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Hypersonic Engine Component Experiments in a High Heat Flux, Supersonic Flow Environment

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HYPERSONIC ENGINE COMPONENT EXPERIMENTS IN A HIGH HEAT FLUX, SUPERSONIC FLOW ENVIRONMENT

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SUMMARY

A major concern in advancing the state-of-the-art technologies for hypersonic vehicles is the development of an aeropropulsion system capable of withstanding the sustained high thermal loads expected during hypersonic flight. Even though progress has been made in the computational understanding of fluid dynamics and the physics/chemistry of high speed flight, there is also a need for experimental facilities capable of providing a high heat flux environment for testing component concepts and verifying/calibrating these analyses. A hydrogen/oxygen rocket engine heat source has been developed at the NASA Lewis Research Center as one element in a series of facilities at national laboratories designed to fulfill this need. This "Hot Gas Facility" is capable of providing heat fluxes up to 450 w/cm^2 on flat surfaces and up to 5000 w/cm^2 at the leading edge stagnation point of a strut in a supersonic flow stream. Gas temperatures up to 3050 K can also be attained. Two recent experimental programs conducted in this facility are discussed. The objective of the first experiment is to evaluate the erosion and oxidation characteristics of a coating on a cowl leading edge (or strut leading edge) in a supersonic, high heat flux environment. Macrophotographic data from a coated leading edge model show progressive degradation over several thermal cycles at aerothermal conditions representative of high Mach number flight. The objective of the second experiment is to assess the capability of cooling a porous surface exposed to a high temperature, high velocity flow environment and to provide a heat transfer data base for a design procedure. Experimental results from transpiration cooled surfaces in a supersonic flow environment are presented.

NOMENCLATURE

c_p	specific heat
h	heat transfer coefficient
k	thermal conductivity
K′	efficiency parameter, $(T_{co} - T_{ci})/(T_{wo} - T_{ci})$
M	Mach number
O/F	oxidizer-fuel mass ratio
Pr	Prandtl number, $\mu c_p/k$
Re	Reynolds number, $\rho V x/\mu$
St	Stanton number, $h/\rho V c_p$
T	temperature
V	velocity
x	axial distance from nozzle exit
η	cooling effectiveness, $(T_g - T_{wo})/(T_g - T_{ci})$

μ viscosity

ρ density

Subscripts:

c coolant

ci coolant inlet

co coolant outlet

g gas

o condition without blowing

wo outside wall

INTRODUCTION

With the increased interest in the United States in supersonic and hypersonic flight, a significant number of technical challenges have surfaced which are critical to the successful development of these high speed flight vehicles. One major concern in advancing the state-of-the-art technologies for hypersonic vehicles is the development of an aeropropulsion system capable of withstanding the sustained high thermal loads expected during flight. Substantial progress has been made on computational understanding the flow physics/chemistry and the resulting aerothermal loads created during high speed flight. However, there is a need for experimental facilities capable of providing a high heat flux environment for testing hypersonic vehicle component concepts and verifying (or calibrating) the analyses.

The high heat loads encountered by the leading edges of a hypersonic aircraft during flight imposes severe demands on the materials and structures. For example, aerodynamic heating at high flight Mach numbers, including the bow-shock/shock interference heating effects on the engine cowl leading edge (refs. 1 and 2), can result in heat flux levels which exceed the capability of most conventionally cooled metallic and potential ceramic materials available for aerospace applications today. As a result, an emphasis has been placed on advancing the technology base to insure the development of unique actively cooled structures capable of withstanding these extreme environmental conditions.

An experimental research effort has been initiated at the NASA Lewis Research Center to compliment the national facilities being developed to study high heat flux in aeropropulsion systems. The focus at NASA Lewis is to assess the capability of actively cooled structures to tolerate the high heating rates typical of hypersonic flight (refs. 3 to 5). The "Hot Gas Facility," a hydrogen/oxygen rocket engine with a 0.9 cm square cross section combustion chamber, provides hot combustion gases up to 3050 K and 4100 kPa to the test articles. A convergent-divergent (C–D) nozzle can be attached to the combustion chamber to achieve supersonic flow of up to Mach 2.5. Test specimens can be mounted in the combustion chamber itself or downstream of the C–D nozzle exhaust to obtain either subsonic or supersonic flow conditions. Heat flux levels up to 450 w/cm^2 on the side wall and up to 5000 w/cm^2 at the stagnation point can be obtained in a supersonic flow field.

