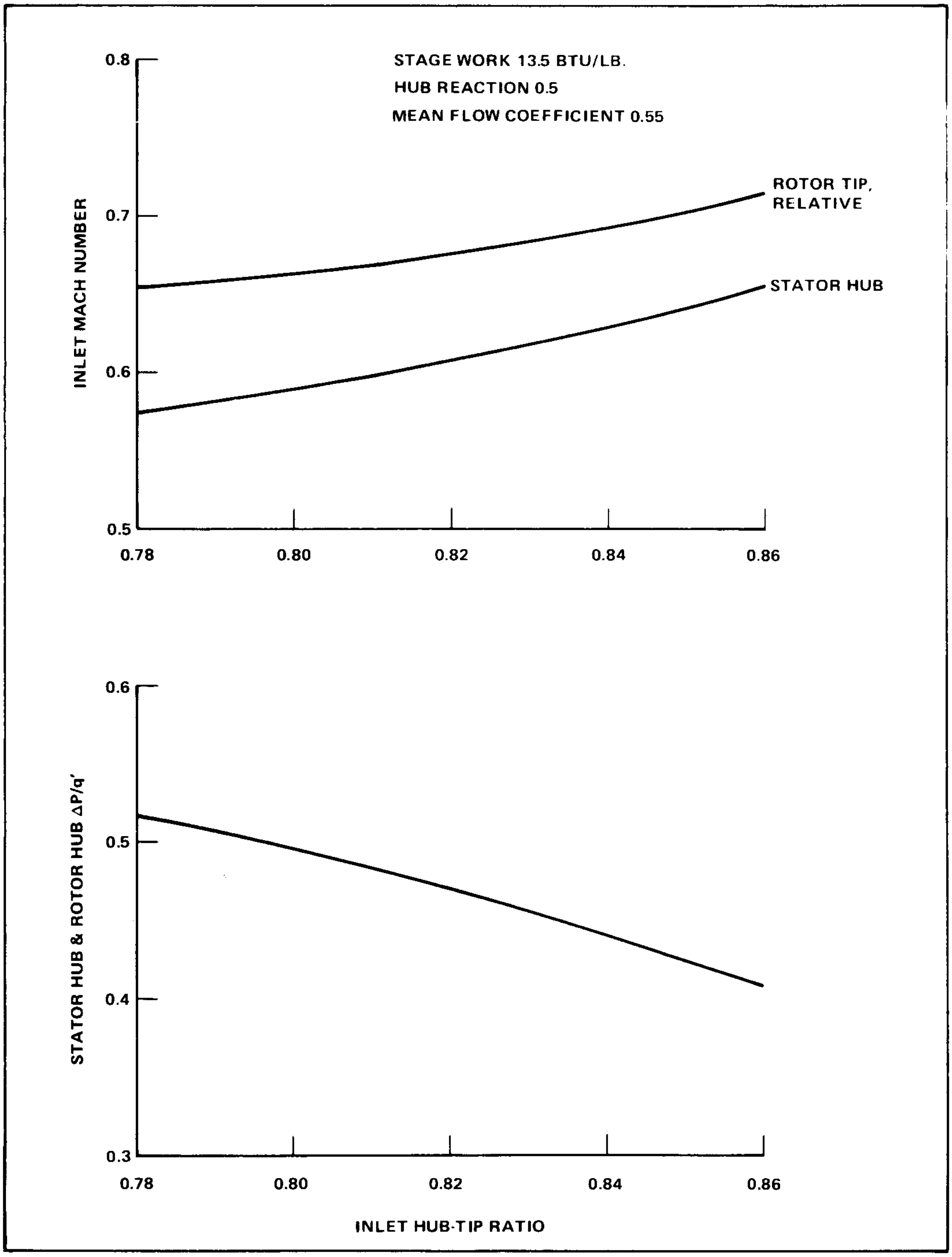


Figure 7. Middle Stage Parametric Study; Inlet Mach Number and Stator and Rotor Hub Wall Loading vs. Inlet Hub-Tip Diameter

