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A HIGH-PRESSURE, HIGH-TEMPERATURE COMBUSTOR AND TURBINE-COOLING TEST FACILITY

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ABSTRACT

NASA-Lewis Research Center is presently constructing a new test facility for developing turbine-cooling and combustor technology for future generation aircraft gas turbine engines. Prototype engine hardware will be investigated in this new facility at gas stream conditions up to 2480 K average turbine inlet temperature and 4.14×10^{6} N/m^{2} turbine inlet pressure. The facility will have the unique feature of fully-automated control and data acquisition through the use of an integrated system of mini-computers and programmable controllers which will result in more effective use of operating time, will limit the number of operators required, and will provide a built-in self-protection safety systems. This paper describes the facility and the planning and design considerations involved.

INTRODUCTION

Compressor pressure ratios and turbine inlet temperature of aircraft gas turbine engines have been increasing steadily since the introduction of these engines during World War II. In both commercial and military applications, the trend is toward turbofan engines having compact, high-pressure, high-temperature gas generators. Cycle and mission analyses (such as reported in ref. 1) have indicated potential compressor pressure ratios of 40 for future engines with turbine inlet temperature up to 2480 K (4000° F).

In these analyses, it was assumed that materials and cooling schemes can be developed to operate satisfactorily at these elevated pressure and temperature levels where design requirements for the combustor and turbine components are beyond current technology. The higher compressor pressure ratios will be accompanied by higher compressor discharge temperatures. As a consequence, the compressor discharge air, which is the coolant for the combustor and the turbine, will have lower heat sink capacity. At the same time, the cooling requirements will increase because the combination of high pressure and high temperature will result in higher heat fluxes to engine parts exposed to the gas path environment. These high heat fluxes will impose problems such as high temperature gradients through the turbine vane walls as discussed in ref. 2. These temperature gradients and other heat transfer and combustion problems that are generated in the high-energy gaspaths of advanced engines cannot be properly evaluated by extrapolation of data obtained in the lower-energy gaspaths of current engines or test rigs. An engine or a test rig designed for high-pressure, high-temperature operation is required for such evaluation; but none exists currently. Simulation of such an environment in a specially designed rig is the most practical way to provide for this evaluation.

To achieve this simulation the NASA-Lewis Research Center has designed and is building a new facility consisting of independent and parallel combustor and turbine-cooling test sites with supporting service systems. An integrated system of mini-computers and programmable controllers will provide automated control, safety monitoring and data acquisition for the entire facility. This paper describes some of the design considerations and general characteristics of the facility and the test sites.

MATCHING TEST FACILITY CAPABILITIES TO ENGINE CHARACTERISTICS

The planning objective for the new high-pressure, high-temperature test facility was to simulate the gaspath environments of advanced turbofan engines in test sites that would accommodate prototype combustor and air-cooled turbine components. A combustor test site, a separate turbine-cooling test site, and the necessary service systems to provide and control the desired environment were the basis for this planning. Predicted future engine characteristics, such as the trend in compressor discharge temperature with compression ratio as shown on fig. 1, were used as a guide for establishing the operating levels of the new facility. Because the uniqueness of this test facility lies more in the pressure capabilities than in the temperature capabilities, it has been named the High Pressure Facility or HPF; it will be referred to by this abbreviation hereafter.

Design Criteria

Exact duplication of all actual engine operating environments was a desirable goal. However, technical and economic considerations sometimes dictated compromises that resulted in reasonable approximations of the desired goal. Existing air handling equipment at NASA-Lewis can provide pressurized and tempered air as indicated by the "Mode 0" block on fig. 2. The flow rate for this air supply is in excess of 136 kg/sec (300 lb/sec), the nominal maximum pressure rating is 1.14×10^6 N/m^2 (165 psia), and the nominal maximum temperature rating (using an existing indirect preheater) is 534 K (500° F) with short time capability to 589 K (600° F) at lower flow rates.

Design point conditions for the HPF of 40 atmospheres (4.14×10^6 N/m^2 or 600 psia) and 894 K (1150° F) at the combustor inlet were chosen as being representative of advanced turbofan engines. Increasing the temperature and

pressure of the air supply to these design point levels would require the expenditure of considerable energy in direct proportion to the airflow. The pressure increase would clearly require mechanical compression; both centrifugal- and axial-flow compressors were considered for this purpose. A survey of available drivers to power the required compressor(s) indicated that the pressure increases should be taken in two tandem steps and that the airflow should be limited to about 91 kg/sec (200 lb/sec). Greater compression ratios and/or greater flow rates would require drivers that were significantly larger and more costly. A turbine flow rate of 91 kg/sec (200 lb/sec) is a reasonable approximation of advanced turbofan engine characteristics.

The choice of the method of achieving the required temperature increase in the airstream was not as obvious as that for the pressure increase. Heating the air prior to compression would impose severe operating requirements on the compressors. Using indirect heat exchangers in the high pressure airstream after compression would result in bulky and expensive air-tempering equipment. The simple expedient of burning fuel directly in the airstream (vitiation) was not satisfactory because the resulting oxygen depletion was unacceptable for combustion research. The final choice was to use the heat of compression of relatively low efficiency compressors to achieve the temperature rise along with the pressure rise.

By separating the total energy increase in the airstream into two independent and tandem steps, the following advantages accrued:

1. Moderate-size, lower cost industrial gas turbine units could be used to drive the individual compressors

2. The temperature increase was divided between two compressors, thus simplifying thermal design problems, at least for the first compressor

3. The size and cost of the compressors was reduced

4. Operation at less than maximum design conditions could be done with less equipment.

Several factors dictated the choice of axial-flow over centrifugal-flow compressor. Chief among these factors was the more rapid metal temperature response and lower thermal stresses associated with the lower mass of axial flow compressors. The aerodynamic design for the compressors was obtained by scaling a proven 10-stage, 2.4:1 pressure ratio compressor that is currently in use in the 10 foot-by- 10-foot Supersonic Wind Tunnel at NASA-Lewis. This original compressor is 4.52 m (14.85 ft) in diameter; scaling for the new facility compressors resulted in a 0.6 m (23.6 in.) diameter for the first compressor and a 0.45 m (17.6 in.) diameter for the second compressor. These compressors are to be driven individually through similar gearboxes by identical gas turbine drivers. Adding the first of these facility compressors to the existing NASA-Lewis air-handling equipment would result in airstream conditions shown by the "Mode I" block on fig. 2; using both facility compressors in tandem would result in airstream conditions shown by the "Mode II" block. The representative engine compressor curve from fig. 1 is also shown on fig. 2. The capabilities of the new facility will permit simulation of typical engine compressor discharge (combustor inlet) conditions over most of the range of this curve.