Two experimental programs conducted in this facility are discussed in this report. The first is an oxidation/erosion experiment on protective coatings. The objective of the experiment is to investigate the survivability of a coating on a simulated leading edge configuration under a high velocity, hot gas environment. Specifically, two areas are addressed: (1) to determine if high velocity hot gas impingement on leading edges will cause erosion of the heat exchanger face sheet protective coating in a fuel rich (reducing) environment, and (2) to assess the effect of pre-oxidation of the coating on its (rate of) erosion in an oxygen rich (oxidizing) flow.

The second experiment investigates transpiration cooling effectiveness of a compliant rope seal. Moving panels in high speed flight propulsion systems require high temperature capable seals to prevent the high pressure, high temperature gases from damaging the support structure. Therefore, the objective of the transpiration cooled seal effort is to assess the capability of actively cooled compliant seals to protect engine structure in a high temperature, high heat flux environment. This type of seal concept has been shown to be a viable candidate to seal the gap between moveable and static engine walls in the engine flow path. However, more detailed heat transfer data on transpired porous ceramic surfaces are needed to permit accurate design evaluation.

Results from these experiments in a supersonic flow environment are presented. These data will guide the design and development of actively cooled surfaces exposed to high heat flux in aeropropulsion systems.

BACKGROUND

Vehicle flight at hypersonic speeds in the atmosphere at high dynamic pressure presents significant heat transfer challenges throughout the propulsion system and on the airframe because of the high aeroheating loads. Local heat flux levels in the stagnation region of the engine cowl and struts range from 2000 to 10 000 w/cm^2. Very high heat flux occurs in high speed flight when there is a shock-on-shock interaction at the cowl or strut leading edge. As the vehicle accelerates through the atmosphere the airframe shock sweeps across the engine inlet creating an interaction with the bow shock on the engine leading edge component (fig. 1). The physical phenomena are reasonably well understood. Six classes of interaction have been identified (fig. 2). However, the most severe heating occurs with the type IV interference heating phenomena. Holden (ref. 1) and Glass et al. (ref. 2) have measured stagnation point heat transfer augmentation factors of more than 20:1 for type IV interference heating phenomena. Local stagnation heat flux up to 57 kW/cm^2 are predicted under these circumstances. This high heat flux occurs within a narrow band which sweeps across the structure as the vehicle accelerates. A typical Schlieren photograph of a type IV interference pattern is shown in figure 3. Navier-Stokes real gas analysis has been able to capture most of the flow physics and chemistry of this phenomena. However, the challenge is finding methods to reduce the heat load and/or finding cooling schemes and materials that can tolerate this environment.

HOT GAS FACILITY AND OPERATING CHARACTERISTICS

General Description

The Hot Gas Facility can provide Reynolds number, Prandtl number, enthalpy, and heat fluxes similar to those experienced during hypersonic flight without shock interference heating. When operated in the oxygen rich mode, the atmospheric partial pressure of oxygen can also be simulated. Hydrogen/oxygen combustion gases ranging in temperature from about 1300 to about 3050 K at combustion chamber pressures up to 4100 kPa are achievable. The products of combustion are water vapor and either hydrogen or oxygen depending on whether the test is run fuel rich ($O/F < 8$) or oxygen rich ($O/F > 8$). The Prandtl number for these mixtures is in the range of 0.6 to 0.8 which is comparable to air. In addition, the ratio of specific heats is in the range of 1.2 to 1.5 which is also comparable to that of air. Reynolds number similarities can be maintained up to 1 000 000/m. The rocket engine combustion chamber, mounted horizontally, has a square cross section of 0.9 cm. The test stand and the exhaust scrubber tank inlet pipe are shown during firing in figure 4.

Gaseous hydrogen coolant is used for these test articles in the facility. A liquid nitrogen heat exchanger is also available to provide gaseous hydrogen coolant at 90 to 100 K to the test articles. Hydrogen coolant can be supplied to a test specimen at a maximum flow rate of 0.068 kg/sec and a maximum pressure up to 10 000 kPa.

In general, the facility is operated in short bursts of about 5 or 6 sec. However, tests as long as 30 sec have been conducted with no damage to the facility hardware.

