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AN OVERVIEW OF AEROSPACE GAS TURBINE TECHNOLOGY OF RELIVANCE TO THE DEVELOPMENT OF THE AUTOMOTIVE GAS TURBINE ENGINE

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ABSTRACT

The NASA-Lewis Research Center (LeRC) has conducted, and has sponsored with industry and universities, extensive research into many of the technology areas related to gas turbine propulsion systems. This aerospace-related technology has been developed at both the component and systems level, and may have significant potential for application to the automotive gas turbine engine. This paper summarizes this technology and lists the associated references. The technology areas included are: system steady-state and transient performance prediction techniques, compresscr and turbine design and performance prediction programs and effects of geometry, combustor technology and advanced concepts, and ceramic coatings and materials technology.

THE NATION HAS ESTABLISHED THE GOAL OF reducing its consumption of petroleum derived fuels. Since the transportation sector consumes over 50% or approximately 1.9×10^{13} mega joules (18 QUADS) per year of the crude oil used in the nation, it is a prime area for implementing conservation measures. For highway vehicle applications, the gas turbine has been identified as a viable alternative propulsion system (1)*. It has the potential for improved fuel economy over the present spark ignition internal combustion engine while meeting future emission standards, and has the inherent ability to operate not only on petroleum derived fuels but also on fuels derived from other sources.

Most of the technology base for the gas turbine has evolved from over 30 years of effort by the military, NASA, and the industry to meet the military and commercial needs for improved aircraft propulsion systems. For NASA, these technology activities have been conducted both in-house and through contracts and grants with industry and universities by its lead center in propulsion technology, the Lewis Research Center (LeRC). Because the automotive gas turbine will build upon the aerospace base, it is the purpose of this paper to describe aerospace related technology developed in programs conducted by LeRC that may help the gas turbine powered automotive vehicle to meet or exceed the goal that has been set. Although well known to the automobile industry, the general status and character of the automotive gas turbine and its differences from the aircraft application will be outlined to provide a basis for discussion.

Installing a gas turbine in an automotive vehicle is not a new idea. Programs have been conducted by many firms over approximately the last 30 years as reported in (1). The following is an excerpt from the reference:

"Rover demonstrated the world's first gas turbine passenger car in 1950. The Chrysler Corporation has also been seriously pursuing development of passenger car gas turbines since 1950, and tried out a prototype 50-car fleet on the public in the 1964-1966 time period. General Motors announced its first automotive turbine in 1954 and unveiled its experimental Firebird II regener-

*Numbers in parentheses designate References at the end of paper.

ative GTE sedan in 1955. Ford, Williams Research, Volkswagen, and others have likewise embarked upon turbine passenger car development programs. None of these programs have, as yet, resulted in a mass-production version of the automotive gas turbine."

U. S. manufacturers have been developing gas turbine propulsion systems for many applications. Most recently, the Chrysler Corporation has been working with the government to develop the gas turbine engine and car shown in Fig. 1. Similarly, Detroit Diesel Allison (DDA) Division of General Motors is conducting a performance improvement program with its heavy duty gas turbine engine installed in Greyhound buses and tractor trailer trucks. A cutaway view of the DDA heavy duty engine is shown in Fig. 2. The Ford Motor Co. conducted a pilot production program in the early 1970's, installing their heavy duty gas turbine engine in truck and marine applications. The engine was scheduled for production in the mid-1970's, but regenerator problem deferred the decision.

Each of these programs has contributed toward defining the current state-of-the-art of the automotive gas turbine engine. Currently these gas turbine engines are operating at turbine inlet temperatures of approximately 1310 K (1900° F), and are basically constructed of metallic components (i.e., turbine, combustor, etc.) with the exception of ceramics being used in some regenerator applications.

In response to the national goal to reduce the consumption of fuels derived from crude oil and to meet the federal emission standards, the Division of Transportation Energy Conservation of the Department of Energy (DOE) is sponsoring a program to develop and demonstrate fuel efficient gas turbine powered highway vehicles. This program is being carried out by the Government with the major U. S. automobile manufacturers. Program management is under the direction of DOE. "Project management" responsibility for the various elements of the program has been delegated to LeRC.

Although the DOE program considers the entire propulsion system (engine and transmission) this paper will address only the engine related technology. Specifically, the areas to be discussed in the paper are: steady state and transient systems analysis, compressors and turbines, combustors, and materials and coatings. Papers covering seals, bearings, and instrumentation technology developed in the LeRC managed programs are being presented in separate papers during this session.

The following individuals contributed to this paper: John Klann and John Zeller, System Performance Analysis Techniques; Robert Wong and Arthur Glassman, Compressors and Turbines; David Anderson, Albert Juhasz, Donald Schultz, and Richard Niedzwiecki, Combustors; and Stanley Levine, Curt Liebert, and Richard Ashbrook, Coatings and Materials.

DESCRIPTION OF THE AUTOMOTIVE GAS TURBINE ENGINE

The gas turbine operates on a Brayton thermodynamic cycle. The design efficiency of this cycle is basically controlled by the operating temperatures of the cycle and the efficiency of the components used to mechanize the cycle. Typically, a gas turbine propulsion system is well suited for applications where the engine spends a large amount of time operating at or near 100% of its design speed and design operating temperature (i.e., engines for subsonic aircraft, auxiliary power units, etc.). The automotive application, however, imposes additional requirements on gas turbine propulsion system which must be met if the system is to be a viable alternative. The basic requirements are:

- Fuel efficient operation over a duty cycle that requires significant operation at off-design part speed and part power conditions.
- Minimum impact of small-scale effects on component efficiency.
- Competitive manufacturing, initial and life cycle costs.
- Acceptable acceleration.
- Environmental acceptability (emissions and noise).
- Acceptable reliability and safety.
- Acceptable volume and weight.
- Multi-fuel capability.

The gas turbine system has shown a good potential for meeting these automotive requirements.

In the ideal Brayton cycle, the working fluid (air) is compressed from ambient conditions. Energy is added to the working fluid in the form of heat, and energy is extracted from the working fluid in an expansion process. The cycle can be operated with a number of variations. For an automotive application, however, an open regenerated cycle using rotating turbomachinery is commonly of most interest. The components used in this case are a compressor, combustor, heat exchanger, and turbine. Multiple options exists in the type of components used to mechanize the cycle and the configuration in which these components are arranged. The primary options available are:

- Compressor - axial or centrifugal
  - single or multiple stages
  - fixed or variable geometry
- Heat Exchanger - stationary recuptera-tor
  - rotating regenerator
- Turbine - radial or axial
  - single c: multiple stages
  - fixed or variable geometry
- Engine Shaft Arrangements - single or multiple

Engine Configurations - The arrangement of engine components into single or multiple shaft configurations is a complex problem with trade-offs required between cost, performance, transmission type, and engine size. In the single shaft engine, the compressor and power turbine are mounted on a common shaft and therefore must operate at the same speed. This common shaft is then directly coupled to a transmission. While this engine configuration is relatively simple, it requires a rather complicated continuously variable transmission (CVT) to operate over the speed range required for the automotive application. A single shaft configuration that was the subject of study in (2) is shown schematically in Fig. 3.

In the multiple shaft configuration, referred to as a free turbine engine, the compressor and compressor drive turbine are mounted on a common shaft. This portion of the assembly including the combustor, often referred to as the gasifier assembly, is followed by the power turbine (free turbine) which is mounted on a second shaft. In this configuration the gasifier turbine is coupled aerodynamically to the power turbine and the power turbine is coupled to the transmission. This configuration can be operated over the automotive duty cycle using a conventional three-speed transmission. A schematic of a two shaft engine (Chrysler) is presented in Fig. 4. The state properties shown in the figure are representative of the operating conditions for this engine. Turbine inlet temperatures are limited to approximately 1325 K (1925° F), the compressor pressure ratio is approximately 4:1, and the regenerator operates to a temperature of approximately 1017 K (1370° F).

Another multiple shaft gas turbine configuration is the three-shaft engine discussed in (3) and shown schematically in Fig. 5. This configurations key feature is the use of a third turbine on a separate shaft that can assist in driving the compressor, the vehicle, and the accessories. The use of the third turbine according to the reference permits close tailoring of the engines performance and operating characteristics to the requirements of the automotive duty cycle. While the third shaft adds complexity to the engine, the transmission requirements can be satisfied with a relatively simple planetary gear box.

There are many variations possible in the gas turbine engine. However, the three briefly described here, the single, two- and three-shaft configurations are representative of the primary variations.

Future Requirements - Today with the heavy emphasis on fuel economy, the primary performance targets for the advanced automotive gas turbine can be summarized as:

(1) Increases in turbine inlet temperature to the 1645 K (2500° F) range.

(2) Improved part power performance.

These performance factors, along with the manufacturing cost, must be considered in the development of these engines. The high level of turbine inlet temperature places requirements on the hot section components well beyond the technology in current automotive gas turbine engines where these components are made of metal alloys. Because component cooling is not currently being considered because of its impact on cost, the higher temperature requirements will require development of ceramic materials for these components. A preliminary analysis of the potential benefits and problems associated with the introduction of ceramics to an advanced gas turbine engine is presented in (4).

The second target of good part power performance will require development of the off-design performance capability of the turbomachinery. One of the factors being considered for increased part power performance is the addition of variable geometry to the turbomachinery. Another factor required is higher operating temperature heat exchangers, which is another area where high temperature ceramic materials are being considered.

APPLICABLE AEROSPACE TECHNOLOGY

The following sections present a summary of the turbine engine technology that has been developed through aerospace-related programs that have the potential to help meet the goals and requirements for the automotive gas turbine engine.

SYSTEM PERFORMANCE ANALYSIS TECHNIQUES - Both steady-state and transient performance predictions programs and analysis techniques have been developed and used extensively at LeRC as an effective tool to analytically determine or evaluate the performance characteristics of various gas turbine systems. They are described as follows:

Steady-State Performance Program - As discussed previously, there are many possible gas turbine configurations that could be beneficial in an automotive application. In order to identify these and determine the most promising configurations, each one must be evaluated on its own merit. One of the primary considerations is the fuel economy potential when operating at design and off-design conditions over a driving or duty cycle. To perform this task, a generalized computer code which was developed for analysis of aircraft gas turbines has been modified for automotive use. The code is referred to as the NAVY/NASA Engine Program (NNEP), and is described in (5). It has evolved from a series of previous Government and industry-developed aircraft gas turbine computer codes into a versatile and advanced assessment tool. The current version of NNEP is the result of a joint effort of the Naval Air Development Center and LeRC, and is a nonproprietary code that is freely available to the industry. With some modifications and additions, NNEP is being applied at LeRC to automotive gas-turbine analysis.

Since NNEP is an outgrowth of many previous gas turbine computer codes, it is a highly sophisticated and generalized tool. Through data input statements, the user builds the engine configuration by numbering and naming the type of components, defining the component performance characteristics, indicating component interconnections, and defining the engine design conditions. Each initial calculation in NNEP is for an engine design point condition. Furthermore, NNEP allows more than one engine arrangement to be specified at a time. Thus, while running the code, flow paths or mechanical arrangements can be switched to simulate variable cycle engines (for aircraft engine analysis) or power transfer among shafts (for automotive engine analysis).

The types of flow and mechanical component available in NNEP include: inlets, ducts, compressors, combustors, turbines, heat exchangers, water injectors, flow splitters, flow mixers, nozzles, shafts, and loads. Performance of each of these components is handled in separate NNEP subroutines. The compressor, duct, and turbine subroutines allow the specification of engine bleed flows. The turbine subroutine also has built-in cooling options.