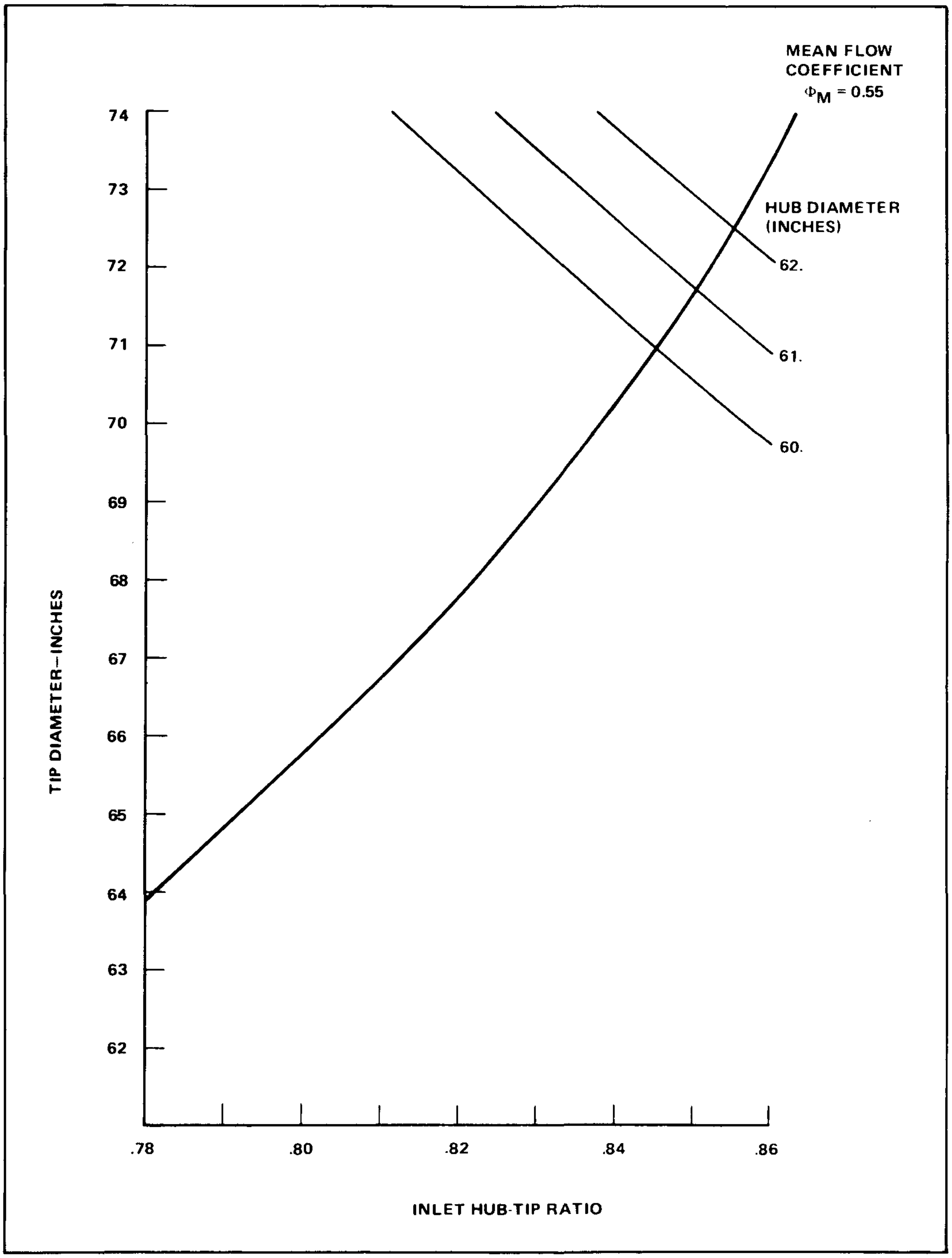


Figure 8. Middle Stage Parametric Study; Tip Diameter vs. Inlet Hub-Tip Ratio

the inlet and outlet guide vanes and exit diffuser. The 'meanline' analysis program calculates 'meanline' velocity triangles by solution of the continuity, momentum and energy equations. The resultant of this design system defines all stage flowpath diameters, thermodynamic conditions, axial velocities, cooling bleeds and estimated overall compressor efficiency.

Certain assumptions of axial distributions of independent compressor design parameters are necessarily used to establish the flowpath shape. The general approach was to initially design for a constant mean diameter flowpath, using as a guide, the preliminary stage characteristics selected from the parametric screening studies. This procedure is somewhat iterative only until a selection of all stage design variables is established which best fits the overall compressor program objectives. The final choice of flowpath for the compressor conceptual design is shown on Figure 9 using the three final reference stage designs

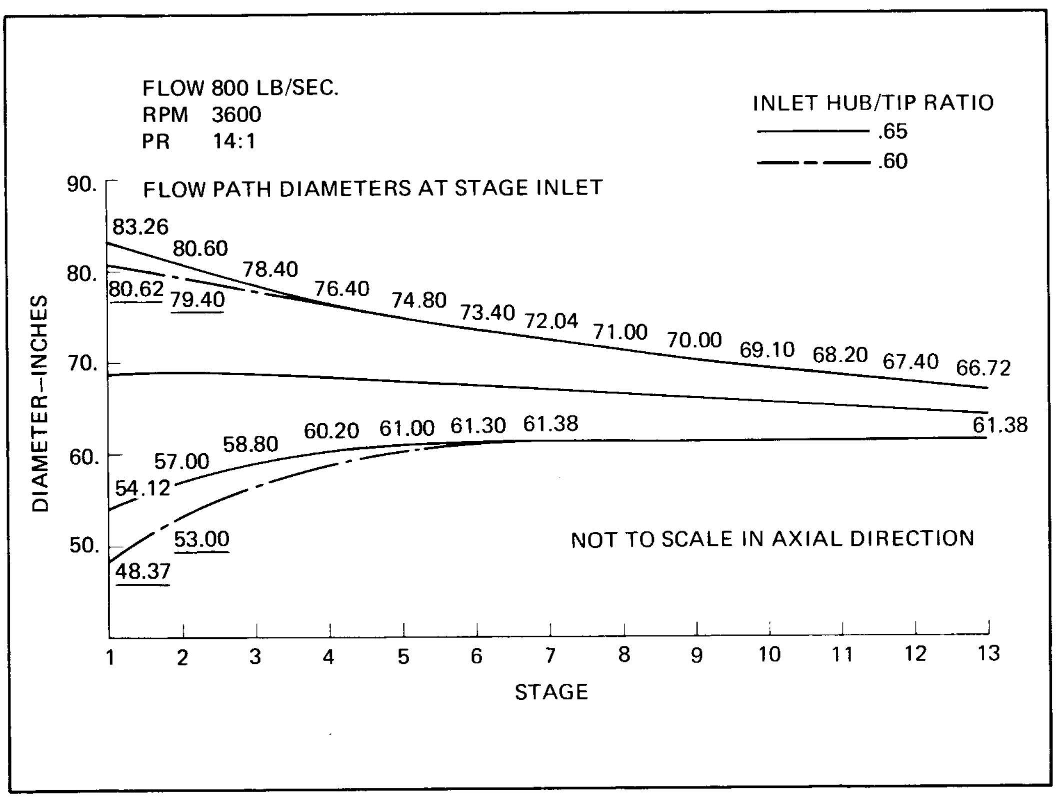


Figure 9. Flowpath Diameters at Stage Inlet for the Reference Thirteen Stage Compressor.

tabulated above for the first, middle and last stages. The hub diameter was chosen as constant from the middle through the last stage. Two hub-tip ratio designs are displayed as alternatives for the first stage. They represent the best fit of survey data and each is to be evaluated further in the detail stage design where an interaction of both aerodynamic and mechanical requirements will establish the stage design with the most potential.