A convergent/divergent (C–D) nozzle with a 1.85 area ratio can be added to provide a supersonic Mach number of about 2.0. With a nozzle extension attached, a Mach number of about 2.6 can be attained.

Data Acquisition Capabilities

TRADAR 2.5, a high speed, analog-to-digital recording system is used to record up to 100 data channels at 100 samples per second per channel of time dependent data. Data from pressure, temperature, flow, and strain sensors can be recorded for subsequent computer processing. A 32-channel recording oscillograph is used to provide immediate data feedback on critical data channels in any given test. Twenty parameters can be recorded in this fashion. There are 48 data channels for each of the type K, type E, and type R thermocouples. Sixty-four channels of signal conditioners as well as 12 frequency-to-voltage converters are available for use with pressure transducers. For photographic documentation, the facility has high-speed motion picture cameras which operate at up to 400 frames/sec as well as video and 35-mm remotely-operated cameras. An infrared camera system is used to measure the surface temperature of the Nextel seal material during these experiments.

Leading Edge Configuration

The leading edge coating erosion/oxidation specimens are shown mounted on the exhaust jet center line in figure 5. For these experiments, the C–D nozzle is installed without the nozzle extension to yield a Mach 2.0 condition at the test specimen leading edge. This is an under expanded flow condition. However, it provides similar wall shear conditions as that expected in actual flight.

Metallurgical tests are performed periodically during the thermal cycling experiments to track the behavior of the coating. Two specimens are tested in a fuel rich (reducing) environment and two specimens are tested in an oxygen rich (oxidizing) environment. Each of these specimens are subjected to thermal cycling with a goal of about 150 sec total exposure.

The heating rates on a typical leading edge model are determined from a heat transfer calorimeter specimen fabricated from titanium aluminide. The calibration specimen is subjected to hot gas flow for about 2 sec at an O/F ratio ranging from 1.17 to 1.7. Data from these tests are reduced by solving the transient wall temperature response of a semi-infinite body to a step change in boundary conditions. The stagnation heat flux levels determined by this method are consistent with that predicted by correlations in the literature for cylinders in cross flow.

Seal Configuration

The transpiration cooled seal specimens were mounted on a plate attached to the C–D nozzle extension. This provided a supersonic free stream Mach number of 2.5 (fully expanded) across the test surface. The nozzle exhaust plane and the front surface of the calibration plate is shown in figure 6. This view shows the static pressure instrument array as well as the heat flux sensor array. The tubing from each of the 22 surface pressure taps were of equal length to provide equivalent response times for each measurement. A transducer range of 170 kPa absolute was selected to bracket an expected measurement near atmosphere. Six garden-type heat flux gages were located on the calibration plate centerline in the approximate axial position of the ceramic seal. These gages were calibrated for a heat flux range of 0 to 450 w/cm^2.

An infrared camera is located to the right (not in view in fig. 6) to obtain thermal data of the calibration plate through the exhaust plume. Thermocouples on the plate and a blackbody reference downstream of the plate are used to calibrate the infrared camera system.

The various parts that make up the seal material test fixture are shown in the figure 7. These consist of the nozzle extension, the seal fixture and coolant plenum, and the cover plate forming the flow surface. The coolant flow is metered through a rigid mesh plate on the coolant plenum.

A close up view of some sample ceramic braided material is shown in the figure 8. To fabricate the samples, the Nextel material is initially braided into a hollow rope. This rope is then cut in the axial direction yielding the flat specimens shown. These samples are folded over the fixture and plenum chamber and clamped in place.

The heating rates on the side wall extension are measured by garden type gages. Several tests are conducted at various chamber pressures to determine the optimum operating condition for the seal tests. The optimum condition is defined as the fully expanded flow condition which would minimize an optimization parameter composed of the standard deviation of the measured static pressures and/or heat flux.

RESULTS AND DISCUSSION

Coating Erosion/Oxidation Experiments

All four test articles were tested beyond the original time/cycle goal at erosive and/or oxidizing conditions representative of high speed flight. The coating performed well during the fuel rich erosion tests. No coating cracking or spalling was detected during the thermal shock and thermal cycle tests.

During the oxygen rich tests, coating degradation was observed to take place with increasing thermal cycles. Figure 9 depicts this progressive process with photos taken after each test run in the facility. Coating data are still being evaluated to determine the coating functional characteristics.