Description of HPF

A schematic diagram of the HPF that resulted from this design planning is shown on fig. 3. Air-pressurized to 1.14×10^6 N/m^2 (165 psia) in the existing NASA-Lewis air system is tempered in an existing preheater to 534 K (500° F). This air passes through the first facility compressor to emerge at

2.28×10^6 N/m^2 (330 psia) and 725 K (845° F). The air then passes through the second facility compressor and exits at 4.55×10^6 N/m^2 (660 psia) and 894 K (1150° F). To facilitate the startup of the air system, all or any portion of this airflow can be routed directly to an existing exhaust system. In normal facility operation, the airflow will be directed to either the turbine-cooling or the combustor test site. To avoid the cost of flow control valves and metering stations for each of the test sites and to provide for leak-proof separation between test sites, a common control valve, a common metering station, and a flanged swinging elbow are used in the combustion air piping. Downstream of each test site are a pressure control valve and an isolation valve. Cooling air for ten separately-controlled cooling air systems for various parts of the turbine-cooling test site is taken from the facility compressor discharge at a point ahead of the metering run for combustion air.

The test sites and the control room for the new HPF are located in a rehabilitated section of an existing building; new adjoining structures were built to house the facility compressors and drive units. The physical layout of the facility is shown in a cutaway perspective view in fig. 4. Facility services such as air, fuel, cooling water, electric power, etc., are available in the vicinity of the building. The design for the new facility provides for interfaces with and the distribution and control of these services to the test sites as required. Control and data handling for these services and for the components of the HPF are automated to make the most efficient use of operating personnel and operating time. A programmable controller supervises the sequencing of facility systems operations. The operation and data acquisition for the test sites are fully automated through the use of an integrated digital minicomputer system called the Digital Control Center (DCC). A second program-

mable controller acts as a back-up to handle the shutdown of the test sites in the event of a failure of the DCC.

The following sections of this paper are general descriptions of the DCC. the turbine-cooling test site, and the combustor test site.

DIGITAL CONTROL CENTER

Automation of the test sites in the HPF involved two primary tasks. The first task was the control of the research experiment; the second task was acquisition, display, and transmission of the research data to a central data collector for later processing by a separate central computing system. Early in the design planning for automation, decisions were made to use digital control and to use several digital mini-computers, each with a dedicated task, rather than a single medium-size computer. This latter choice permitted complete separation of tasks between computers and allowed variations in the degree of security for the different tasks according to relative criticality to overall operation. For example, a control function is more critical in nature than a data gathering function; therefore, the rigid program in the mini-computer supervising controls functions should not be jeopardized by on-line changes that may be made in the more flexible program in the mini-computer supervising data gathering. In the final system design, four digital mini-computers were used for the purpose of automated control and data handling for the test sites. These four mini-computers plus peripheral equipment comprised the Digital Control Center (DCC).

General Description

Figure 5 shows a simplified schematic of the DCC that will serve as a guide for following this description. The four mini-computers in the DCC are interconnected; however, separation of tasks between the computers

simultaneously allows security in the control programming and flexibility in the data handling and operation areas. The computers are labeled according to their dedicated task as the Input, Control, Operation, and Research Computers. The computers and the peripheral equipment are located in the control room (shown in fig. 4) and are powered by a 30 KW uninterruptable power supply.

The Input Computer serves as data acquisition controller, data buffer and intermachine communication buffer. This computer will be programmed in Assembly language. The control Computer directly controls approximately 20 process variables on the turbine-cooling rig and eight process variables on the combustor rig, with update rates of 20-150 per second. This computer also will be programmed in Assembly language. The Operation Computer supervises overall control of the experiment such as setpoint calculations during start-up, test point and shut-down operations. This computer will use a real-time operating system with high-level language application programming. The Research Computer continuously displays the information needed by the test director, and selectively stores, collates, and transmits data to a Central Data Collector via telephone lines. High level language application programming will be used in this computer also. Associated with each of the four computers is a magnetic core memory; also, all except the Input Computer have associated printers for making hard-copy records of preliminary data.

Data Acquisition

The acquisition of research data will be performed by the DCC in a semiautomatic mode (i.e., automatic collection and manual control of data flow by the test director through the use of programmed push-button selector switches). On command from the Input Computer, the multiplexer-digitizer

will scan all the various types of signal inputs, digitize them as required, and load them into the appropriate memory within the Control, the Operation, or the Research Computer. This process of loading raw fixed-point data into memory will run continuously at a rate commensurate with the requirements of the control and data acquisition/display systems.

Certain selected stored data in these memories will be used in calculations such as flow rates, averages and ratios. The results of these calculations, as well as temperature and pressure data, will be displayed in engineering units on several cathode ray tube (CRT) displays. When the test director, after viewing these CRT displays, determines that the desired operating point has been reached, he will initiate the recording of data by push-button control, causing the most recent complete set of data in memory to be transferred to temporary storage on a magnetic disk associated with the Research Computer. These accumulations of data on the disk can be transferred to the Central Data Collector either during or after the run at the test director's discretion.

The first category of inputs to the DCC are analog signals from various sources such as strain gages, pressure transducers, thermocouples and position indicating devices. All these signals enter the DCC through the 1024 channel multiplexer/digitizer. Initially only 704 of these channels are being used; the remaining 320 channels allow for future expansion. The digitizer is an amplifier-per-channel unit that can accommodate instrument-type signals in the range of 5 MV to 10 V full scale. Analog calibration voltages may be switched into each of the input channels in place of the normal signals to allow verification of proper operation. The switching of the calibration voltages is done under computer program control. These signals enter the DCC via the

Input Computer.