The stagewise distribution of axial velocity is represented by the mean stage flow coefficient. The mean flow coefficient is maintained nearly constant throughout the entire flow path providing a nearly constant axial Mach number in the front one-third and then decreasing steadily to provide an exit Mach number of 0.3 at the design point. The stagewise distribution of hub reaction was held constant throughout the flowpath at 0.5. For the case where the first stage hub reaction is 0.6, it would be gradually reduced to 0.5 through the first half to finally reach a level of 0.5 by the middle stage. This selection of reaction was chosen to maintain a moderate level of blade inlet Mach number and a desirable distribution of loading between the stage rotor and stator which experience indicates would produce reasonable blade profile designs.

The number of stages for this study was fixed at 13 for this preliminary conceptual design. Other compressor configurations of 11 and 15 stages will be studied along with the 13 stage design in the follow-on DOE-sponsored program subtasks to establish a range in the number of stages to provide a variation in pressure ratio per stage and to establish a measure of the degree of risk or aggressiveness in design.

Stage Detail Design

The three final reference stages, inlet, middle and last, identified in the final screening studies and used to establish the final overall compressor flow path configuration, were selected for further more detailed aerodynamic design study. During this phase of the aerodynamic design process the three stages were more fully defined, and initial blade and vane profiles selected.

The definition of the flowfield calculation for each stage was accomplished by using a 'streamline' analysis program to establish full-span aerodynamics. The axisymmetric flow calculations determine vector diagram parameters and fluid properties along seven design stream surfaces. Flowpath annulus geometry, inlet temperature and pressure, rotor RPM, flow, rotor work and stator absolute swirl were input to each of the three stage calculations. These inputs were based on data generated during the screening and preliminary compressor 'meanline' analysis procedures. These detailed aerodynamic stage designs establish Mach number and aerodynamic loading levels in a full spanwise direction and provide a means for the initial selection of the number of blades and vanes and their respective chords.

Selected results of this streamline analysis for stage 1 are tabulated in Table 4. Four first stage designs have been carried through detail aerodynamic

Table 4

SELECTED AERODYNAMIC STAGE CHARACTERISTICS OF FOUR ALTERNATIVE FIRST STAGE DESIGNS

	Stage 1-1	Stage 1-2	Stage 1-3	Stage 1-4
Inlet hub-tip ratio	0.600	0.600	0.650	0.650
Hub Reaction	0.5	0.6	0.5	0.6
Rotor tip relative inlet Mach number	1.036	1.100	1.046	1.114
Rotor meanline Diffusion factor	0.388	0.379	0.372	0.365
Rotor meanline Wall-loading	0.340	0.336	0.303	0.304
Rotor tip speed, ave., ft/sec.	1262.	1262.	1300.	1300.
Rotor total pressure Ratio	1.337	1.334	1.334	1.331
Stator hub Inlet Mach number	0.759	0.679	0.823	0.726
Stator meanline Diffusion factor	0.372	0.360	0.348	0.325
Stator meanline Wall-loading	0.315	0.277	0.282	0.247
Stage efficiency	0.856	0.855	0.844	0.844

design in order to provide alternate design choices for mechanical stress and frequency analysis. Any one of the four designs may be used, based on the judgment of which is most suitable to achieve the overall compressor design objectives.

Selected results of this streamline analysis for stage 7 and stage 13 are tabulated in Table 5. There are intended to be used only with the 13 stage configuration.

Table 5

SELECTED AERODYNAMIC STAGE CHARACTERISTICS OF STAGES 7 AND 13

	Stage 7	Stage 13
Inlet hub-tip ratio	0.852	0.920
Hub reaction	0.500	0.500
Rotor tip relative Inlet Mach number	0.723	0.549
Rotor meanline Diffusion factor	0.387	0.403
Rotor meanline Wall-loading	0.368	0.392
Rotor tip speed, Avg., ft/sec	1128.	1049.
Rotor total pressure ratio	1.241	1.151
Stator hub inlet Mach number	0.643	0.512
Stator meanline Diffusion factor	0.413	0.435
Stator meanline Wall-loading	0.391	0.428
Stage efficiency	0.898	0.897

After the axisymmetric streamline flow analysis was completed, preliminary blading was selected for each blade and vane using established Westinghouse design correlations. Preliminary values of incidence angles, deviation angles, solidity, and thickness to chord ratio were selected. The definition of each row was carried sufficiently far to determine if any major aerodynamic problem areas existed and to provide initial airfoil size for structual analysis iterations, but not so far as to represent finished airfoil designs. Selection of basic blade and vane section shapes for the three reference stages was based on the Mach number regime in which they would operate, in order to provide adequate design stall and choke margin. In general, the first stage rotor blades and vanes which operate in the supersonic Mach number regime are DCA (double circular arc) airfoil shapes. Blading in the subsonic Mach number regime of the middle and last stages will use the standard Westinghouse W65 airfoil shapes.

Some of the more important aerodynamic blade and vane airfoil features selected during the aerodynamic design phase for stages 1, 7, and 13 are tabulated in Table 6 and Table 7.