Data taken from these tests, in conjunction with data from other national laboratories, are being compiled for use in current design decisions regarding coating development and application on high speed flight hardware.

Transpiration Cooling Experiments

Heat flux distribution.—Four typical heat flux distributions on the calibration plate shown in figure 10 represent the variation with chamber pressure. The heat flux level for the first four axial locations do not vary significantly with chamber pressure. However, there is a significant effect at the last two locations which apparently reflect the degree of over or under expansion of the flow. This effect is the result of an expansion fan propagating across the gas stream from the top edge of the exhaust nozzle. Based on the surface pressure disturbances and the variation in heat flux at the downstream location, the probable optimum operating chamber pressure is 2030 kPa for the data shown (O/F = 2.2). The optimum chamber pressure for an O/F of 1.8 is about 2070 kPa. This is based on the pressure optimization parameter and the heat flux distribution which shows similar trends as the 2.2 O/F data. In addition to the heat flux gages, surface thermocouples were located fore and aft of the heat flux gage locations. The time response of these thermocouples were analytically modeled as semi-infinite solids and used to determine heat flux at these locations. The results compared reasonably well with the heat flux gage data.

Stanton number.—The Stanton number (based on the calibration plate heat flux gage measurements) is shown in the figure 11 and compared with results from the literature. These data compare favorably with data from the literature for flat plate geometry and conditions in the range of Mach 2 to 4. An issue has been raised following the transpiration cooling experiment concerning the gas stream boundary conditions. At this point, at least, the heat flux and recovery gas temperature results compare very well with what was expected. The two primary variables in this correlation are the gas stream temperature and the heat flux. The gas stream temperature was calculated from the O/F ratio measured, a combustion efficiency, and a recovery factor. The heat flux was obtained from calibrated gages. The pattern factor of the gas stream was not considered. However, this should not bias the results significantly.

IR temperature data for seal.—An infrared imaging system was used to obtain temperature data from the ceramic seal material during the 5-sec test sequence. This system was calibrated during the previously discussed calibration phase of the program. A typical image of one instant during the test is shown in figure 12. Data included in the post processing are maximum, minimum, and the mean temperature within a predefined area (such as the whole seal area only or discreet locations within the seal area).

Localized temperatures were also obtained by post processing the infrared temperature data. For instance, defining areas of the seal that represent the axial location of the heat flux gages used in the calibration tests, a correspondence was obtained between uncooled surface heat flux and the cooled surface temperature. A plot of typical temperature profiles is shown in figure 13 for various run times from 2 to 4 sec. The downstream temperature (gage position 6) is not plotted since the material did not respond at this location. These data can also be cross plotted as time histories as required. Cross plotting these data would show an increase in temperature to about 3 sec then a drop in the surface temperature at 3.5 sec followed by a continuing increase. This second-order phenomena indicates a more complex heat transfer phenomena then initially considered. Also, 5 sec may not be sufficient to reach steady state temperature at the surface.

Nextel seal cooling effectiveness.—An average surface effectiveness is shown in figure 14 and is based on the overall average seal surface temperature at about 4.3 sec into the test. These temperature data, initially processed assuming a surface emissivity of 0.91, are corrected to an emissivity of 0.87. Data are included for both GN_2 and GH_2 at ambient as well as cryogenic temperatures. Also included on the figure is a transpiration cooling correlation based on a modification of work from Moffat and Kays (ref. 6). A modification of the heat balance has been combined with the theoretical correlation of Mickley-Spalding (refs. 7 and 8) to account for difference between the coolant exit temperature and the outer material surface temperature that would exist in actuality (heat exchanger efficiency less than one).

$$\eta = \frac{\left(\frac{c_{pc}}{c_{pg}}\right) K' \left[e^{(\rho V_c / \rho V_g St_o)} - 1 \right]}{1 + \left(\frac{c_{pc}}{c_{pg}}\right) K' \left[e^{(\rho V_c / \rho V_g St_o)} - 1 \right]}$$

Intuitively, this efficiency parameter (K') could be high at low blowing rates and the parameter would be low at high blowing rates. This type intuition is included in the modified correlation shown where K' is temperature difference ratio efficiency parameter. The data are shown to correlate reasonably well between hydrogen and nitrogen gases and for both cryogenic and ambient coolant temperatures. However, these data do not correlate well with that expected from results in the literature. There is some correlation of the data with the slope of the modified correlation at high blowing rates. But at the low blowing rates, these data do not correlate which raises issues that are discussed below.