Additional NNEP inputs include controls and/or optimization variables. Controls are used to balance (or unbalance) engine flow conditions and/or specify the desired engine operating conditions. The optimization variables can be assigned to maximize or minimize engine performance parameters at design or off-design operating conditions. The code is limited to a maximum of 60 total components, controls, and optimization variables. As a result, nearly any type of gas-turbine configuration and its operational controls can be synthesized in NNEP.

Off-design component performance is described in NNEP through the use of input tables or maps. The tables are normalized to the component design conditions. NNEP can accommodate three-dimensional component maps. For example, a configuration with both variable turbine and compressor geometry could be synthesized, and its variable geometry setting could be optimized for fuel consumption at each engine operating speed.

Recent additions to the NNEP are a component sizing and a weight subroutine. These subroutines, constructed from aircraft design procedures, have demonstrated accuracies within ±10 percent. Unfortunately, these routines are not currently applicable to automotive gas turbine analysis.

There have been two additions to NNEP's basic capabilities to facilitate automotive gas turbine analysis. These additions include subroutines for generating preliminary compressor and turbine (turbomachinery) design characteristics, and for outputting engine performance maps for use with a Driving-Cycle Analysis computer code. The preliminary turbomachinery design subroutines are based on synopses of models for radial-flow compressors, radial-flow turbines, and axial-flow turbines. Use of these subroutines in NNEP produces design-point component efficiency predictions, rotor dimensions, and mean-section velocity diagrams. Thus, turbomachinery design tradeoffs and their effects on engine performance can be investigated directly rather than on a parametric basis.

The subroutine which has been added for outputting engine performance maps allows two transmission options which affect the form of outputting. Either a continuously-variable speed-ratio (CVT) or three-speed transmission may be specified. With the CVT option, the engine performance map is outputted as a single operating line. The three-speed option results in a complete matrix of engine operating conditions.

One example of the intended use for NNEP in automotive analysis is a current effort aimed at making a preliminary definition of an Advanced Gas Turbine Powertrain. The fuel economy potentials of a matrix of Advanced Gas Turbine configurations is being screened with NNEP in combination with the Driving Cycle Code. Multiple-shaft arrangements are being studied with both the CVT and three-speed transmissions. Single-shaft configurations are being studied with a CVT.

For each configuration and set of operating constraints, NNEP is being used to examine engine performance as a function of design point compressor pressure ratio. Results from NNEP are then being used in the Driving Cycle code to find the best design conditions for vehicle fuel economy. Comparisons among the screening study configurations will be used to identify promising arrangements to be carried into more detailed conceptual design studies.

Transient Performance Prediction and Analysis - In addition to steady-state turbine engine performance, transient engine performance is of importance to overall propulsive system operation. Transient conditions exist when a change in output power is deliberately requested or when external engine conditions change. Whether a turbine engine is being used in an aircraft application or an automotive application, a knowledge of its transient performance is necessary. The manner in which a turbine engine will perform transiently is today dictated by a closed loop (feedback) control system. The controller must regulate performance at an output power condition and be able to quickly and predictably take it from one operating point to another. It must do this while avoiding conditions of overspeed, turbine overtemperature, combustor blow-out, etc.

To understand the dynamic interactions of aircraft turbine engine system components, including the controller, and to assist in designing a control strategy which will guarantee some specified performance, computerized simulations of this complex system are being used. These simulations are detailed accurate analytical representations or models of the engine systems. These simulations have been accomplished using both digital and hybrid (analog and digital) computers. Digital simulations of the transient and steady-state characteristics of complex systems are useful when the simulation must be utilized by several different organizations. Universal software languages (FORTRAN and CSMP) enable the simulation to operate on different types of computing equipment. Digital simulations of engines, however, do not normally run in real-time without major simplifications. Hybrid engine simulations, however, can operate in real time (6 to 8). The value of a hybrid simulation for the development of controllers for advanced technology turbine engines will be the subject of the next few paragraphs.

The trend in controllers for aircraft turbine engines is toward an electronic digital computing device. This trend is due to the high computational requirements of control laws for advanced engines and the potential cost benefits such a device may yield. A digital computer controller is a sampled-data controller (control update inputs at distinct intervals of time after sampling new performance information) and relies on a stored software program to accomplish its control action. Some method of debugging, refining as necessary, and finalizing this control software is required. In addition, it is valuable to evaluate this software in real time since a closed loop control is sensitive to timing considerations, especially when the control is designed to exercise various priority levels of protection and regulation.

Real-time hybrid computer simulations of the engine process as an approach to controller design and evaluation has been used in two successful programs (9 and 10). Figure 6 shows how such simulation capabilities are utilized. The hybrid computing system models all aspects of the propulsion process including the actuators and sensors.

Figure 7 is a partial description of how the component characteristics (compressor maps, burner efficiency, etc.) are connected to generate a transient engine simulation representative of engine performance characteristics. When a real-time simulation has been designed, its quality is verified by comparing engine transient performance information obtained from the simulation to actual data from the engine. Figures 8(a) and (b) are samples of such a verification for an F-100 aircraft turbofan engine simulation. Both curves show close correspondence between simulated and actual engine response to an instantaneous movement of the power lever from idle to intermediate power.

The control algorithms are programmed in fixed point assembly language (to accomplish the computing speed required) on a separate digital control computer representative of the capabilities that would eventually exist in the actual computer hardware controlling the real engine. After suitable refinement of the control algorithms and their software implementation, the control computer can be switched over to operate the actual engine hardware.

A development technique utilizing real time simulations can save much time and avoid the risks involved with developing control laws on the actual engine hardware. Engine running can be minimized or at least deferred until a high degree of confidence in the controller's ability to guarantee safe transient and steady-state engine operation has been achieved.

The application of electronic controls to advanced aircraft engines is very close to reality. Automotive turbines will have the same difficult control task, and a digital electronic control can help satisfy the stringent control requirements. The use, therefore, of simulations and especially real-time simulations to develop the digital computer control algorithms would prove quite beneficial. When improvements in the engine components are required, or the need for control refinements arises, a simulation can effectively and efficiently be used to determine the effects in engine performance which can be expected. Sensitivity studies of component production tolerances can also be accomplished through parametric studies with these simulations.

COMPRESSOR AND TURBINES - Advances in the technology of turbomachinery have been directed toward developing a better understanding of the fundamental flow processes that occur in compressors and turbines from which improved design methods and performance prediction techniques have been developed. The scope of these activities has included both the areas of basic experimental and analytical research, as well as full scale rig and engine tests of compressors and turbines for various aerospace and, more recently, automotive applications. Those aspects of the work which are felt to be most applicable to the automotive gas turbine engine are described in the following sections.

Performance Prediction Programs - The ability to predict analytically the design point performance for a centrifugal compressor, radial turbine, or axial flow turbine has been well established. The methods and computer programs developed at LeRC and through LeRC sponsorship are described in (11 to 16). With appropriate design input information such as rotative speed, flow, inlet state conditions, and power or pressure ratio, the programs can be used to calculate and/or optimize the design point efficiency, the number of stages, the dimensions of the flow path annulus, and the velocity diagrams. For axial flow turbines, the program of (11) is simplified to calculate the flow conditions at the mean blade height only. Also, for any given turbine, all stages have the same shape of diagram, with the shape depending on the stage work factor and degree of reaction selected.

The design point program of (12) calculates the radial variations of flow angle and velocity from blade hub to tip. Both free vortex and nonfree vortex designs can be generated or analyzed. Loss coefficients are either input to the programs or calculated internally to the program by the method described in (13).

Similar methods and programs for calculating the design point performance for centrifugal compressors and radial inflow turbines have also been developed, and are described in (14 to 16).

The need for calculating the off-design performance characteristics of these components over their potential operating range was noted previously in the section titled System Performance Analysis Techniques. Successful methods have been developed and coded for axial and radial turbines, and they are described in (17 and 18), respectively. The radial turbine program is based on a meanline analysis of the flow and is for a single-stage turbine. The axial turbine program is applicable to turbines having up to eight stages. The program can be run as a meanline analysis, or can allow for radial variations in loss and flow conditions. Two loss options are provided: a kinetic energy coefficient-inlet recovery coefficient method, and a total pressure loss coefficient method. The analysis is applicable from zero to design speed, and the work done may vary up to the maximum as limited by discharge annulus area choking.

The value of an off-design performance program depends upon its ability to accurately predict performance over a wide range of conditions. An experimentally determined performance map, as reported in (19) for a single-stage axial flow turbine, is shown in Fig. 9. Data were obtained over a range of speed from 40 to 100 percent of design, and for pressure ratios of 1.4 to 2.0. The program of (12) (with coefficien selected to match the design-point performance) was used to predict the turbine work and flow for the same range of conditions. Over the entire map, the predicted performance was within 1 percent of the experimentally obtained values. Thus, the validity of the program was demonstrated. It should be noted, however, that such good agreement may not be experienced for all turbines.

The methods being developed to predict

the off-design performance of centrifugal compressors, such as the method described in (20), are still evolving. The efficiency variations predicted by the method of (20) have not proven to be realistic as yet, and experimentally-obtained values must be used.

Blade Design Programs - In order to minimize blade losses, it is necessary to calculate accurately the flow conditions for a given geometry so that the flow distribution throughout the blade passage can be controlled to avoid conditions associated with high losses. A typical flow passage for an axial and radial flow rotor is shown in Figs. 10 and 11, respectively. Gradients in velocity occur across the passage from blade-to-blade and from hub-to-tip, Fig. 10, or hub-to-shroud, Fig. 11, as the result of gradients in static pressure necessary to turn the flow or satisfy radial equilibrium requirements. In order to calculate the flow field in these passages, quasi-three-dimensional analysis techniques using meridional plane solutions together with blade-to-blade plane solutions have been developed. Two basic computation methods are employed: a stream-function method covering the entire blade passage, and a velocity-gradient method covering the guided or covered portion of the channel passage. The programs are described in (21 and 22), respectively. An example of the measured vs. predicted velocity distribution using the first method is shown in Fig. 12(b) for the stator blade profile shown in Fig. 12(a). The predictions agree quite well over most of the blade surface.

In another program, described in (23), the flow in just the meridional (radial-axial) plane of an axial, mixed, or radial flow machine can be determined analytically. The program has been extended, (24) to allow solutions to be made in annular passages without blades. This program as well as the programs of (11 to 13, 17, and 19 to 23) were used to design the turbomachinery and ducting sections upstream and downstream of the power turbine of the Upgraded automotive gas turbine engine described in (25) and shown previously in Fig. 4.

Small-Scale Effects - Uppermost in importance in the application of turbomachinery to the automotive gas turbine engine is the potential effect of small scale on performance. In numerous programs conducted at

LeRC, the effects of Reynolds number and size on performance have been investigated for both axial and radial flow compressors and turbines. Some of the more significant results are summarized in the following paragraphs.

In (26) an extensive study was made to correlate the effects of turbine size and Reynolds number to turbine losses. The performance of a total f 19 single stage axial flow turbines, ranging in size from 10.2 to 35.6 cm (4 to 14 in.) in diameter, and from 10^4 to $2x10^6$ in Reynolds number were included in the study. The results indicated that there was an effect on performance due to Reynolds number below $2x10^5$. This is shown on Fig. 13 by the negative slope characteristics of the curves for turbines 3, 5, and 6. Above this Reynolds number, however, the effect was negligible. The results also showed a difference in the loss parameter between several of the turbines at the same Reynolds number, and this was attributed to geometric factors. Of the various factors considered, stator throat area appeared to correlate the loss parameter most closely, and this correlation is shown in Fig. 14.

Similar programs have also been conducted at LeRC on radial inflow turbines (27 and 28), where the effects of Reynolds number, diameter, shroud clearance, and rotor configuration on performance were investigated. The efficiencies were affected by Reynolds number variations over the range of values investigated, and these results are shown in Fig. 15. However, there was no significant difference in performance due to differences in rotor diameter between the three diameters investigated of 8.89, 11.68, and 15.29 cm (3.50, 4.59, and 6.02 in.). The performance was also generally insensitive to shroud clearances and blade loading over the radial portion of the rotor, but was found to be sensitive to the geometry of the rotor exducer and exit diffuser duct.