Table 6

SELECTED STAGE ONE AIRFOIL CHARACTERISTICS FOR FOUR ALTERNATIVE DESIGNS

	Stage 1-1	Stage 1-2	Stage 1-3	Stage 1-4
Rotor				
Number of blades	23	23	24	24
Aspect ratio	1.508	1.508	1.495	1.492
Solidity (meanline)	1.100	1.100	1.000	1.000
Max thickness to chord (tip)	.030	.030	.030	.030
Max thickness to chord (hub)	.090	.090	.090	.090
Blade section profile	DCA	DCA	DCA	DCA
Stator				
Number of vanes	36	36	36	36
Aspect ratio	1.928	1.928	1.935	1.957
Solidity (meanline)	1.200	1.200	1.100	1.200
Max thickness to chord (tip)	.100	.100	.100	.100
Max thickness to chord (hub)	.050	.060	.060	.060
Vane section profile	W65	W65	W65	W65

From the selected results of this analysis, Mach numbers are seen to moderate after the first stage. The outlet guide vane following the last stator will reduce the exit airflow Mach number to 0.3 in addition to removing all exit swirl before entering the diffuser. Pitchline wall loading indicates a gradual increase to more highly loaded rear stages. Loadings for all stages are somewhat higher than current Westinghouse experience and will require further design refinement in axial velocity and/or compressor stage work distributions. However, they are not believed to present any unsolvable problem in achieving the program goals.

Table 7

SELECTED STAGE SEVEN AND THIRTEEN AIRFOIL CHARACTERISTICS

	Stage 7	Stage 8
Rotor		
Number of blades	87	148
Aspect ratio	1.501	1.505
Solidity (meanline)	1.343	1.200
Max thickness to chord (tip)	.050	.050
Max thickness to chord (hub)	.130	.130
Blade section profile	W65	W65
Stator		
Number of blades	82	148
Aspect ratio	1.505	1.505
Solidity (meanline)	1.200	1.200
Max thickness to chord (tip)	.100	.100
Max thickness to chord (hub)	.100	.100
Vane section profile	W65	W65

3.4 MECHANICAL DESIGN CONSIDERATIONS

Conceptual Design Selection

Conceptual Mechanical Design studies were conducted in parallel with the conceptual aerodynamic study to determine if there are any mechanical considerations that might lead to significant changes of the blade and vane chords and thicknesses for the three basic stages selected in the aerodynamic study.

The mechanical design effort was therefore concentrated on blade and vane considerations which consisted of preliminary determination of blade and vane stresses and blade natural frequencies. The rotor itself was assumed to be of the integral drum type configuration made from a single piece forging or welded elements. The mechanical layout of this compressor configuration will be carried out under Tasks 1 and 3 of the DOE funded follow-on program.

A critical speed study of the latest configuration of the high reliability turbine has not been performed. A preliminary analysis of an earlier version of the engine with both a nineteen stage and a eleven stage compressor indicated an increase of both the first and second critical speeds compared to the W501 engine. A potential problem with the second critical was noted; corrective steps will be applied to future design efforts.

It is anticipated that the closer blade tip clearances required to achieve the high efficiency goal will necessitate the introduction of active blade tip clearance control on the rear stages. This will be studied in further detail in the preliminary design covered under Task 3 of the DOE funded program.

Mechanical Analysis

The three reference stages (inlet, middle and last), established during the aerodynamic design phase were used as a starting point for the mechanical analysis. Blades and vanes were analyzed to determine their stress levels as well as their vibratory characteristics. This analysis covered four variations of stage one rotor blade (Figure 10); four variations of stage one variable geometry vanes (Figure 11); stage seven blade and vane (Figures 12 and 13); and stage thirteen rotating blade (Figure 12) and vane (Figure 13). Because of the wide chord of the first stage blade airfoil an axial dovetail root based on an existing W501 dovetail design was selected. Tangential dovetails were selected for stage seven and thirteen blade root attachments. At this point it is assumed that the second stage blade will also feature an axial dovetail and the remaining stages will all incorporate circumferential dovetails. The first stage vane, because of variable geometry requirements, was designed as a cantilevered airfoil. A non-supportive inner shroud, retained by pins extending from the vane's inner diameter, was treated analytically as a load applied to the vane tips. To avoid the troublesome fabricated junction between the vane airfoil and the shrouds, stage seven and thirteen vanes may be made of cast segments.

A preliminary dynamic mechanical analysis by computer program based on the three reference airfoil sections (hub, midspan, tip) derived from the aerodesign and interpolated to provide twenty mass points along the span of each blade or vane was conducted for each typical stage investigated. Scale factors ranging from 0.50 to 1.50 were applied to the reference blade and vane chord lengths to determine the optimum chord dimensions. Once the preliminary stresses and natural frequencies of the blade (or vane) were calculated, the magnification