Several issues have been raised since the results from the seals experiment do not meet expectations based on similar experiments. These issues are summarized as follows: (1) the gas stream conditions are not sufficiently well known (i.e., the effective gas temperature is less then that calculated); (2) shocks and flow separation adversely affect the boundary conditions relative to the calibration data; (3) the IR pyrometer does not sense the true steady state seal surface temperature because of: surface roughness or waviness, the unknown transparency of the gas stream, an unknown Nextel material reflectivity and emissivity; (4) high surface temperature gradient within the 0.02 by 0.027 pixel area biased results to lower temperature; and (5) the seal has not reached steady state during the 5-sec test time. Each of these issues are being addressed systematically.

SUMMARY OF RESULTS

Two high heat flux experiments related to high speed flight are discussed. For the leading edge coating erosion/oxidation experiments, all four test articles were tested beyond the original time/cycle goal at conditions representative of high speed flight. The coating performed well during the fuel rich erosion tests. No coating cracking or spalling was detected during the thermal shock and thermal cycle tests. During the oxygen rich tests, the coating degraded progressively throughout the test series.

For the transpiration cooling experiments on a porous ceramic material (Nextel), the heat transfer results (~55 data points) did not meet expectations and, consequently, raises some issues concerning the validity and interpretation of the calibration results as well as the measurement of seal temperature and gas temperature during the cooling experiments. Once these issues are resolved though, these data present an opportunity to initiate a design procedure based on realistic data. A correlation based on a modified Mickley-Spalding is proposed that predicts the trends in the high blowing rate data. However, the low blowing rate data do not correlate.

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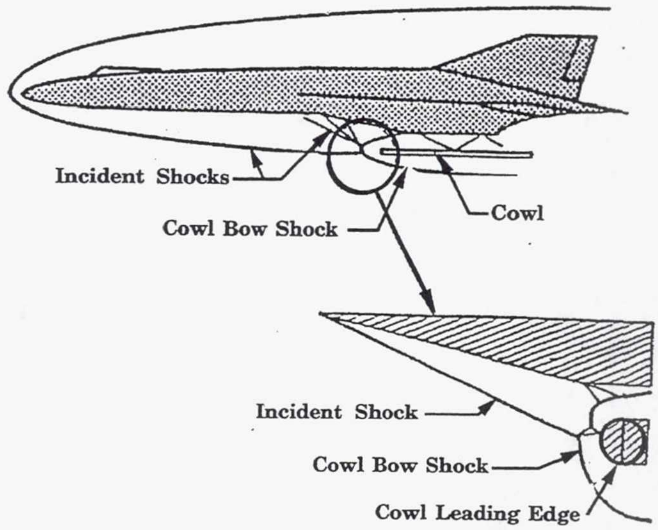


Figure 1.—Shock-shock interference heating at leading edge.

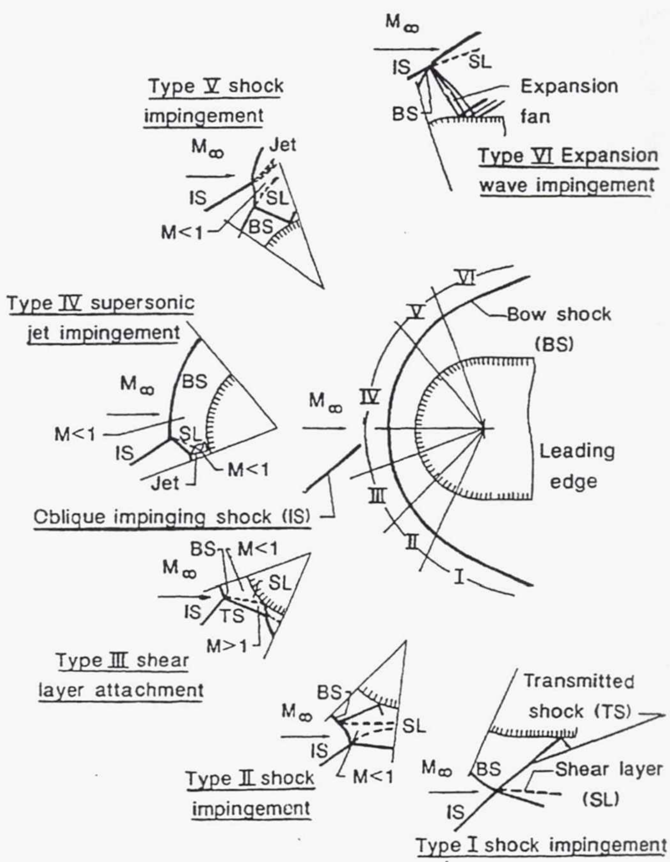


Figure 2.—Six types of shock interference patterns and their location.