The effects of variations in Reynolds number on the performance of a small centrifugal and axial flow compressor has also been investigated and compared (29 and 30). Both compressors were designed for the same application, and are shown in Figs. 16 and 17. The centrifugal compressor had a tip diameter of 15.2 cm (6 in.). The axial flow compressor had six stages and a tip diameter of 9.4 cm (3.7 in.). A comparison of losses with percent change in inlet pressure (shown as percent of design Reynolds number in Fig. 18 for the two compressors), showed the centrifugal compressor to be less sensitive to reductions in Reynolds number than the axial flow compressor. If similar trends in performance occur with down-sizing, as might be required for an automotive application, the results suggest an axial flow compressor would suffer a larger reduction in efficiency than a centrifugal compressor.

Variable Geometry - The effect of varying turbine stator or compressor diffuser setting angles to extend the off-design operating range and efficiency of turbines and compressors has been explored in several programs. These investigations have been directed toward such aerospace applications as space power generation and jet engines for supersonic flight, but have also included the effects of variable geometry on automotive gas turbine compressor and turbine performance.

The results of a recent investigation of the variable geometry power turbine for the sixth-generation Chrysler Baseline engine, (31), are currently being evaluated. Reference (32) describes the measured performance, velocity diagrams, and exit diffuser characteristics of the turbine operating at its design stator chord setting angle of 35° from tangential. Initial results describing the performance at off-design stator setting angles are shown in Fig. 19. The sharp variation in stage efficiency with stator setting angle is evident. The effect of stator endwall clearance, necessary to allow for actuation of the stators, is also noted on the figure.

The flow mechanisms which cause this variation in efficiency have been the subject of numerous investigations. The results are summarized in (33) for a turbine having the same design pressure ratio, and stator and rotor reaction characteristics as the Chrysler Baseline power turbine. A cross-section of the stage is shown in Fig. 20. From the experimental results, a loss breakdown analysis was made which showed the relative effects of varying the stator angle on stator, rotor, and incidence losses. The results are shown in Fig. 21 for the three stator setting angles shown in Fig. 20 (design, 30% open, and 30% closed from design) which corresponds to an angle variation from design of approximately

-7° and +8°. The effect of varying the turbine pressure ratio on efficiency and losses is also shown on the figure. From the results, it is evident that considerable changes occur in the turbine when variable geometry is imposed. The results emphasize the need to determine these changes relative to the intended application such that the best design can ultimately be made.

COMBUSTORS - Ongoing programs in combustion research include the five basic areas of combustor aerodynamics, liner cooling, fuel preparation, alternate fuels, and exhaust emissions. Much of the effort has, of course, been directed toward the advancement of aircraft gas turbine combustion systems, but because of its fundamental nature, it is equally applicable to the automotive gas turbine combustor. In addition to this basic research, several exploratory investigations of various advanced combustor concepts have demonstrated performance characteristics which are felt to be particularly attractive to the automotive application.

Basic Research - Combustor aerodynamics are the aerodynamics associated with the design and analysis of the combustor geometry and its effect on air flow distribution, pressure losses, and exit temperature profiles. Through the use of a computer program developed under LeRC sponsorship (34), extensive analyses of this complex problem have been made. An example of the use of the program as a design or analysis tool is described in (35).

The efforts in liner cooling have been directed towards the development of analytical temperature prediction programs (36), and improved film cooling techniques. A perforated-sheet liner has been developed which is one of the simplest and lowest cost film cooled liners to fabricate. It has been tested in a turbojet combustor rig and compared to a conventional stepped-slot film cooled liner (37). No adverse effects were found using the perforated-sheet liner relative to its film cooling effectiveness, combustion efficiency, pressure drop, or exit temperature profile.

Investigations into the design and performance of various advanced fuel injection and fuel preparation techniques have been conducted and extensively documented. Two injection techniques, the air-blast and air-assist nozzles, (38 and 39) have been found to have superior emission characteristics compared to more conventional fuel injection techniques. Fuel premixing and prevaporizing techniques have been investigated for various advanced combustor concepts such as the catalytic combustor described in the next section. Fundamental investigations of premixing-prevaporizing combustors (40 to 42), have established that this system can achieve very low NO_x levels, Fig. 22. Also, the use of fuel staging between two or more injectors has been effective for controlling the performance and emissions of a combustor over the throttlable range of an engine.

Efforts at LeRC or under LeRC sponsorship on alternate fuels range from the study of the technical problems of converting shale oil or coal-derived syncrudes to turbine fuel, to evaluations of the thermal stability and testing of these as well as other alternate fuels in combustor rigs and engines. Comprehensive studies of the refining requirements and thermal stability characteristics that have been made to date are given in (43 and 44). The comparative results of running a combustor on Jet A and Diesel number 2 fuels (45) showed comparable levels of NO_x and CO emissions between the two fuels, but approximately twice the unburned hydrocarbons and smoke number with Diesel number 2.

The initial emphasis in the emission investigations was to develop and demonstrate the technology required to reduce the HC, CO, and NO_x pollutants in current and future aircraft engines. The results of a portion of this broad based program are summarized in (46). Other areas of emissions investigations have included the effects of water injection (47), exhaust gas recirculation (48) heat pipe regenerators (49), the effect of fuel temperature on NO_x formation (50) and exhaust odors, (51). Referring to this latter reference, the recent results from a series of combustor rig tests run with five different fuels showed that odor intensities were related to the concentration of oxygenates in the exhaust, Fig. 23, which were in turn proportional to the efficiency of the combustion process, regardless of the fuel used.

Advanced Concepts - The high inlet and outlet combustor temperatures required for advanced aircraft and regenerative automotive gas turbine engines have resulted in both favorable and unfavorable effects on the combustion process. The high temperatures are conducive to high combustion efficiencies and hence low hydrocarbons and carbon monoxide emissions. However, the high temperatures also increase the formation of NO_x, and several unique and advanced combustor concepts have been under investigation to minimize these NO_x formations.

The principal of the LeRC developed swirl-can combustor for aircraft application, shown in Fig. 24(a), is to divide the flame zone into many small zones through the use of many small combustor modules, called swirl-cans. This approach provides leaner combustion through premixing, and reduced hot gas residence times. Typical results have shown a 30 to 35 percent reduction in the NO_x emissions, Fig. 25, compared to conventional combustors operating at conditions representative of commercial jet aircraft engines.

A second concept, premixed-prevaporized combustion which was noted in the previous section, atomizes, vaporizes, and mixes the fuel and air to allow combustion to take place at leaner fuel-air ratios. The predicted NO_x emission levels for this type of combustor for a typical current and projected future automotive gas turbine engine are shown in Figs. 26(a) and (b). One application of this concept is currently under development for the Chrysler Upgraded Gas Turbine engine, Fig. 24(b).

Two additional concepts under investigation are the catalytic combustor and the multi-element combustor, shown schematically in Figs. 24(c) and (d). These two combustor concepts are described in more detail in the following sections.

Catalytic Combustor - The combustion process in this concept is heterogeneous and requires a prevaporized, premixed fuel-air preparation system as shown enlarged in Fig. 27. The mixture is then reacted in a catalyst bed where the catalyzed surface of the bed allows combustion to take place at equivalence ratios well below the lean flammability limits (52). The maximum combustion temperature reached during the reaction process is only slightly above the turbine inlet temperature. As a result, low NO_x is possible with high combustion efficiency. Preheating of the bed during cold starting is required to initiate the process for the catalysts investigated thus far.

The concept has been under experimental investigation in combustion rigs for several years. Most of the catalyst test elements consisted of a cylindrically-shaped block of ceramic honeycomb material coated with a noble metal catalyst. The results obtained thus far are presented in (52 and 53). Based on these results, the predicted NO_x emission levels, as shown in Fig. 26 for the automotive application, are considerably below either the goal levels or the premixed-prevaporized combustor levels.

Multi-Element Combustor - The principal of this concept is to reduce the formation of NO_x by removing sufficient temperature from the flame to prevent the adiabatic flame temperature from being reached, and by increasing the hot gas velocities through the combustor by the use of turbulent flame holding to reduce the flame residence time. The concept is shown schematically in Fig. 28(a) for a single combustor element. Unmixed fuel and air enters a round stepped combustor passage located adjacent to a dilution air passage intrigally cast in a ceramic block. Premixing occurs at the inlet of the combustor passage, and homogeneous combustion occurs in the downstream portion of the passage. The steps provide high turbulence and good flame holding characteristics. Heat is transferred from the flame to the surface walls of the combustor passage, and hence to the dilution air through the ceramic interface between the two passages. A photo of the flame in an experimental single-element combustor is shown in Fig. 28(b). A complete combustor would consist of multiple elements of combustor passages interspersed between dilution air passages as shown in Fig. 24(d). As engine power or speed is varied, fuel to the individual combustor passages is staged or regulated to vary the overall fuel-air ratio of the combustor. Emission measurements from the single element tests, extrapolated to a multi-element combustor sized for the automotive application is shown in Fig. 26. Compared to the premixed-prevaporized and catalytic combustors, the results are very encouraging thus far.

COATINGS AND MATERIALS - Ceramic Thermal Barrier Coatings - A ceramic coating system has been evolved for cooled metal engine components such as rocket nozzles, combustor liners, turbine blades and vanes. The coating serves to insulate the metal parts from the hot combustion gasses through its low thermal conductivity (approximately 1.5 W/m K) and high reflectivity (from two to four times that of a typical high temperature metal alloy). As a result, the metal temperature of the cooled coated parts, or the amount of cooling air required to cool the parts is reduced.

The coating concept is illustrated in Fig. 29 for a typical cooled metal surface. Both a metallic bond coat and a ceramic insulating layer are applied to the surface by plasma spraying. The most recent and successful coating system consists of an insulating layer of yttria stabilized zirconia ceramic (ZrO_2-$12Y_2O_3$) 0.025 to 0.076 cm (0.01 to 0.03 in.) thick applied over a nickel-chromium-aluminum yttrium alloy (NiCrAlY) base coat 0.013 cm (0.005 in.) thick.

A paper given at this conference last year (55), described the system in detail including its performance and durability in tests on turbine blades, vanes, and combustor liners. Because of considerations of small size and low cost for an automotive gas turbine application, the use of the concept may be limited to such components as the combustor liner, its interconnecting duct to the turbine, and vane shrouds. The coating is currently being considered for the vane shrouds and portions of the combustor for an army tank engine.

In rig tests of a combustor from a commercial jet engine, Fig. 30 and (56), a coated liner operated at metal temperatures 130 to 210 K (230° to 380° F) cooler than an uncoated liner over the operating range of the combustor. Smoke concentrations and soot were also reduced due to the coating reflecting a larger amount of the incident radiation back to the flame. The reflected energy reduced the amount of unburned hydrocarbons which could also be beneficial to increasing combustion efficiency and reducing emissions. This particular combustor, however, already had a high efficiency of 0.999, and no significant change in either the efficiency or emissions was observed.

<u>Ceramic Materials Development</u> - Critical to meeting the needs for fuel efficient high performance gas turbine engines of the future is the availability of high temperature low cost materials for the burner and turbine components. In the case of the automotive gas

turbine engine, this also includes the regenerator component. Ceramics offer the best potential for meeting these requirements because of their low cost and low density (projected to be about 1/10 the cost of superalloys and 1/3 the density), as well as their high strength at operating temperatures up to 1670 K (2600° F) and higher. For turbine blades where the primary stress results from centrifugal force, their low density and high strength-to-density ratio makes them particularly attractive. However, the lack of ductility and very low impact resistance of ceramics probably won't permit their use in hot section components of aircraft engines until the problem of impact failures of the ceramic parts and the effect on flight safety can be solved. There is, however, a much greater likelihood that they will see service in automotive turbines in the foreseeable future. As noted previously, ceramics are already in use in automotive regenerators, and also in experimental turbines where an engine demonstration of an all-ceramic turbine has recently been completed.