The second category of inputs to the DCC are digital signals in the form of discrete on-off type signals. These discrete signals go directly to the Control, Operation, and Research Computers. Both program-controlled and program-interrupting type signals are involved. The program-controlled signals are used by the user application program to check the status of switch contacts. For example, the computer may be programmed to detect the closing of a limit switch before advancing a probe actuator. The interrupting type signals are used for devices that, because of priority or short duration conditions, require fast service from the computer. All push button selector switches for controlling data flow, for example, will be associated with this type of signal.

The third category of inputs come from special instrumentation sources such as fluid flow and speed transducers (interfaced through rate converters), pneumatic multiplexers, a turbine blade metal temperature mapping system, a rotary data package (for turbine blade temperature and pressure measurements) and a combustion gas analysis system. The fluid flow transducers are turbine-type flowmeters for fuel and water flow measurements and the speed transducers are magnetic pickups for measuring turbine shaft speed. Pulse-type signals from these devices will be conditioned by rate (frequency-to-digital) converters to obtain digital word outputs that are proportional to the period of the input pulse trains.

The turbine blade metal temperature mapping system is a customized photo-electric system that was developed by NASA-Lewis for surface temperature measurements on rotating blades. (For detailed description of this system, see ref. 3.) The system is capable of resolving a spot diameter of 0.05

cm (0.02 in.) on a blade moving at tip speeds of the order of 300 to 400 m/sec (1000 to 1300 ft/sec). Approximate real time displays of blade temperature profiles at steady-state operating conditions can be generated for a single blade or for small groups of blades. To handle the data from this system, a fifth dedicated mini-computer (not currently included in the DCC) will be interfaced with the Research Computer. This fifth computer will do limited calculations on the data and will drive its own CRT displays. Through the interface, the Research Computer will record these data and transmit them to the Central Data Collector.

The pneumatic multiplexers scan certain gaspath and cooling air pressure signals. Each multiplexer has 48 channels for measuring steady-state pressures; 6 channels in each multiplexer are assigned to calibration pressures. The rotary data package (see fig. 4) scans thermocouple and pressure signals from the turbine blades. There are 72 channels on this package; a maximum of 10 of these channels can be used for pressure signals (see ref. 2). Both of these devices will operate essentially in a free-running mode (i.e., continuous scanning and readout). The outputs of each device will be amplified, digitized and stored in a small memory resident within its interfacing hardware. Under program control, this information will be transferred into memory of the Research Computer. One rotary data package and four pneumatic multiplexers will be used with the turbine cooling rig; four pneumatic multiplexers will be used with the combustor rig.

The data obtained from the combustion gas analysis system (which is described in the section on the combustor test site) will go th the Input Computer via a microprocessor interface. These data will be displayed in engineering units on a CRT connected to the Research Computer and also will

be transmitted to the Central Data Collector for further processing.

Control System

Actual control of the test rigs is accomplished by the Control Computer. As previously indicated, there are 20 separate process variables, each having its own control loop, that must be controlled for the turbine-cooling rig and eight variables for the combustor rig. The Control Computer performs the function of 20 separate process controllers on a time-sharing basis. The demands on this computer are severe because of the large number of control loops involved and the high speed at which these loops must be serviced. The majority of the loops have a very fast response time, are highly interactive, and are required to perform accurately under dynamic conditions. These operating characteristics require the use of sophisticated control strategies specifically tailored for each loop and often incorporating advanced compensation schemes such as feedforward control, adaptive control and optimization. The Control Computer, for example, reads fresh data, calculates the set point error, and computes the valve position using whatever compensation is needed for the loop. The control loops are serviced sequentially. As soon as each loop has been serviced and the position set point transmitted to the control valve, the computer proceeds to service the next loop.

The automated supervision of the overall test site operation is accomplished by the Operation Computer. The major tasks that this computer performs include:

1. Prerun functional check of control equipment;

2. Start sequence to get rig to a designated idle condition;

3. Ramping to and from designated data point conditions, and holding designated steady-state data point conditions

4. Normal or Emergency shutdown of test rig;

5. Supply continuous updated displays of test rig operating conditions.

Before a given test run, a table of data point conditions is entered into the Operation Computer, thus establishing the test conditions to be set and maintained by the computer.

Although overall test site operation is handled by the Operation Computer, there are situations where unprogrammed action must be initiated by the operation; therefore, suitable controls have been provided to allow operator intervention without changing the computer software.

Prior to making any research runs, a hybrid simulation of the process controls systems for the test sites will be run. To accomplish this, analog models of both the combustor and the turbine-cooling site control/process systems will be programmed on an independent analog computer. The Control Computer will then be interfaced temporarily to this analog computer and simulated test runs will be made to determine overall systems response times, the effects of the interactions between systems, etc. It is intended that this simulation system will be a continuously available system-development tool so that any future changes to a control system can be checked by simulation before any hardware or software is actually modified or added.

TURBINE-COOLING TEST SITE

The turbine-cooling test site was designed for turbine-cooling research on various prototype air-cooled configurations of turbine vanes, blades and blade tip shrouds (endwalls). This test site consists of a rotating turbine cooling test rig for exposing these components to simulated engine environments, a waterbrake to absorb the power output of the turbine, and the necessary piping, controls, instrumentation, etc.

Design Criteria for Turbine-Cooling Rig

To simplify hardware requirements, a single stage turbine was chosen for the test rig. This turbine stage was designed to meet the conditions imposed on the first stage of the core (high pressure) turbine of a high-bypass ratio turbofan engine. From the turbine-cooling point of view, this stage is the most critical part of the engine because the heat fluxes (associated with the high gaspath temperatures and pressures) are maximum at this point in the engine cycle.

A conceptual design of the turbine-cooling rig was established by NASA-Lewis; the main features of this concept were:

- A single-stage air-cooled turbine

- Individually replaceable and interchangeable turbine blades and turbine vanes

- Test cascade(s) in both blade and vane rows with cooling air flows to these cascades controlled independently from the cooling air flows to the remaining slave or workhorse airfoils in the row

- Replaceable air-cooled turbine blade tip shroud with independently controlled cooling-air flow system

- Ready access to the turbine section for ease of servicing and/or replacement of turbine components (stator, rotor and tip shroud assemblies)

- Combustion-air inlet flow path and combustor casing internal geometry conforming to a specified NASA annular combustor design

- Simultaneous maximum conditions of 4.14×10^6 N/m^2 (600 psia) pressure and 894 K (1150° F) temperature in the combustor and cooling air supplies. Initial gaspath maximum temperature of 2200 K (3500° F); ultimate gaspath maximum temperature of 2480 K (4000° F). (All gaspath components down-

stream of the turbine designed for the ultimate condition.)