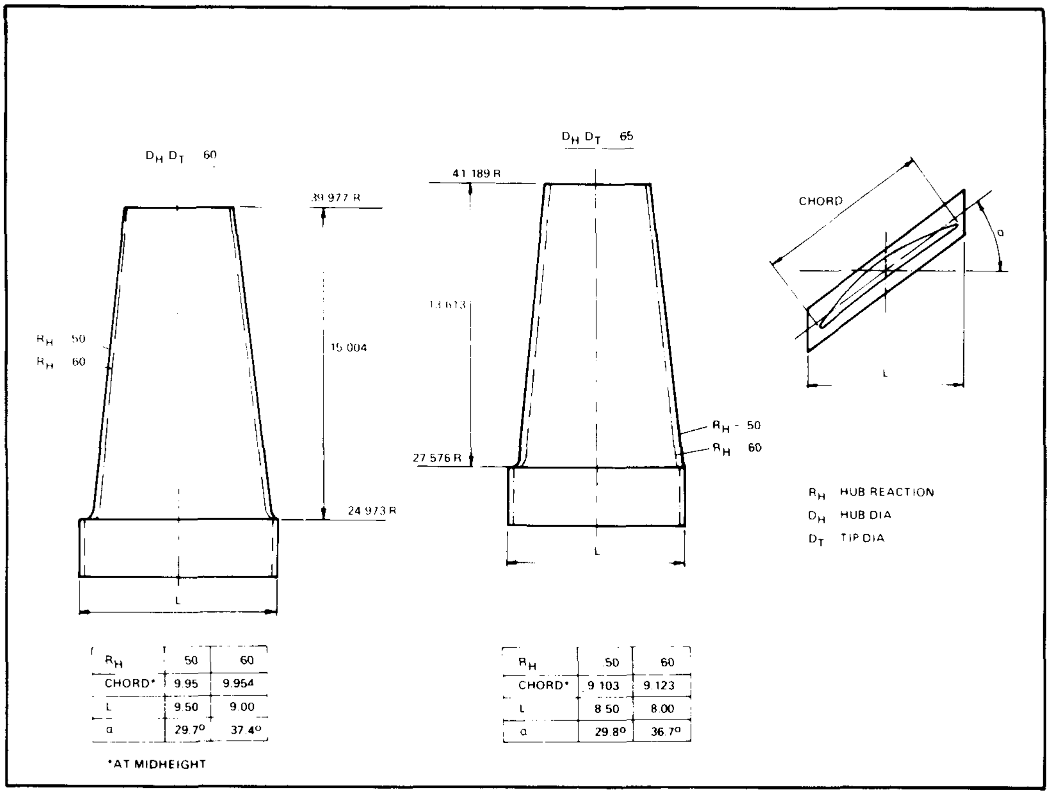
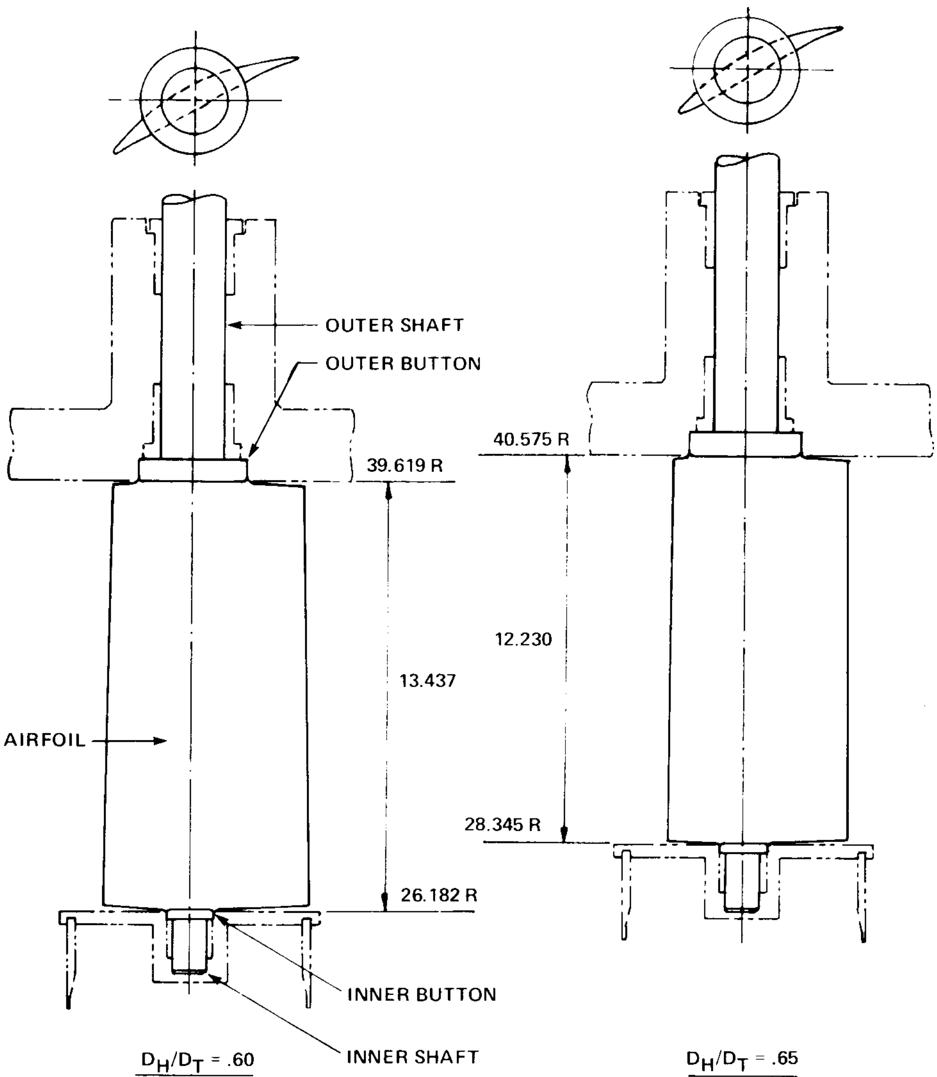