The key elements in the study and development of ceramic materials technology as it applies to the particular components of the engine are summarized in the following paragraphs. Currently the most promising structural ceramics appear to be Si_3N_4 and SiC. Extensive screening studies of 35 different ceramics in the LeRC Mach 1 burner rig, Fig. 31 and (57), have shown that Si_3N_4 and SiC based ceramics have the most favorable resistance to thermal shock. These ceramics are also much more oxidation resistant than superalloys. Figure 32 shows a ceramic and a cooled superalloy blade which had been subjected to identical exposures in the Mach 1 burner rig at 1470 K (2190° F). The ceramic blade shows little effect of the exposure, but the alloy blade is badly cracked and eroded.

The Si_3N_4 and SiC ceramics also have excellent high temperature creep rupture properties, as may be seen in Fig. 33. The figure compares commercially available hot pressed Si_3N_4 and experimental Si_3N_4 and SiC to the strongest known conventionally cast vane alloy, WAZ-16, and the oxide dispersion strengthened (ODS) alloys.

Combustor liners, turbine vanes, and interconnecting ducts which are relatively low stressed components will have to withstand peak cycle operating temperatures up to 1640 K (2500° F), as well as rapid changes in gas temperatures due to engine light-off and shut-down transients, and throttle excursions. To meet these requirements, parts such as stator vanes may typically have to operate at stresses up to 50 MN/m^2 (7000 psi), which is close to the current capability of the commercially available Si_3N_4 shown on Fig. 33. Rotor blades blades, which will operate at somewhat lower temperatures, may typically have to operate at stresses up to 207 MN/m^2 (30,000 psi) which is close to the current capability of the experimental SiC shown on the figure.

Work under LeRC sponsorship is currently directed towards improving the creep-rupture strength of Si_3N_4 by reducing the alkali metal, O_2, and densifying additives (58). Further improvements in strength and impact resistance, as will be required for turbine blade applications, are also being investigated through improved ceramic processing procedures, and the development of energy absorbing crushable surface layers (59). Large improvements in the creep rupture and life properties of Si_3N_4 have been achieved through the use of higher purity silicon powders, reduced Ca content, and the substitution of ZrO_2 for MgO as a densifying aid, Fig. 34 and (59). Current work is also showing (60) that Si_3N_4 can be densified to 95% of its theoretical value without the aid of additives through the use of hot isostatic pressing to pressures up to 276 MN/m^2 (40,000 psi). This pressure is 10 times the pressure necessary to achieve this level of density when additives are used. This should permit a reduction or elimination of the additives which reduce high temperature strength, and should lead to further improvements in the high temperature strength properties of the material.

Increasing the purity of α-phase Si_3N_4 powders (61) combined with the application of the crushable energy absorbing layers noted in (59), have provided an increase in impact strength. The use of a porous layer of reaction sintered Si_3N_4, the best all-around approach to date, has resulted in a ballistic impact resistance of 11.4 joules (8.4 ft lb), which is an increase of 6 over the unprotected Si_3N_4.

Investigations have been conducted into the problem of interfacing ceramics with metals (e.g., ceramic blades to metal disks).

Early work, (62 and 63) recognized the need to accommodate the lack of ductility of ceramics by employing generous radii and cushioning interfaces between the blades and disk to prevent stress concentrations, as well as to prevent chemical reactions between the two materials. Successful tests were conducted in aircraft engines up to full power without blade root failures. Interfaces made of a porous or screen material were particularly beneficial. Ductile sheet metal used as a compliant layer at the interface has been used in more recent investigations, (64) and Fig. 35, and have been adapted for use in the ARPA/NAVSEA-AiResearch Ceramic Gas Turbine Engine Demonstration Program.

An overall summary of the current status of advanced high temperature turbine materials, coatings, and technology relevant to the aircraft gas turbine engine is given in (65).

SUMMARY

The automotive gas turbine engine is an attractive alternative powerplant because of its low emissions and alternate or multi-fuel capability. With the performance improvements projected through the application of currently-evolving and advanced technology, it may become a competitive alternative for this application.

In this paper, the authors and contributors present a summary of the current developments and future requirements for the automotive gas turbine engine, as well as an overview of the aerospace-related research and technology developments in several of the areas which may have a significant impact on meeting these future requirements for the automotive application. The areas discussed are system analysis, compressors and turbines, combustors, coatings, and materials.

The NNEP steady-state system performance analysis model, and a transient performance simulation technique using a hybrid computer have been developed and used extensively at LeRC as an effective tool to analytically determine and evaluate various types of gas turbine systems. The NNEP program is currently being used in a screening study to predict the fuel economy potential for a matrix of advanced automotive gas turbine engine configurations.

Computer codes have also been developed

and used extensively at LeRC as well as by the gas turbine industry to predict and optimize the performance, aerodynamic requirements, and geometric configurations for axial and radial flow compressors and turbines. Good agreement between the analytically predicted and experimentally measured performance maps and blade surface velocity distributions have been obtained in most cases. The application of the various codes range from nonbladed annular passages to simplified mean-passage-height blade and stage analysis, to quasi-three-dimensional free-stream and blade channel flow analysis where radial variations in vorticity, work, and losses are accounted for.

Experimental evaluations of the performance and loss mechanisms that occur in compressors and turbines due to small scale and variable geometry effects emphasize the need to accurately predict and design for these effects to assure minimum impact on performance for the automotive application.

Research in fundamental combustor technology has lead to the development of: a computer code which can be used as a design or analytical tool to describe the aerodynamic performance of combustors; an efficient low-cost perforated sheet liner; fuel nozzle designs and fuel preparation techniques such as premixing and prevaporizing, which have achieved very low NO_x emission levels; and a broader technology base for low emission combustion fundamentals, and for the refining, characterization and emission levels of alternate fuels. The performance characteristics of several advanced combustor designs employing multiple swirl cuprs, catalyst, and multiple element concepts show considerable promise for meeting the performance and emission goals for advanced automotive applications.

The development and use of a ceramic thermal barrier coating on a combustor liner resulted in a large reduction in liner temperature, and exhibited the potential to reduce the NO_x emission levels for some applications.

Research efforts in structural ceramic materials have lead to improved processing, design techniques, and materials properties. These efforts have been directed toward the critical goal of fabricating hot section components of ceramic materials to meet the cost and performance requirements for advanced automotive gas turbine engines. Increases in the high temperature stress rupture properties have been made through improved material purities and improved densifying aids, or through the use of isostatic pressing pressures of up to 10 times the pressure necessary to achieve the same level of material density when densifying aids are used. The use of porous surface layers have demonstrated a sixfold increase in impact resistance over unprotected ceramic surfaces. Also, the use of compliant metal interface layers have been developed which reduce stress concentrations and chemical reactions between ceramic and metal parts.

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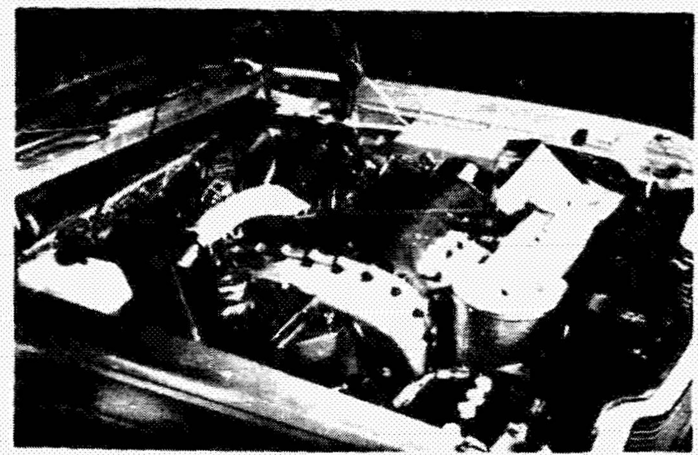

Figure 1. - Chrysler upgraded gas turbine engine installed in Aspen car.

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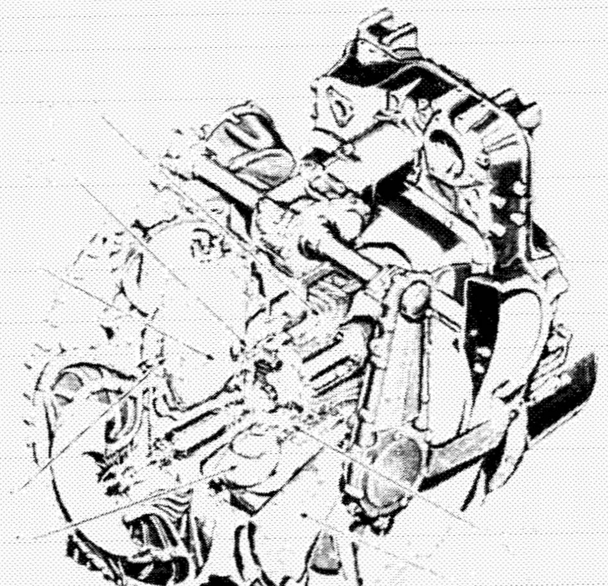

Figure 2. - Cutaway of DDA heavy duty gas turbine engine.

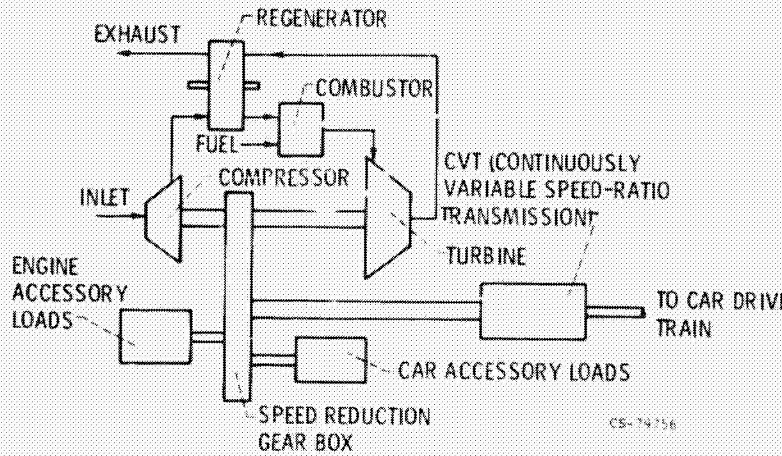

Figure 3. - Schematic diagram of a single shaft gas turbine engine/transmission arrangement.

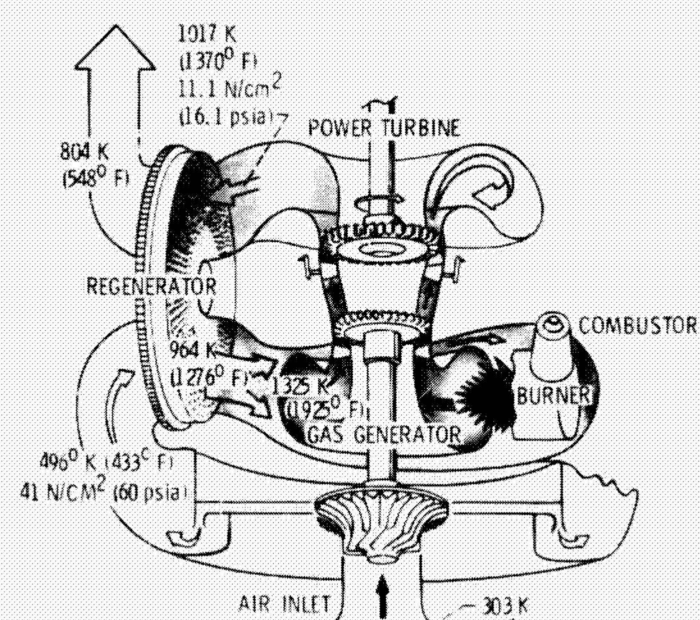

Figure 4. - Schematic of Chrysler upgraded engine.