- Bearing and shaft designed for maximum operating speed of 43,000 rpm
- Turbine power absorbed by a direct-drive waterbrake (no gearbox)

In addition to these conceptual features, the initial design parameters listed on Table I were established by NASA-Lewis. The above information plus other design specifications were included in a contract for the design of the entire turbine-cooling rig and the fabrication of all parts with the exception of the vanes, blades, disk and tip shroud. (These excepted parts are being fabricated under separate contracts.)

Description of Turbine-Cooling Rig

A view of the turbine section of the turbine-cooling rig is shown on fig. 6. Access to the stator assembly is gained by separating the rig at the No. 3 Splitline and moving the combustor section. Access to the rotor and tip shroud assemblies is gained by separating the rig at the No. 2 Splitline and moving the combustor and stator sections.

A ten-vane test cascade occupies the upper portion of the stator assembly which consists of a total of 36 vanes. Test vane air for this cascade is routed to the individual test vanes as indicated on fig. 6. The remaining 26 slave vanes (which do not have the angled inlet duct attached) are cooled by a separate slave vane air supply from the annular manifold which surrounds the outer periphery of the stator assembly. Two diametrically-opposed 6-blade test cascades are located in the blade row. Test blade air for both cascades flows through the centerline supply tube and up the rear face of the turbine disk to enter rear of the twelve test blade bases below the hub platforms. The remaining 52 slave blades on the rotor are cooled by a separate slave blade air supply that flows up the front face of the turbine disk and enters the front

of the s e blade bases. The initial turbine tip shroud is made in eight equal circumf ential segments and has a h eycomb gaspath s ace backed up by porous ven wire cloth and a perfor ed structural suppo... The tip shroud air ente the annular manifold that s rrounds the outer periphery of the shroud assembly as shown on fig. 6.

Positions for mounting radially-actuated water-cooled temperature and pressure measuring probes were provided at Stations 4 and 6 (see fig. 6). Similar probe housings for optical borescopes or optical fiber bundles will be used at these same locations to make infrared surface temperature measurements on the blades and vanes. Typical displays of surface temperature profiles on rotating blades made with such instrumentation can be found in ref. 3. Thermocouples and pressure sensing tubes are mounted on the blades and vanes to measure metal temperatures and gas and cooling-air flow path temperatures and pressures. The instrumentation leads from these measurements on the vanes are routed through the outer pressure shell of the rig to appropriate terminals; the leads from the blades are routed down the front face of the turbine disk and through the center of the turbine and waterbrake shafts to the rotary data package located at the outboard end of the waterbrake (see fig. 4).

The cooling air flow to each of the cooled components in the turbine cooling-rig is weight-flow controlled by the Control Computer, based on measurements from venturis in the supply lines. Cooling-air temperatures and pressures are measured in internal manifolds in each of the separate cooling systems. The exhaust gas leaving the turbine is spray-cooled in the exhaust collector to a 589 K (600° F) temperature level prior to discharge into the exhaust system.

Turbine-Cooling Test Program

The test program for the turbine-cooling rig will cover a variety of internal cooling configurations for the vanes, blade and tip shroud. The test-slave grouping of airfoils in the vane and blade rows will permit changes in the cooling configurations to be tested by replacing only those airfoils in the test cascades; the slave airfoils remain as permanent parts of the rig. The entire tip shroud assembly will be replaced with changes in tip shroud cooling configuration; uniform thermal growth in all segments of the tip shroud could not be assured otherwise. The ranges of gaspath and cooling-air conditions that will be covered by these tests are:

	Pressure		Temperature	
	$N/m^2 \times 10^{-6}$	psia	K	^oF
Gaspath - Combustor inlet	0.52 - 4.14	75 - 600	311 - 894	100 - 1150
Gaspath - Combustor exit	0.48 - 4.0	70 - 580	1090 - 2480	1500 - 4000
Cooling air	0.52 - 4.14	75 - 600	311 - 894	100 - 1150

Test runs will be made at several fixed levels of gaspath temperatures and pressures within the ranges listed above while varying cooling air temperature and cooling-air flow rate. During these test runs, temperature, pressure, and weight-flow measurements will be made in the gaspath and in each of the cooling-air supply lines. Metal temperature measurements will be made on the combustor, the vanes, the blades, the turbine disk, and the tip shroud, and on the gaspath walls downstream of the turbine. Thermocouples will be used to measure the temperatures of discrete points and optical methods will be used to obtain thermal surface maps on the vanes and on the blades.

The cooling effectiveness of the various configurations of air-cooled vanes, blades, and tip shrouds will be evaluated on the basis of these measurements. Comparisons will be made between predicted and measured values of metal temperatures and cooling-air flows; and, based on these comparisons, corrections and improvements will be made to the analytical methods of prediction.

COMBUSTOR TEST SITE

The combustor test site contains a test rig (consisting of a combustor housing and an instrumentation and exhaust section as shown on fig. 7) and the necessary piping, controls, instrumentation, etc. The gaspath for this test site was designed for a combustion airflow equal to the full rated capacity of the HPF; i.e., flow rates of 91 kg/sec (200 lb/sec), inlet pressures of 4.0×10^6 N/m^2 (580 psia), inlet temperatures of 894 K (1150° F), and ultimate gaspath temperatures of 2480 K (4000° F).

Combustor Rig

A combustor housing for this test site is being designed to accommodate the maximum site flow capacity. However, the first combustor that will be tested is the combustor for the companion turbine-cooling rig. This combustor was designed for a maximum combustion air flow of about 68 kg/sec (150 lb/sec) and it will be tested here in its own housing which is part of the turbine cooling rig. The sketch on fig. 8 shows the initial combustor rig which will consist of the combustor housing for the turbine-cooling rig mated through an adapter piece to the instrumentation and exhaust section of the regular combustor rig. The combustor in this assembly is a modular swirl-can combustor similar to many other designs tested at NASA-Lewis. Combustors of this type have achieved exit temperatures in excess of 2370 K (3800° F) and have demonstrated good efficiency and durability (refs. 3, 4, and 5). The overall length

of the combustor housing shown in fig. 7 is 1.26 m (49.5 in.).