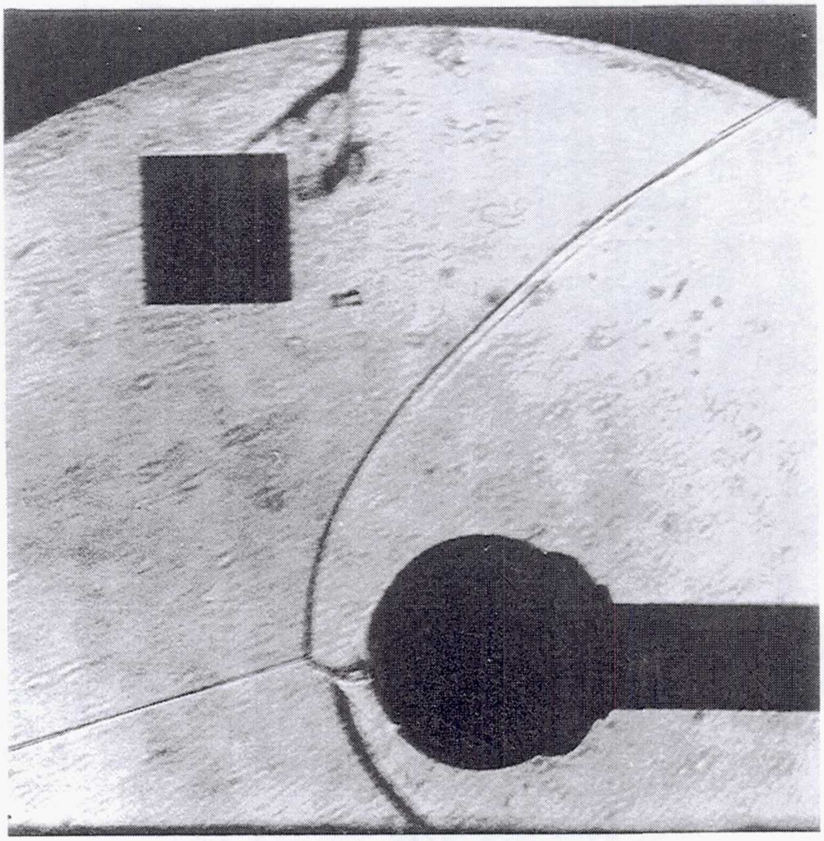


Figure 3.—Schlieren photograph of type IV shock pattern.