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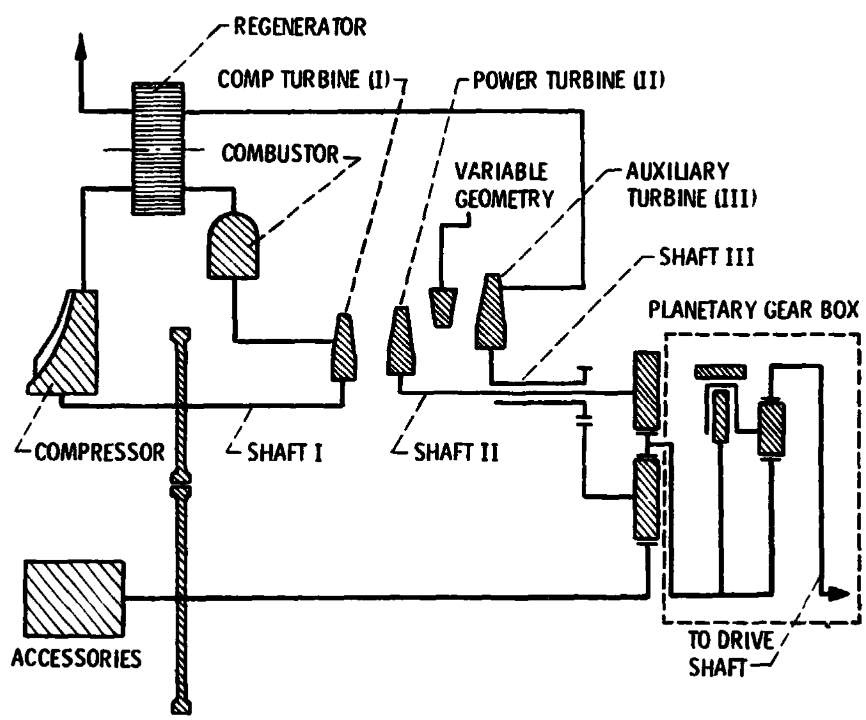

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Figure 5. - Three shaft turbine-transmission system.

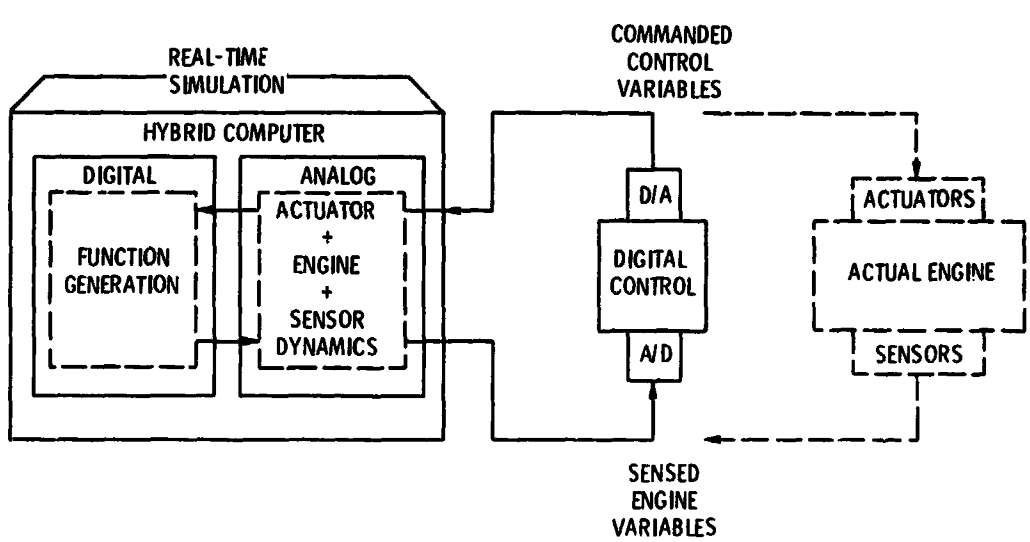


Figure 6. - Use of real-time, hybrid computer simulations for digital control development.

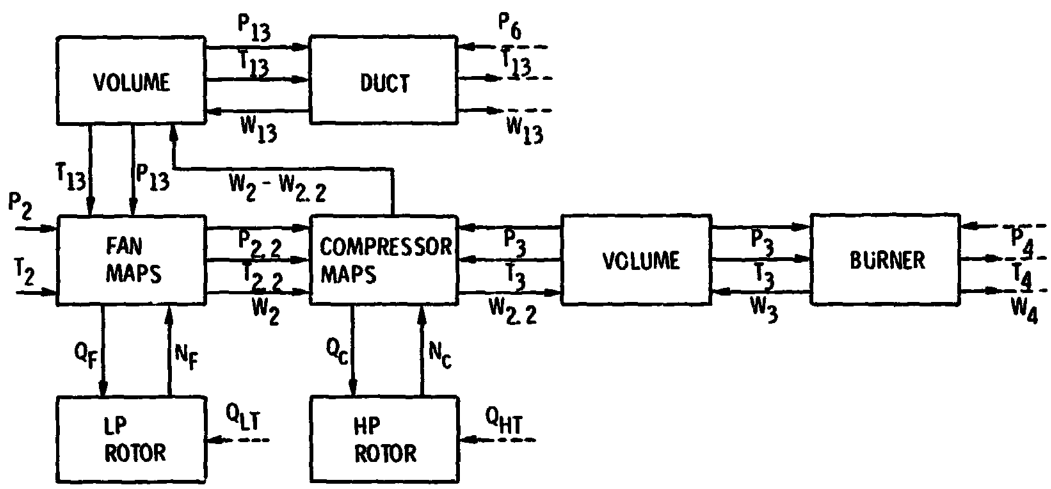
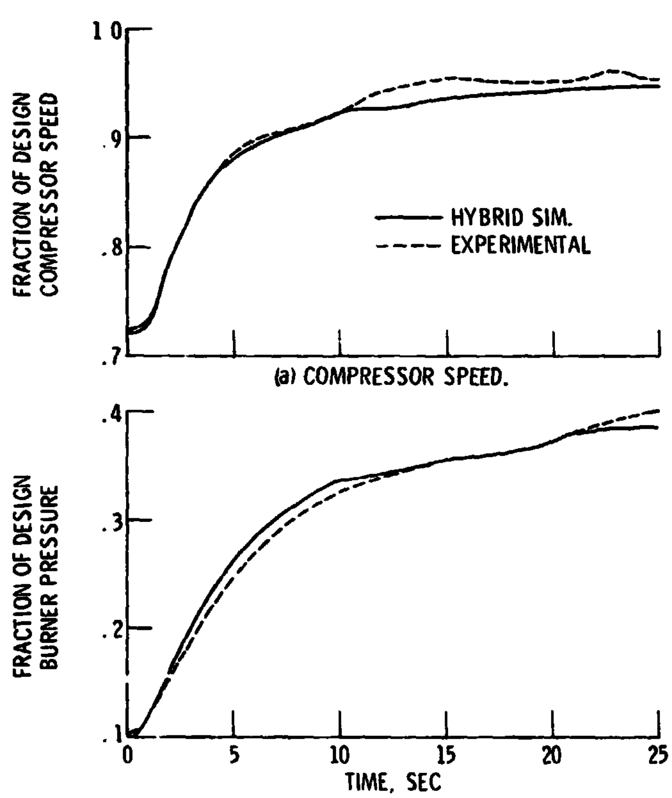


Figure 7. - Block diagram of F100 mathematical model.



(b) BURNER PRESSURE.

Figure 8. - Transient verification of F100 simulation. Idle-to-intermediate power lever slam. Altitude = 30 000 ft, Mach number = 0.7.

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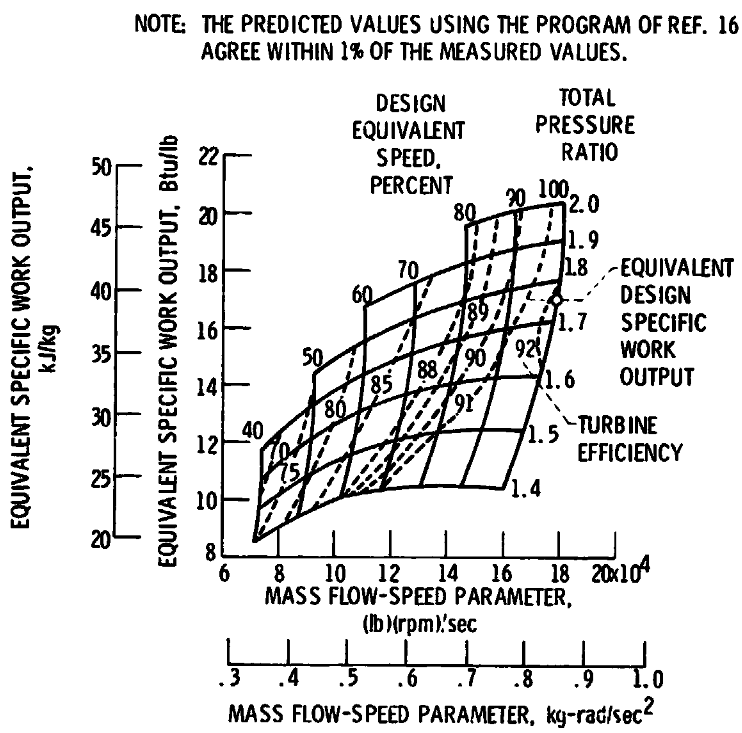


Figure 9. - Experimentally determined turbine performance map.

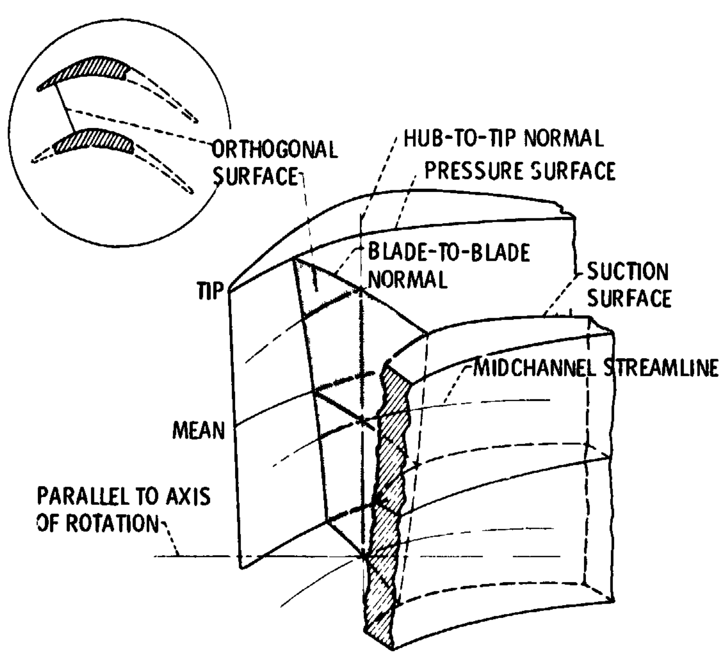
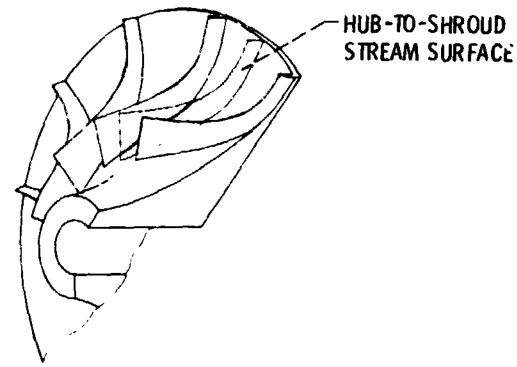
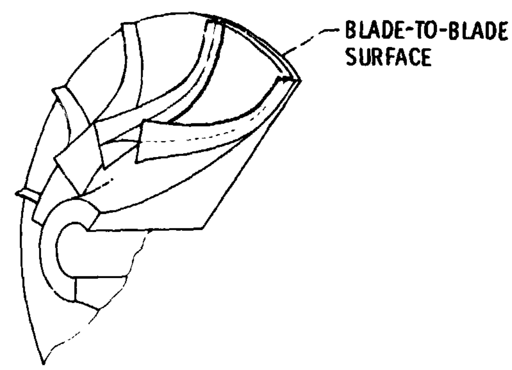


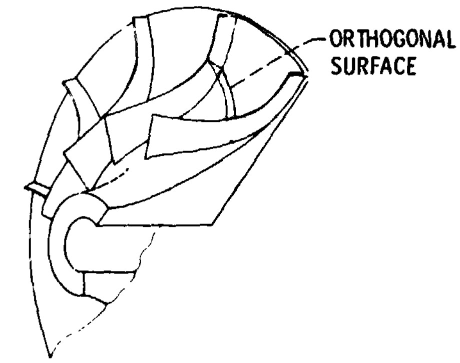
Figure 10. - Typical axial flow blade passage.