The combustor exit annulus has a 0.51 m (20 in.) outer diameter and a 0.43 m (17 in.) inner diameter. The pertinent combustor design and operating parameters are summarized in Table II.

Figure 8 is a photograph of this combustor looking upstream into the array of swirl-can modules. There are two concentric rings of modules with 24 modules in each ring. Fuel is injected into each module, splash-atomized and mixed with air as it passes through a swirler, and burns in the recirculation zone formed downstream of the module. The fuel flow rate can be controlled to four circumferential zones in each module ring, a capability that was found to be very useful in achieving desired exit temperature profiles.

A small amount of liner cooling air is required for two reasons. First, the early buildup and maintenance of a thick air film on the surface of the liner cools and protects the liner for a considerable axial distance. Secondly, the liner wall contour reduces convective heating of the liner compared to more conventional liner geometries. This fuel module and liner combination has been tested at pressures of 0.97×10^6 N/m^2 (140 psia), inlet temperatures of 894 K (1150 ^{o}F) and to exit temperatures approaching 2480 K (4000 ^{o}F). The durability of both the combustor modules and the film-cooled liner at these conditions has been satisfactory.

Standard gaspath instrumentation will be installed in the combustor housing. Fixed rakes at the diffuser inlet will be used to measure the combustor inlet air temperature and total pressure as shown on fig. 7. Provisions have been made to install a limited number of skin thermocouples and/or static pressure tubes on both the inner and outer combustor liners.

Instrumentation at the combustor exit consists of a six-armed rotating probe assembly (two arms each for temperature, total pressure and on-line gas analysis). This assembly (see fig. 7) will rotate through an arc of 180 degrees and can be remotely indexed in small circumferential increments for gaspath sampling. On-line gas analysis will be used to measure combustor emissions and combustion efficiency. Thermocouples will be used to the maximum temperature limit of the Pt-Rh thermocouple wire to indicate combustor exit temperature profiles. From these thermocouple measurements, combustor exit temperature maps similar to that shown on fig. 9 will be generated. On this figure, letters and numbers are used to indicate regions that are either hotter or colder, respectively, than the average exit temperature; the magnitude of the deviation from the average value is expressed as a fraction of the combustor temperature rise. Beyond the temperature limit of the thermocouples, average exhaust gas temperature will be inferred from the gas analysis measurements.

Gas Analysis System

The on-line gas analysis system is fully automated; the instruments within the system can be zeroed, spanned and ranged automatically from the control room prior to a test run. If an instrument cannot be electrically zeroed or calibrated, the need for a manual adjustment will be indicated automatically. After calibration, the instruments are switched to the run mode and are ready for data gathering. If, during the survey of the combustor exhaust, a constituent level rises (or falls) into the next instrument range, the automated system will shift the instrument to the appropriate range. Between data points it is possible to re-set the zero and full-scale readings of the instruments remotely. The instruments used in the automatic on-line gas sampling system and their range of measurement are listed in Table III.

During test runs in the combustor rig, gas sampling will be done simultaneously at five radial gaspath heights by two opposing arms of the rotating probe assembly; these samples will be mixed within the individual arms. Particular attention was given to cooling the gas as quickly as possible to prevent any change in gas composition. After mixing and cooling, the samples from the two arms are routed outside the instrumentation section, mixed together, and sent to gas analysis instruments. All sample routing from the sensing arm to the instruments is done through stainless steel steam-jacketed lines to maintain the sample temperature at a constant 422 K (300° F).

Research Programs

As indicated previously the first combustor research effort will be devoted to the development of the combustor for the turbine-cooling rig. Preliminary tests of this combustor are being conducted at lower pressures in another facility. After these preliminary tests, the combustor will be installed in the combustor rig in HPF. Here, the initial test phases will encompass testing to the $2.07{\times}10^{6}$ N/m^{2} (300 psia) level using only one facility compressor. Later tests will cover the complete pressure and temperature ranges of the HPF. The primary purpose of these tests is to prove that this combustor will be durable and that it will achieve the performance levels required for the planned operation in the turbine-cooling rig. When these tests are successfully completed, the combustor will be transferred to the turbine-cooling rig.

For future combustor testing, a larger combustor housing will be built and the instrumentation section will be modified to accept the larger diameter exit annulus combustors.

CONCLUDING REMARKS

This test facility will provide proper simulation of the gaspath conditions in advanced aircraft gas turbines for the evaluation of prototype combustor and air-cooled turbine designs. Automated control and data handling will permit the most efficient use of operating personnel and operating time. The flexible designs of the combustor and the turbine-cooling rigs will allow frequent changes in the prototype test hardware as the test programs progress. As a research tool, the HPF is unique and promises to have a very productive life.

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TABLE I. - INITIAL TURBINE DESIGN PARAMETERS

Inlet temperature	2200 K (3500° F) [2480 K (4000° F), future max.]	
Inlet pressure	4.0×10^{6} N/m^{2} (580 psia)	
Airfoils —	Vane	Blade
• Number	36	64
• Tip diameter	0.51 m (20 in.)	0.51 m (20 in.) [0.53 m (21 in.) future max.]
• Hub diameter	0.43 m (17 in.)	0.43 m (17 in.) [0.41 m (16 in.), future min.]
• Span	0.038 m (1.5 in.)	0.038 m (1.5 in.)
• Axial chord	0.038 m (1.5 in.)	0.036 m (1.4 in.)
• Aero. profile	Constant section, untwisted	
• Internal cooling configuration	Combination of impingement and full-coverage film cooling	
• Fabrication	Cast shell, sheetmetal insert	