Figure 10. Stage 1 Rotating Blade

factor (of stress) and the engine harmonic order (based on frequency) were also determined. From experience, there is a minimum magnification factor (a_{min}), for each harmonic, below which the probability of blade failure is higher. The relationship between the minimum magnification factor and the harmonic can be expressed approximately by the equation, $a_{min} = 30/H$, see Figure 14. The ratio between the calculated magnification factor (a_{calc}) and the minimum magnification factor (a_{min}) is called the Load Build Up (LBU). The design requirement for blades, expressed in terms of the LBU, is;

$$L.B.U. \geqq 1.00$$



R_H	.50	.60
CHORD*	6.97	6.97
CAMBER*	26.408	26.608

R_H	.50	.60
CHORD*	6.32	6.25
CAMBER*	21.069	26.797

*AT MIDHEIGHT

Figure 11. Stage 1 Stationary Vane

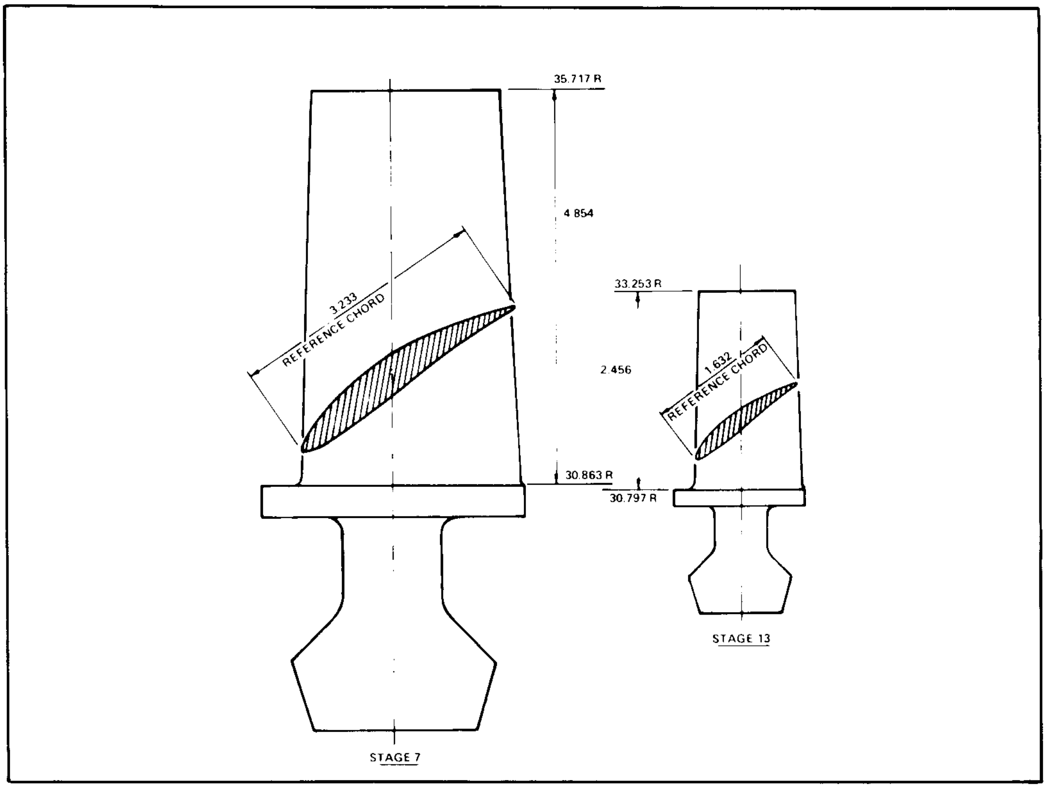


Figure 12. Stage 7 and 13 Rotating Blades

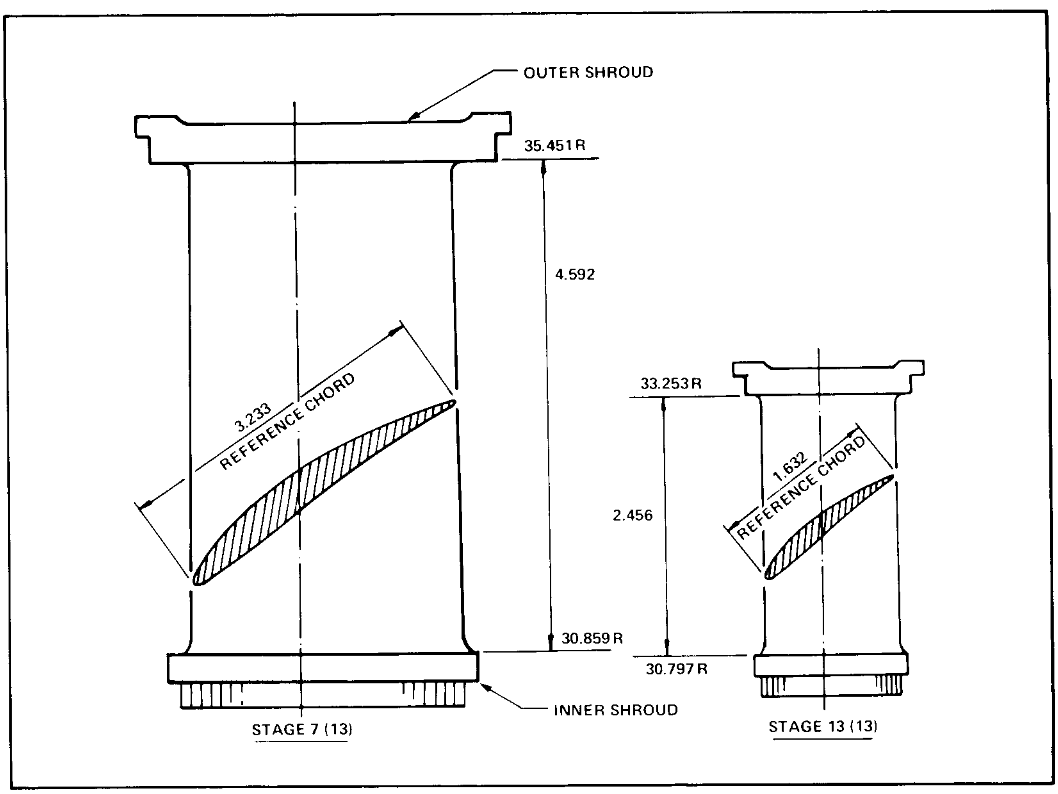