(a) HUB-TO-SHROUD STREAM SURFACE



(b) BLADE-TO-BLADE SURFACE OF REVOLUTION.



(c) ORTHOGONAL SURFACE ACROSS FLOW PASSAGE.

Figure 11. - Typical radial flow blade passages.

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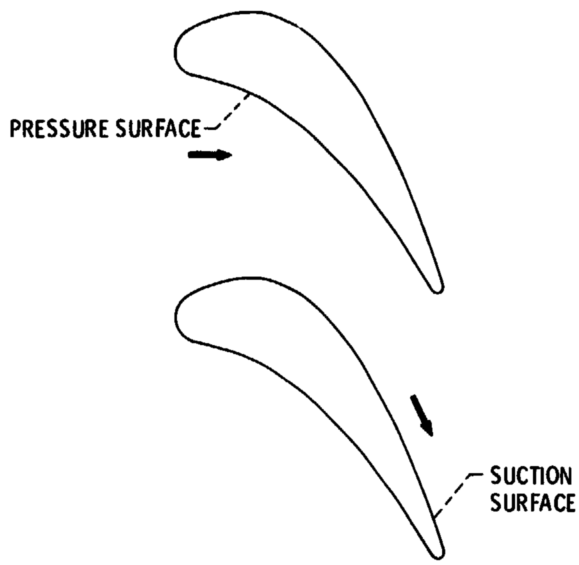
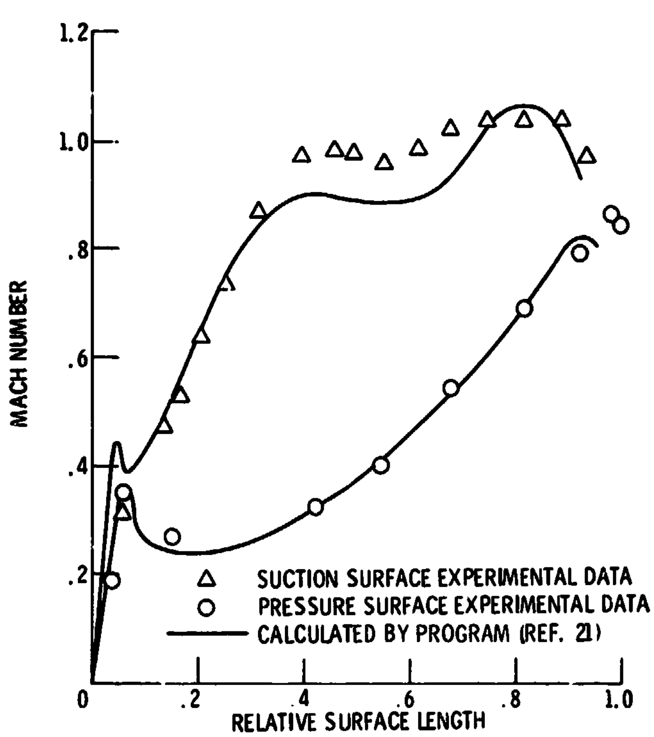


(a) BLADE AND CHANNEL PROFILES.



(b) SURFACE VELOCITIES.

Figure 12. - Comparison of experimental and computed surface velocities around stator blade.

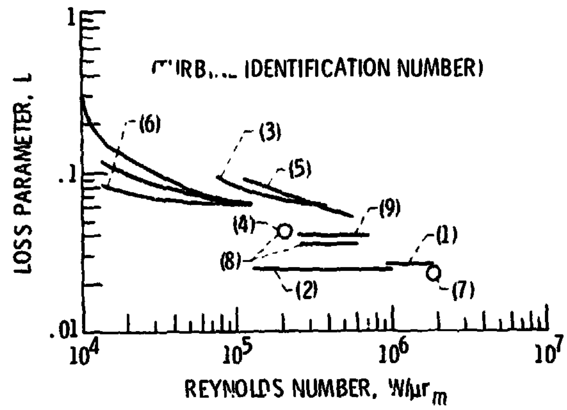


Figure 13. - Variation of loss parameter with Reynolds number.

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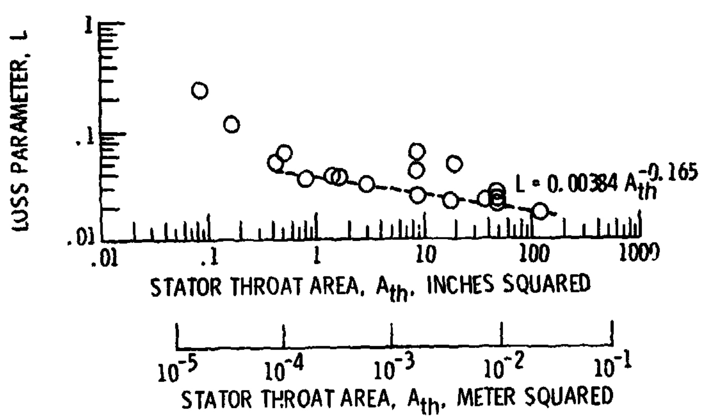

Figure 14. - Variation of loss parameter with stator throat area.

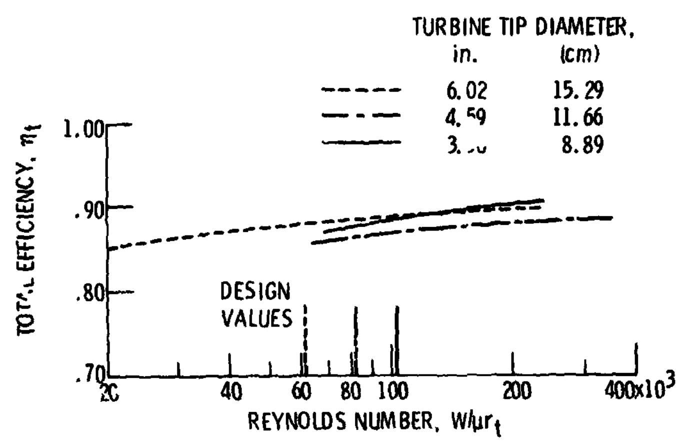

Figure 15. - Comparison of efficiency as a function of Reynolds number between three radial inflow turbines of differing tip diameter.

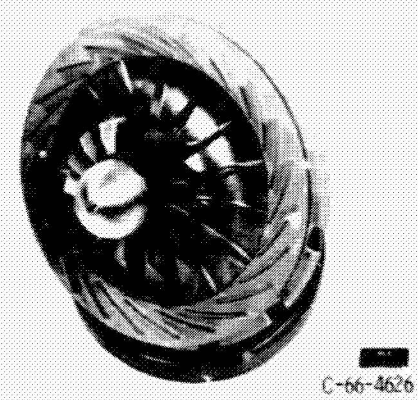

Figure 16. - Impeller and vaned diffuser for 15.2 cm (6 inch) centrifugal compressor.

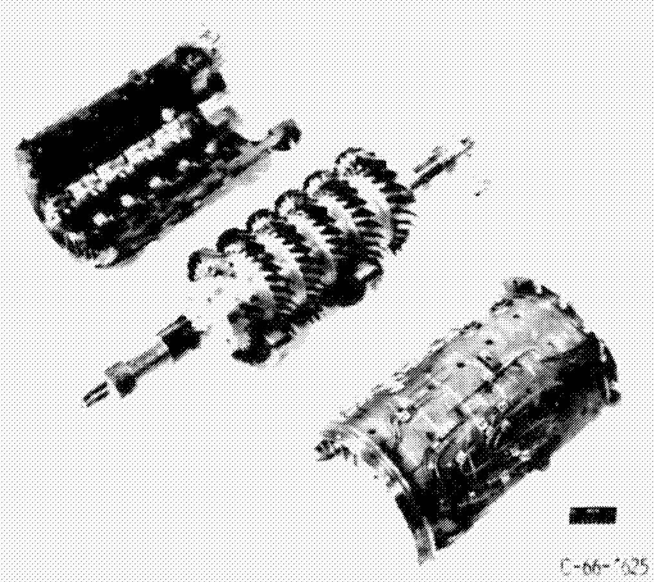

Figure 17 - Rotor and casing for 6-stage 9.4 cm (3.7 inch) diameter axial flow compressor.

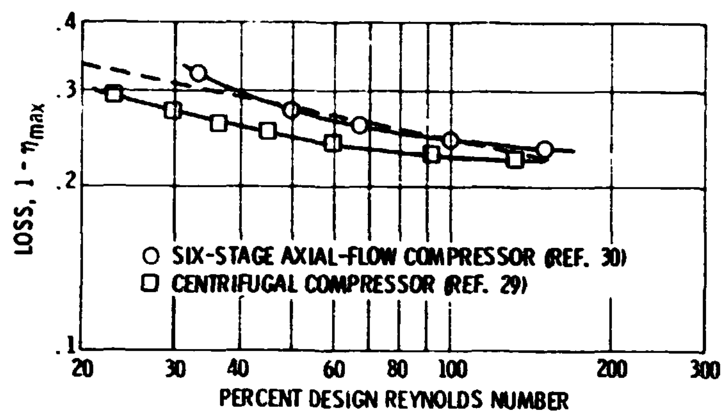

Figure 18. - Loss as function of percent design Reynolds number at design speed.

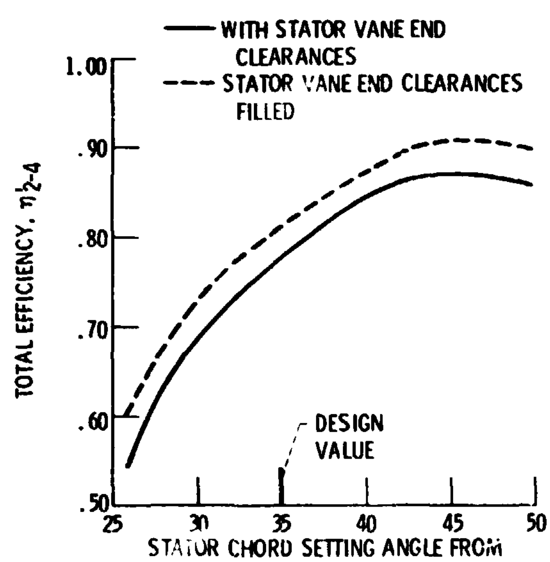

Figure 19. - Effect of stator setting angle and end clearance on Chrysler baseline power turbine performance.