Maximum coolant/gas flow ratios -

• Vane	0.20
• Blade	0.15
• Tip	0.05
• Shroud	0.30

TABLE II. - COMBUSTOR DESIGN PARAMETERS

Ref. area	$0.248\ m^2$ ($384\ in.^2$)
Diffuser area ratio	1.8
Diffuser inlet Mach no.	0.331
Flow parameter, $W_A \sqrt{T}/PA_{REF}$	1.985×10^{-3} (0.02764)*
Reference velocity	18.3 m/sec (60 ft/sec)
Fuel flow rate, max.	4.94 kg/sec (10.9 lb/sec)
Combustor press. loss, isothermal	5.3%
Combustor volume	$0.0368\ m^3$ ($1.30\ ft^3$)
Combustor space rate, max.	4.9609×10^6 (13.5×10^6)**
Cooling airflow, outer liner	10%
Cooling airflow, inner liner	7%

Units: * $kg\times K^{1/2}/sec\times N/m^2\times m^2$ ($lb\times {}^oR^{1/2}/sec\times lb/in.^2\times in.^2$)
** $Joule/hr\times m^3\times N/m^2$ ($Btu/hr\times ft^3\times atmos$)

TABLE III. - GAS ANALYSIS SYSTEM INSTRUMENTS

Constituent	Instrument	Number ranges	Meas. range
UHC*	Beckman 402 FID	4	0 - 2%
CO	Beckman 864 NDIR	2	0 - 15%
CO	Beckman 865 NDIR	2	0 - 1000 ppm
CO_2	Beckman 864 NDIR	1	0 - 15%
NO	TECO 10-A	4	0 - 2%
NO_2	TECO 10-A	4	0 - 2%
H_2O	Beckman 864 NDIR	1	0 - 15%
O_2	Beckman OM-11	1	0 - 25%

*UHC - unburned hydrocarbons.

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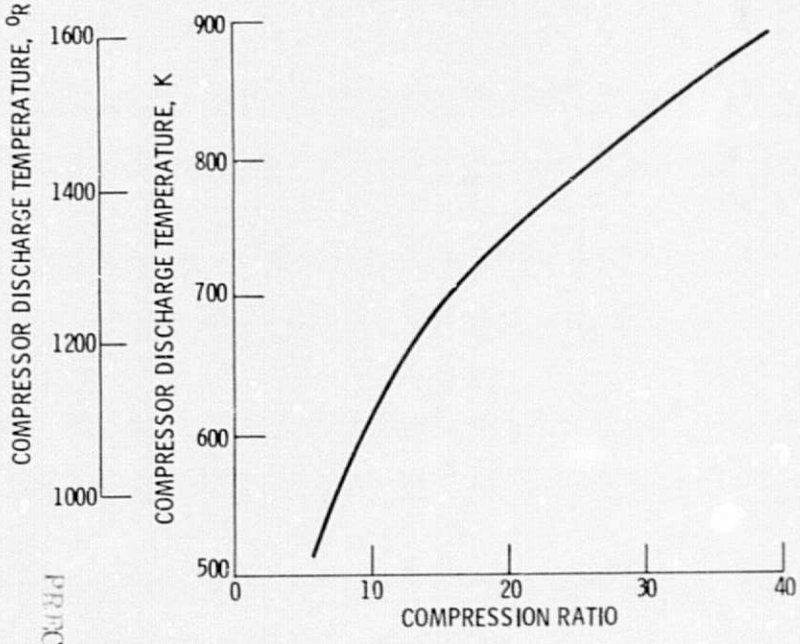

Figure 1. - Turbofan engine compressor characteristics.

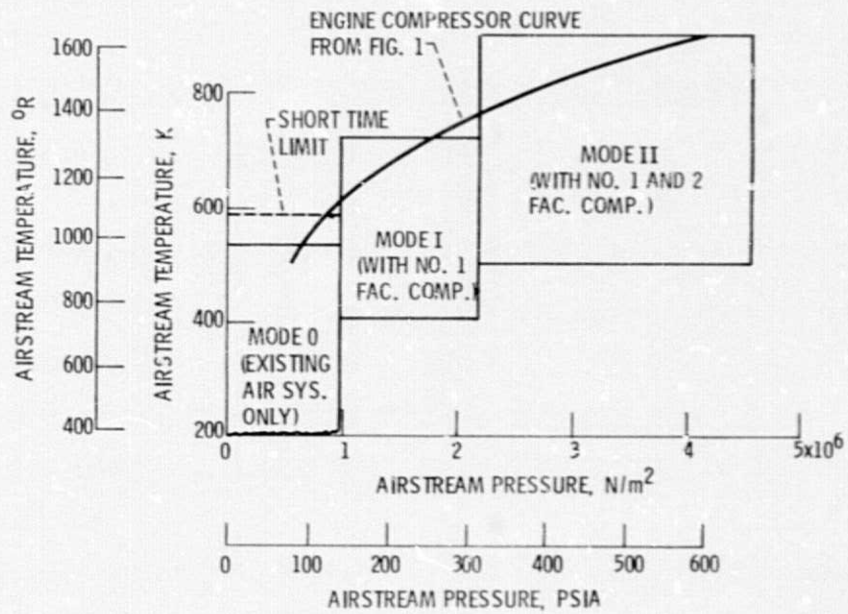

Figure 2. - Design combustion air conditions for HPF.

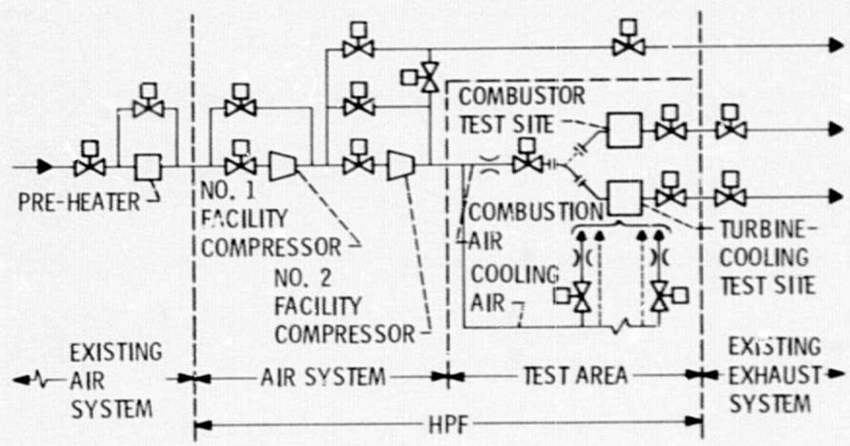

Figure 3. - Schematic diagram of HPF.