Figure 13. Stage 7 and 13 Stationary Vanes; Cast Segments

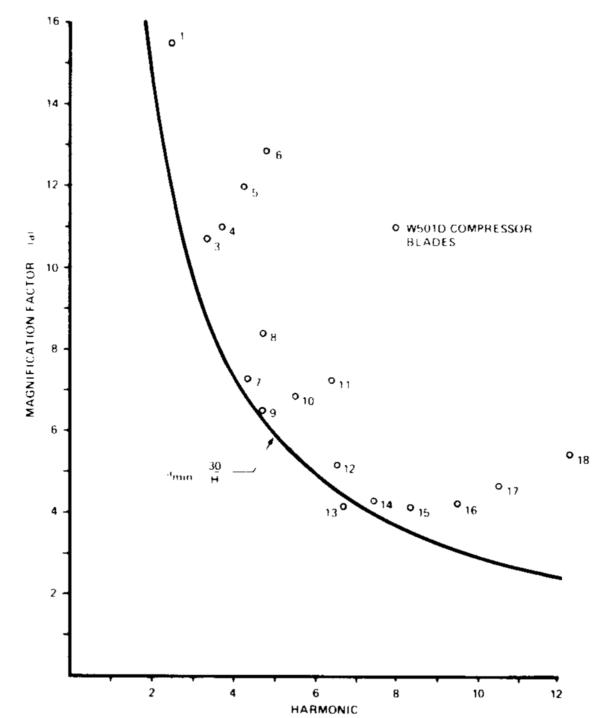


Figure 14. W501D Compressor Blade Magnification Factors

This is to say that on a plot of magnification factor versus harmonic, the calculated value of magnification factor must fall on or above the a_{min} curve. For reference, values of the W501D compressor blades are plotted in Figure 14. A similar plot, using preliminary calculation for the first, seventh and thirteenth stage blades of the reference low aspect ratio compressor, are given in Figures 15 and 16.

A scale factor of one or greater has been applied to the first stage blade in order to meet the design requirement. On the other hand, the blade chords of stages 7 and 13 can be reduced to 75 percent of the reference chord length and still meet the design requirements.

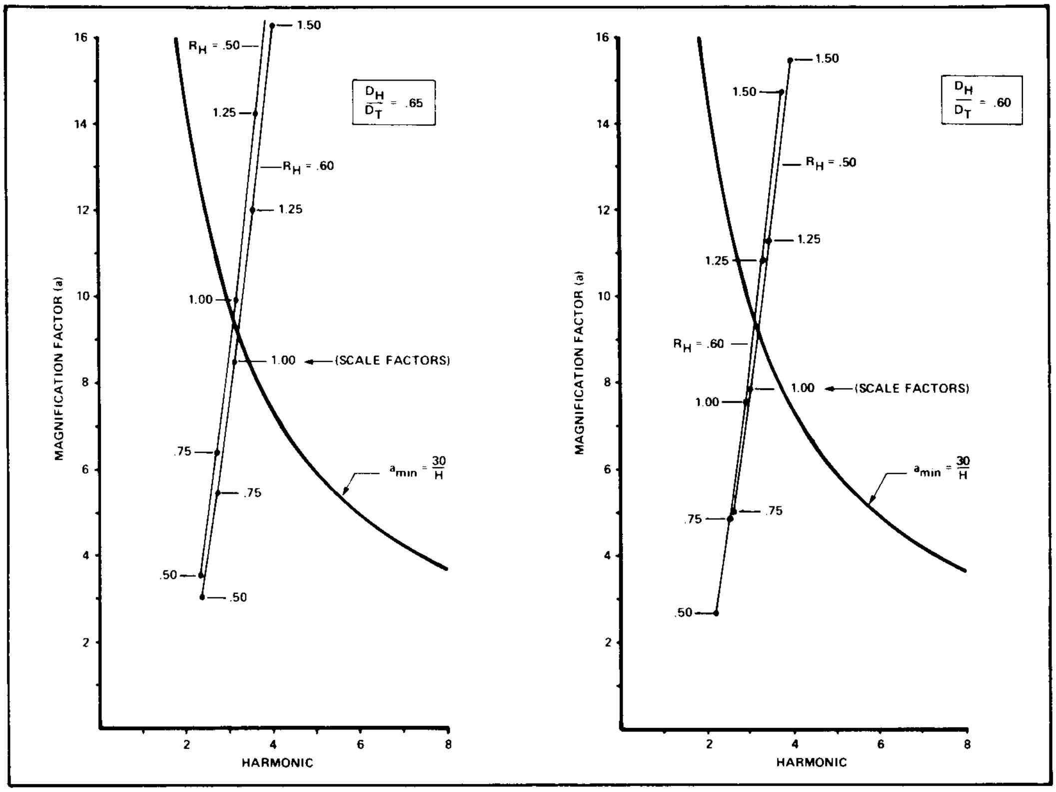


Figure 15. Conceptual Design First and Middle Stage Rotor Blade Magnification Factors