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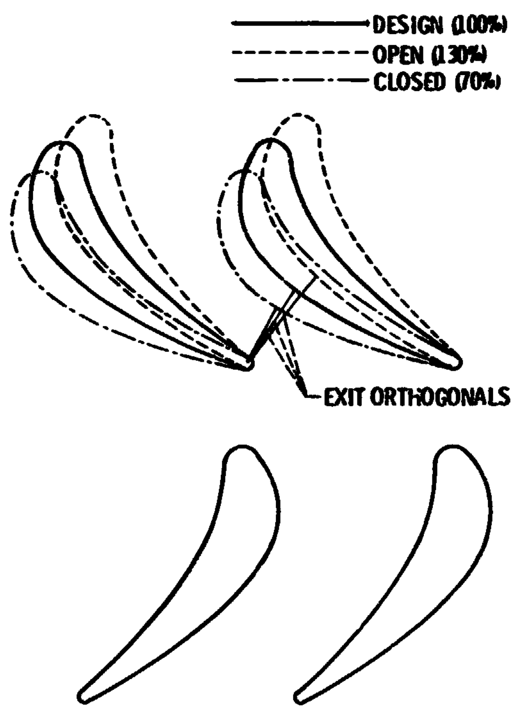

Figure 20. - Stator and rotor blade profile of LeRC variable geometry turbine, ref. 33.

E-9437

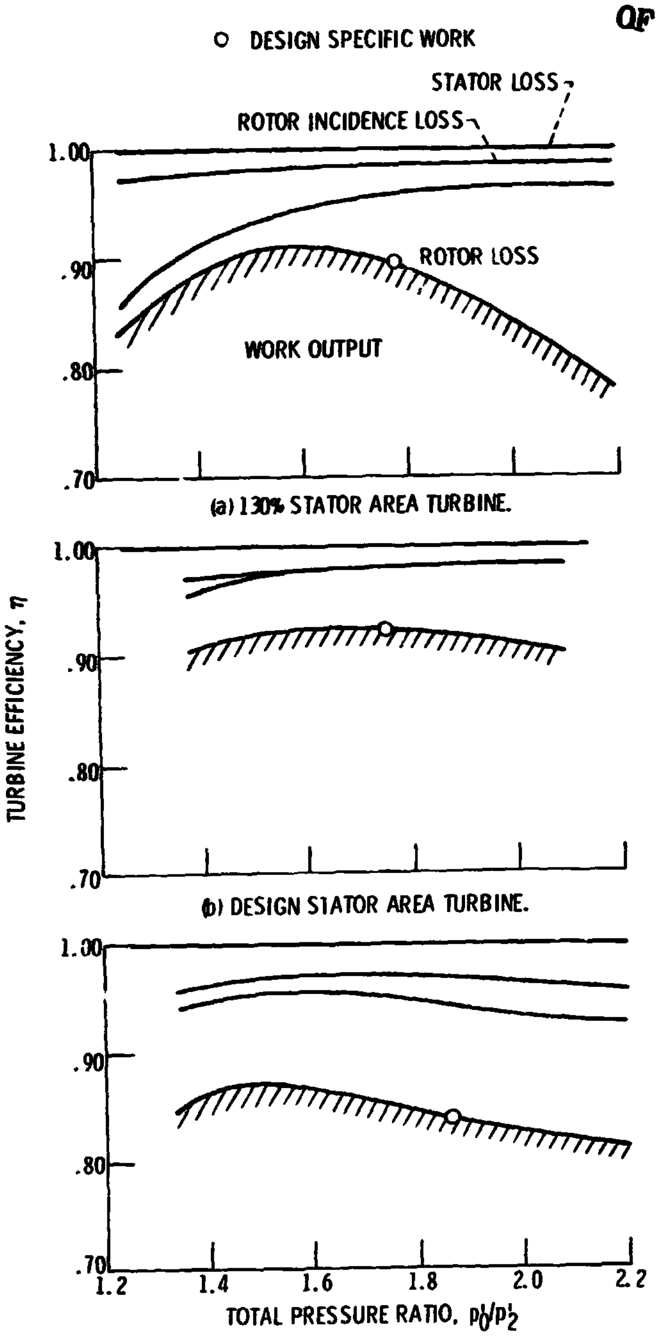

Figure 21. - Breakdown of turbine losses for three stator setting angles at design equivalent speed (ref. 33).

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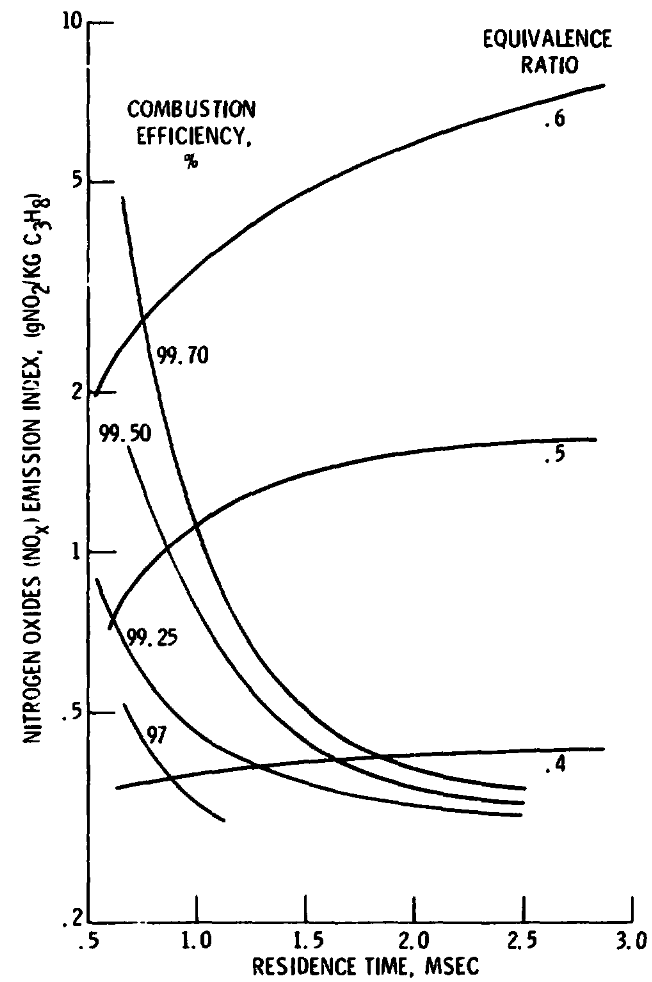


Figure 22. - NO$_x$ emissions from a premixed-prevaporized combustor.

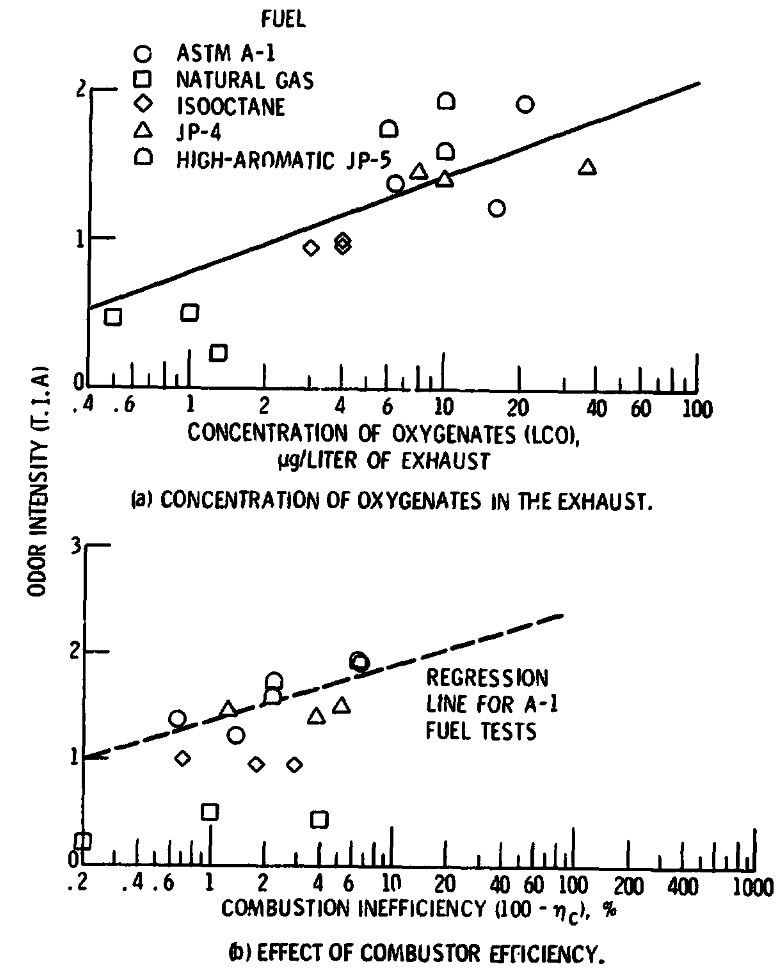
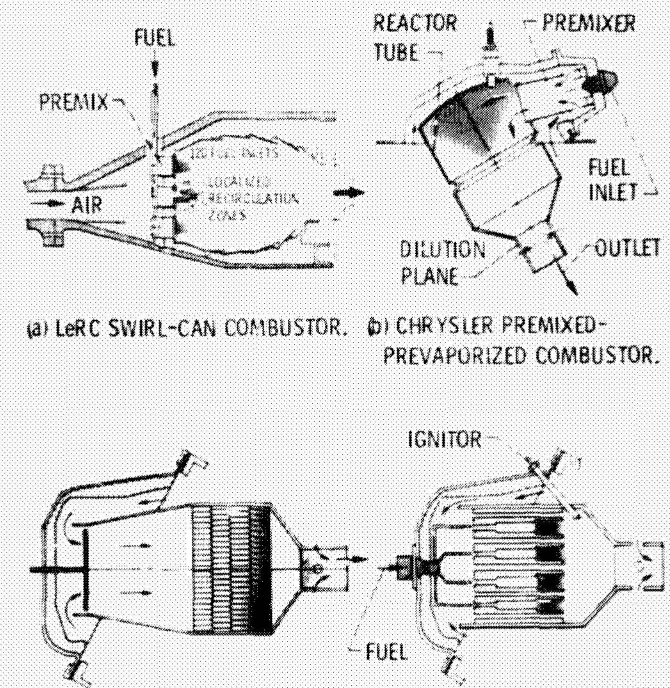


Figure 23. - Correlation between odor intensity for various fuels.



(a) LeRC SWIRL-CAN COMBUSTOR.
(b) CHRYSLER PREMIXED-PREVAPORIZED COMBUSTOR.
(c) LeRC CATALYTIC COMBUSTOR.
(d) LeRC MULTI-ELEMENT COMBUSTOR.

Figure 24. - Advanced combustor concepts.