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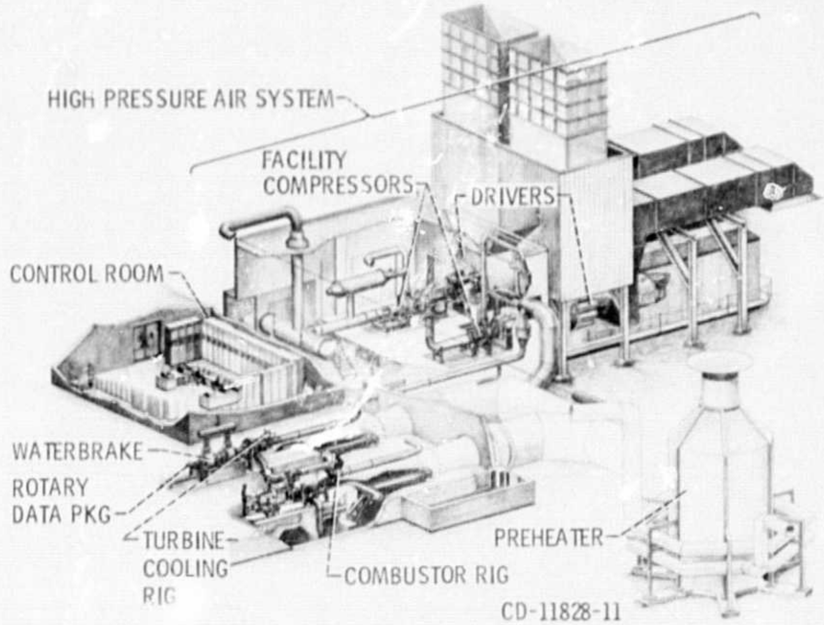

Figure 4. - Perspective view of HPF.

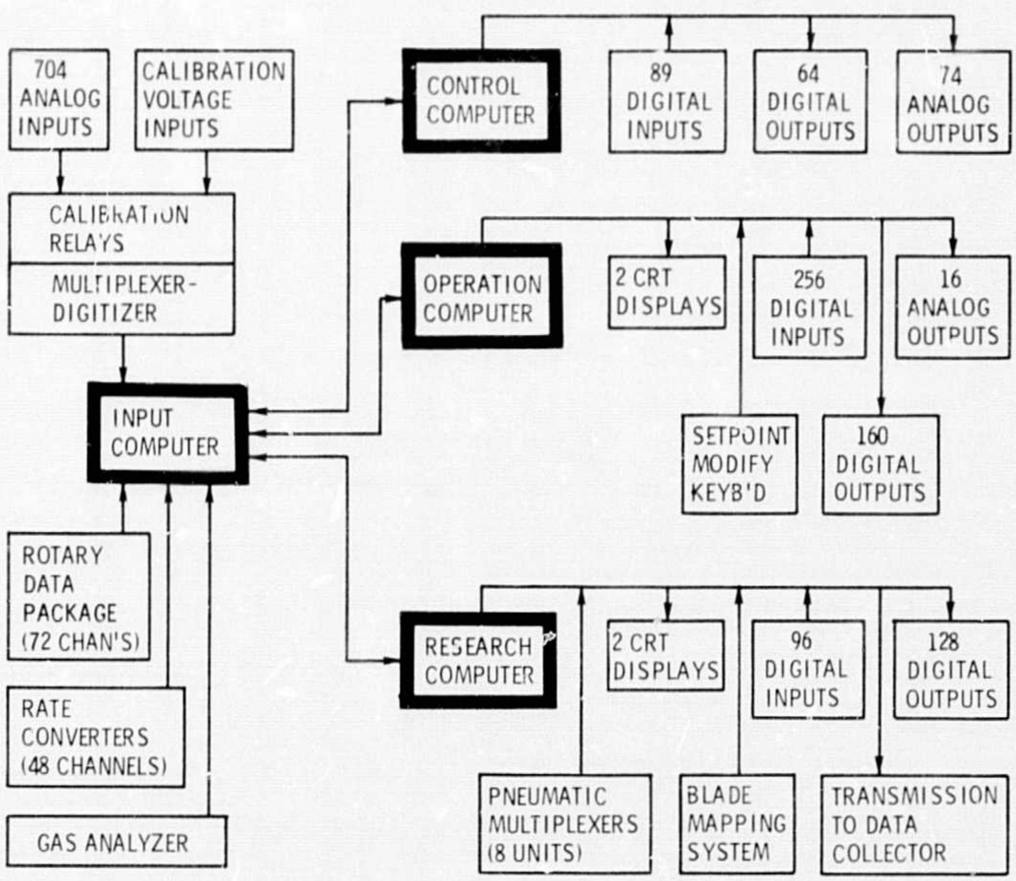

Figure 5. - Schematic of Digital Control Center (DCC).