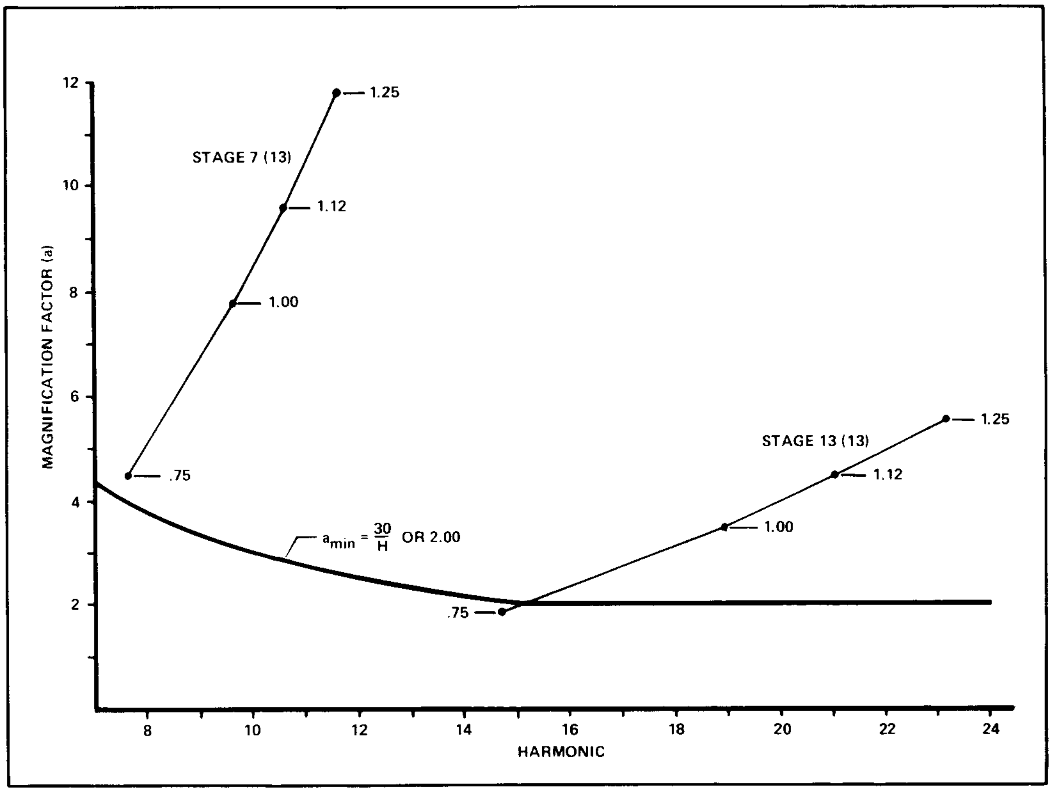


Figure 16. Conceptual Design Middle and Last Stage Rotor Blade Magnification Factors

Complete design analysis of the stationary vanes, as well as the design and analysis of the dovetail roots for the rotating blades of remaining stages, and the groovewall stresses of the compressor will be carried out in more detail under the DOE funded follow-up program.

This conceptual mechanical design analysis was carried out in sufficient detail to assure that the objective of high reliability, durability and ease of maintenance could be met, using predicted 1985 industrial technology based on advanced aero engine compressor development.

Section 4

SUMMARY

This preliminary study was conducted to define a conceptual design of an alternate compressor for the High Reliability Combustion Turbine Engine based on the latest advances made in aero engine compressor technology. This study conducted under Task 9 of EPRI RP1187-2 contract consisted of two major subtasks.

The initial effort involved a data search of various experimental advanced aircraft compressors, and an evaluation of the technology to determine the applicability of certain technological advancements which may be introduced in heavy industrial combustion turbine engines. This review also included a projection of the 1985 technology. The second part of the study covered the preliminary definition of an advanced industrial compressor based on the latest aircraft engine compressor technology and projected advances to be available in 1985 compressor aerodynamic and mechanical design areas.

This design study resulted in the conceptual definition of a 13 stage compressor design providing a pressure ratio of 14:1 with an inlet airflow of 800 lbs/sec (360 kg/sec). This compressor concept, using low aspect ratio blading, will be analyzed in further details in a follow-on program funded by DOE under Contract No. DE-AC03-79E15332. This compressor will also be evaluated against a more agressive approach with eleven stages and a more conservative design consisting of 15 stages, as well as against the present W501 engine compressor. Finally, depending upon the results of their evaluation, a compressor configuration will be selected as a candidate for further study and eventual development.

Section 5

CONCLUSIONS AND RECOMMENDATIONS

Conclusions

This design study identified a number of design factors and constraints leading to a basic conceptual design of an advanced alternate compressor for the High Reliability Combustion Turbine.

- A high performance utility engine compressor with fewer stages featuring low aspect ratio blading based on aero engine advanced compressor technology can be developed to meet the High Reliability Combustion Turbine engine performance goals and reliability requirements.
- To maintain acceptable level of stage work loading, higher blade tip speed will be required.
- Inlet hub-tip ratio will be increased from the present value of 0.5 for the W501 compressor to 0.6 or 0.65.
- Exit hub-tip ratio should also be increased from 0.87 for the W501 to 0.92.
- To maintain closer clearance during normal operation in order to reduce blade tip clearance losses, active tip clearance control may be required in the rear stages area.
- Variable geometry stators will be required for the first two or three stages in addition to the inlet guide vanes.
- Double circular arc (DCA) blade profiles will be required to minimize shock losses in the front stages.
- Higher hub-tip ratio may be beneficial from an overall engine design viewpoint by reducing the rotor thrust balance problem associated with the greater diameter of the three stage turbine.
- Lower aspect ratio blades, with their wider chords, may provide longer blade erosion life, resulting in less engine performance deterioration.
- The reduced number of blades and vanes resulting from the fewer stages and low aspect ratio blading should result in a lower compressor cost. However some of this cost reduction may be offset by the additional complexity due to increased variable geometry and active blade tip clearance control.

Recommendations

Due to the potential advantages listed in the above conclusions, it is recommended that further investigation and design work be carried out in order to establish a basic preliminary design of an optimum compressor configuration which could be developed eventually for the High Reliability Combustion Turbine Engine.