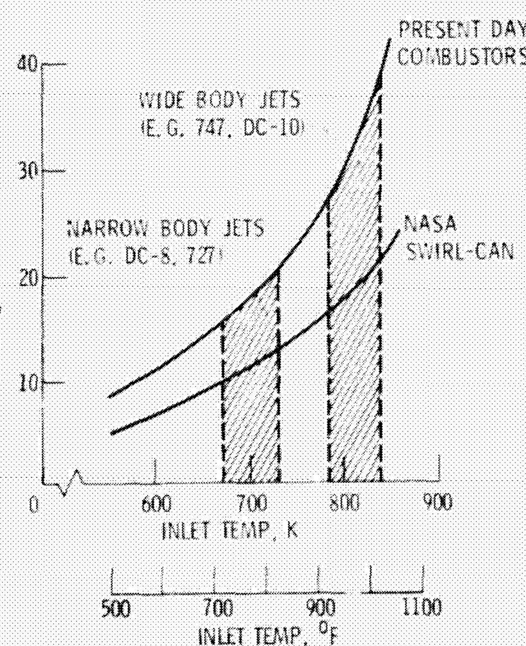
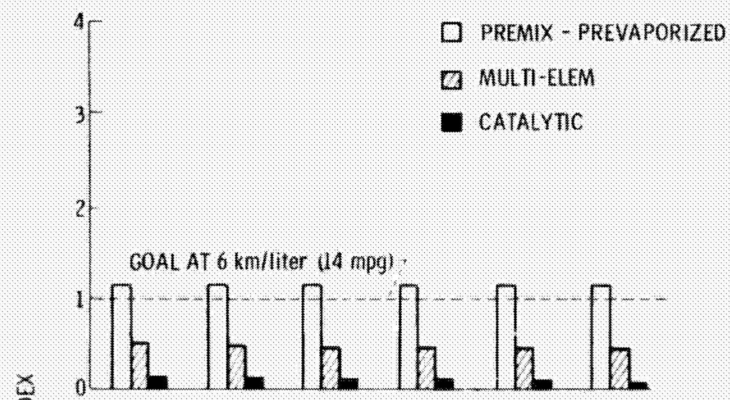
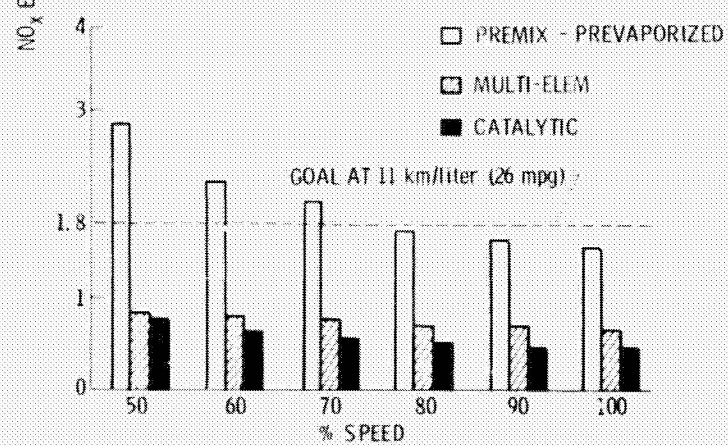


Figure 25. - Comparison of oxides of nitrogen emission levels from conventional combustors and the NASA swirl-can-modular combustor at takeoff conditions.



(a) CURRENT ENGINES - 980 K (1300° F) COMBUSTION INLET TEMPERATURE, 1090 TO 1310 K (1500° TO 1900° F) TURBINE INLET TEMPERATURE.



(b) ADVANCED ENGINES - 1200 TO 1480 K (1700° TO 2200° F) COMBUSTOR INLET TEMPERATURE, 1645 K (2500° F) TURBINE INLET TEMPERATURE.

Figure 26. - Predicted steady-state emissions for automotive gas turbine engines.

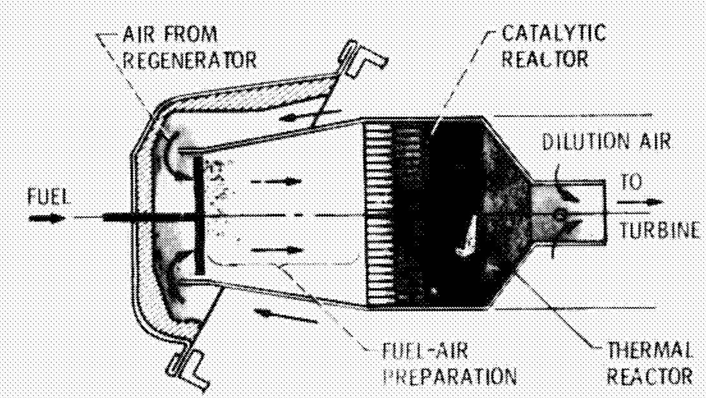
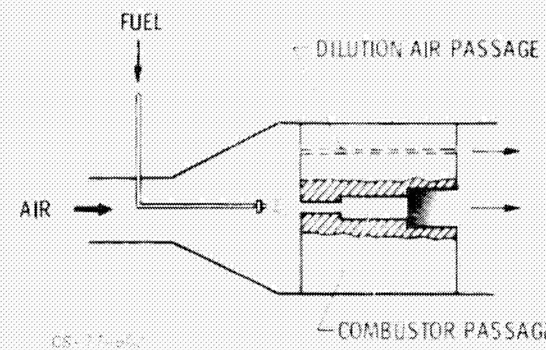


Figure 27. - Enlarged section of catalytic combustor



(a) SCHEMATIC OF CONCEPT.

(b) PHOTOGRAPH OF FLAME IN COMBUSTION PASSAGE.

Figure 28. - Single element of a multi-element combustor.

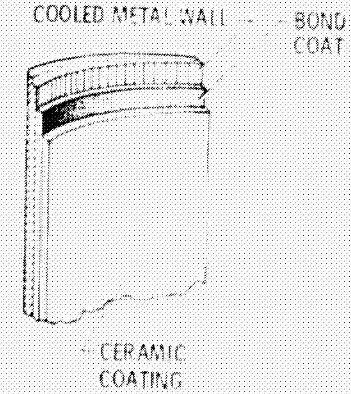


Figure 29. - Thermal-barrier-coated surface.

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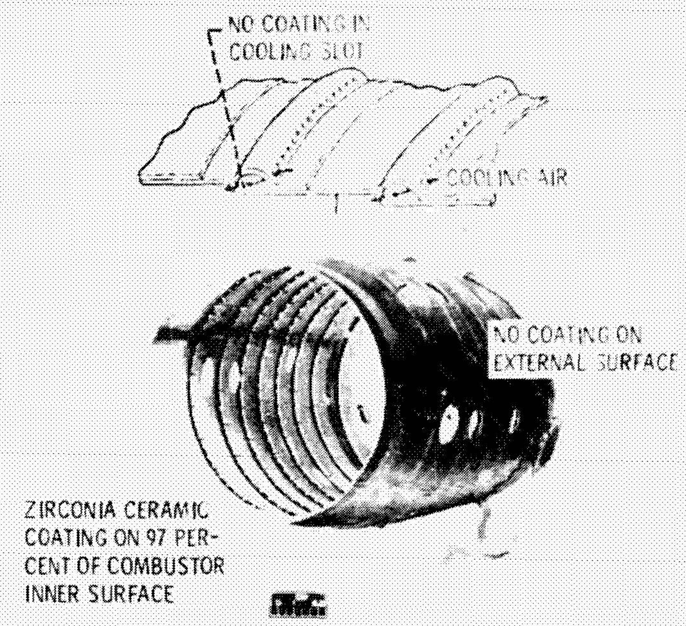

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Figure 30. - Thermal barrier coated combustor liner.

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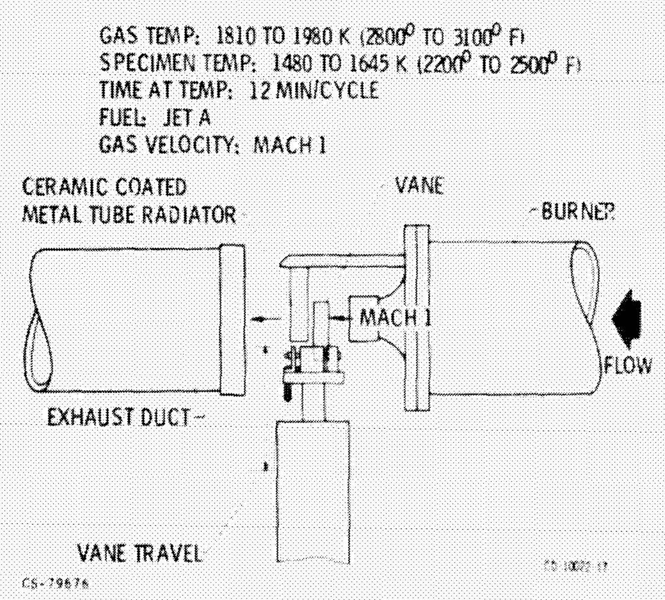

Figure 31. - LeRC Mach 1 burner facility.

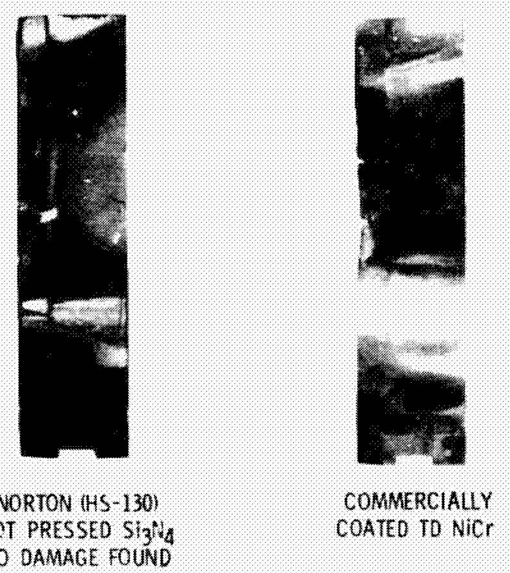

Figure 32. - Ceramic and coated blades after 100 cycles in Mach 1 burner at 1473 K (2192° F).

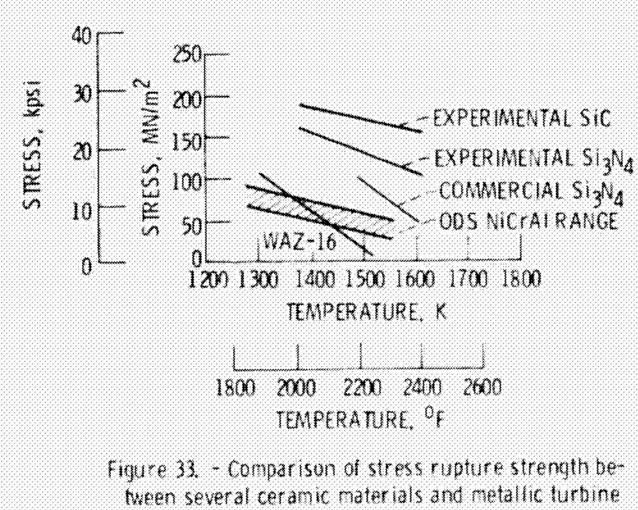

Figure 33. - Comparison of stress rupture strength between several ceramic materials and metallic turbine vane materials.

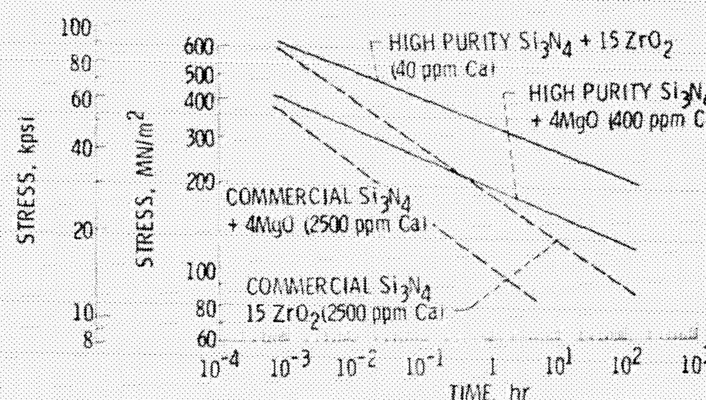

Figure 34. - Effect of starting powder purity and additives on 1598 K (2415° F) rupture properties of hot pressed Si_3N_4.

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Figure 35. - Diamond machined hot pressed silicon nitride turbine blade mounted in alloy disk for hot spin test.