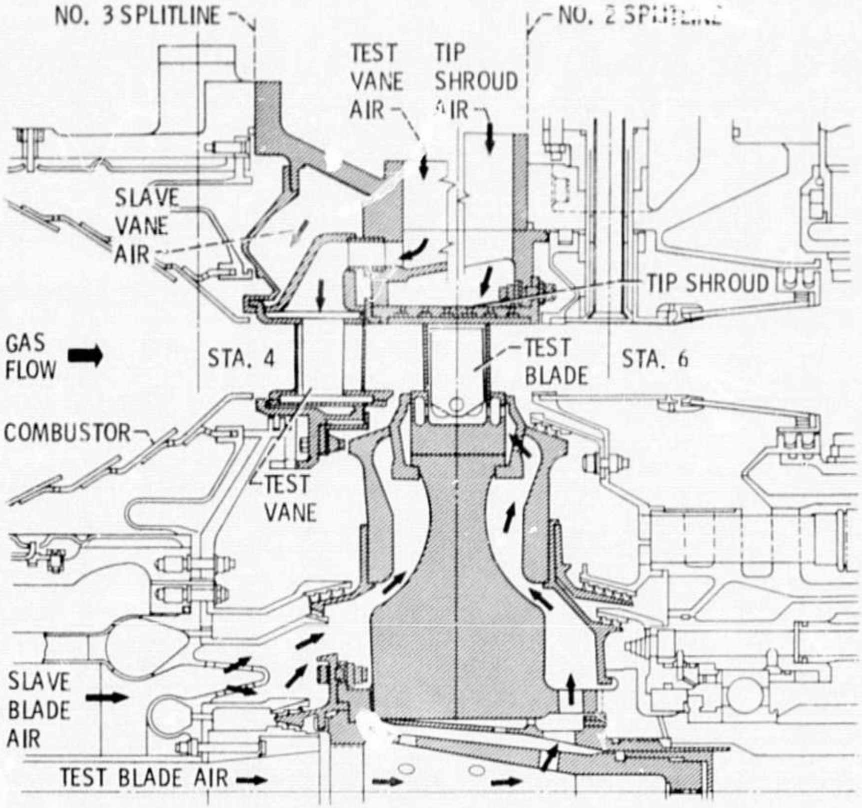

Figure 6. - Test section of turbine-cooling rig.

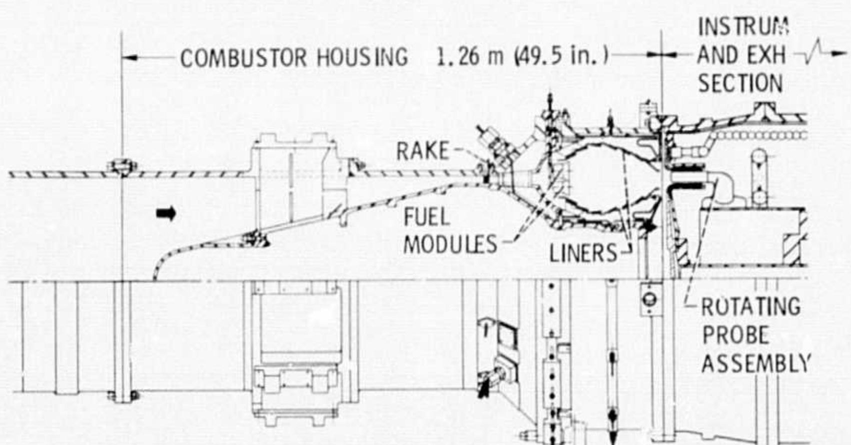

Figure 7. - Initial combustor rig.

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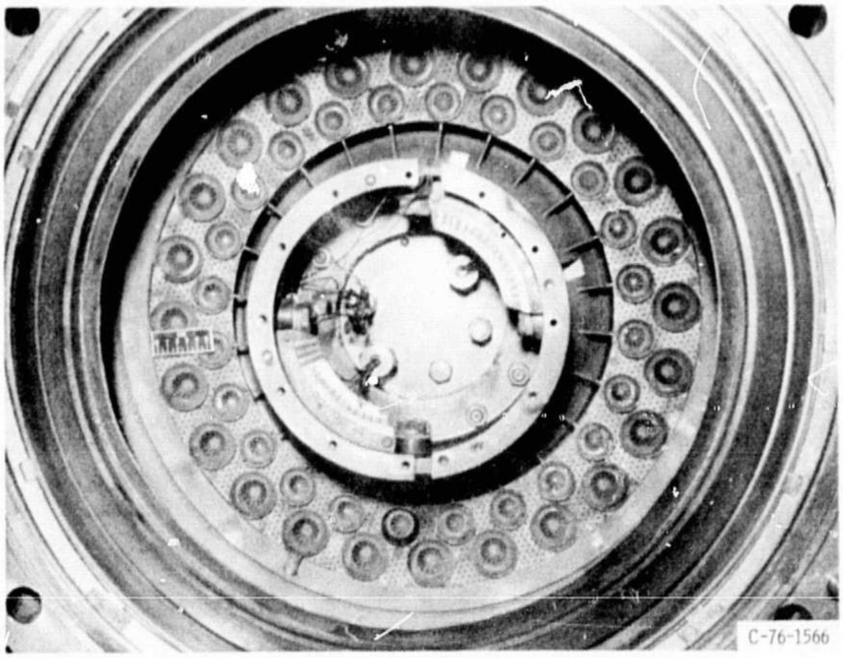
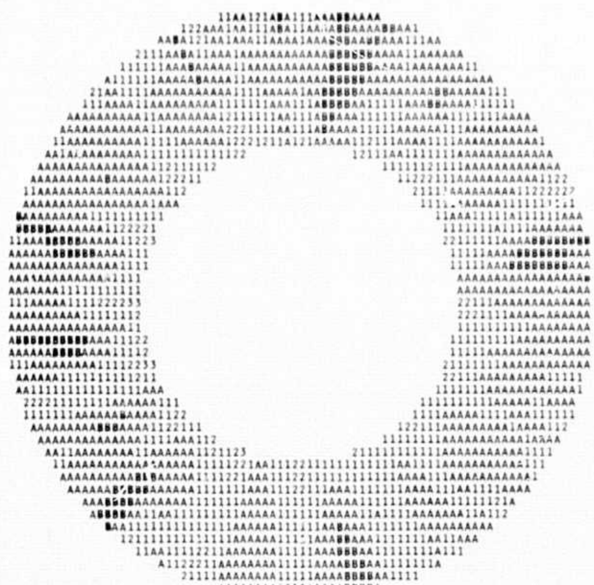


Figure 8. - Upstream view of fuel module array.



LOCAL FACTOR RANGE

J	1.00 TO 0.90
I	.90 TO .80
H	.80 TO .70
G	.70 TO .60
F	.60 TO .50
E	.50 TO .40
D	.40 TO .30
C	.30 TO .20
B	.20 TO .10
A	.10 TO .00
1	.00 TO -.10
2	-.10 TO -.20
3	-.20 TO -.30
4	-.30 TO -.40
5	-.40 TO -.50
6	-.50 TO -.60
7	-.60 TO -.70
8	-.70 TO -.80
9	-.80 TO -.90
0	-.90 TO -1.00

Figure 9. - Exit temperature map for annular combustor.