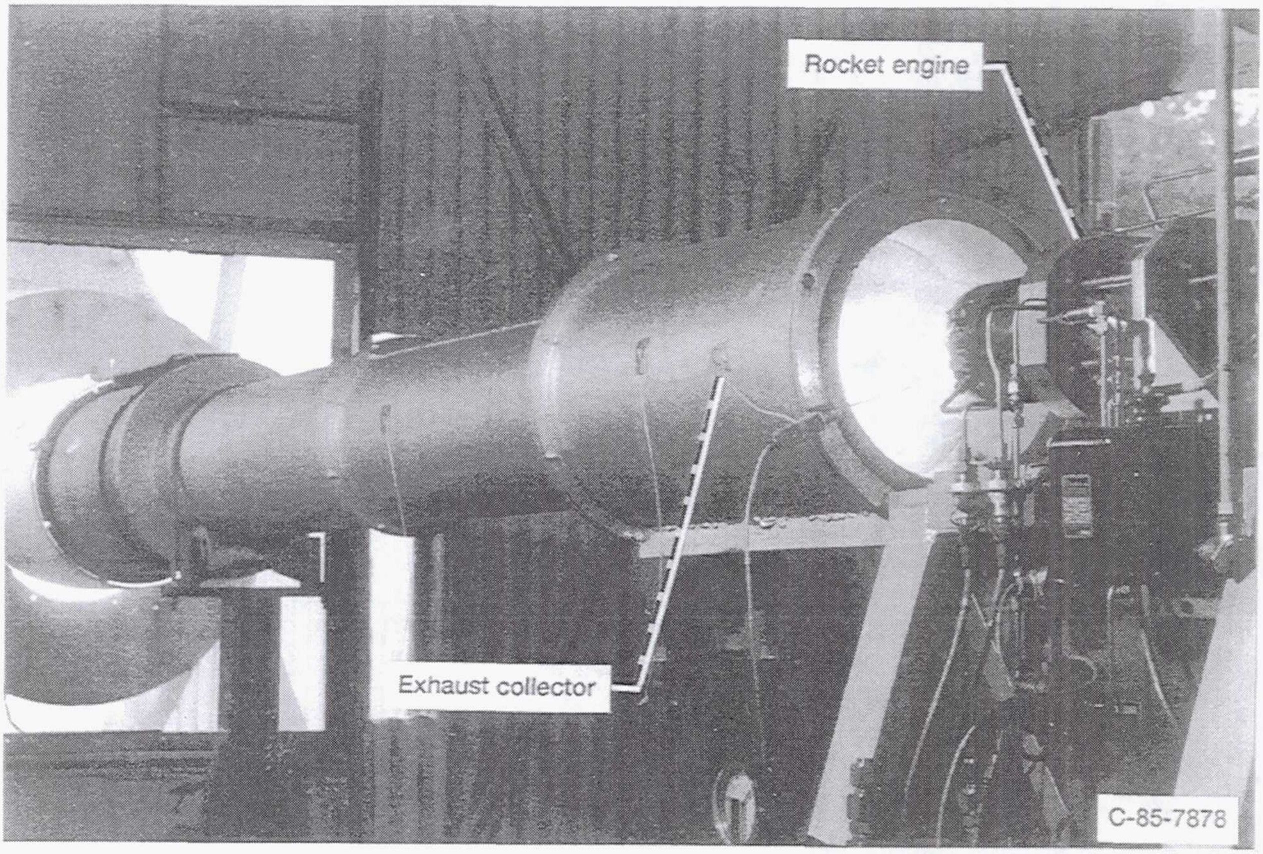


Figure 4.—Hot Gas Facility during firing of the rocket engine.

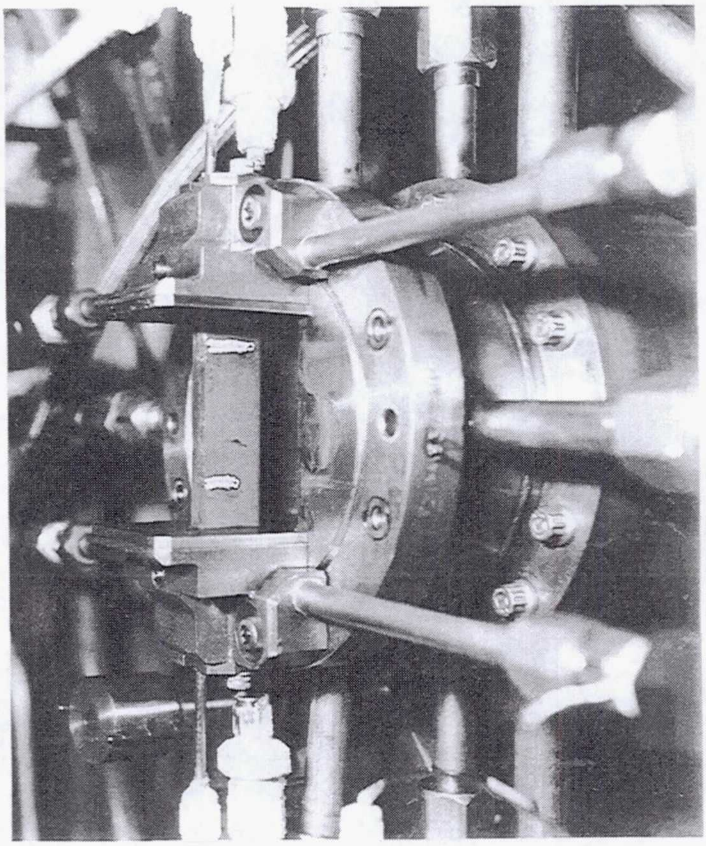


Figure 5.—Coated leading edge specimen mounted at C-D nozzle exhaust plane.

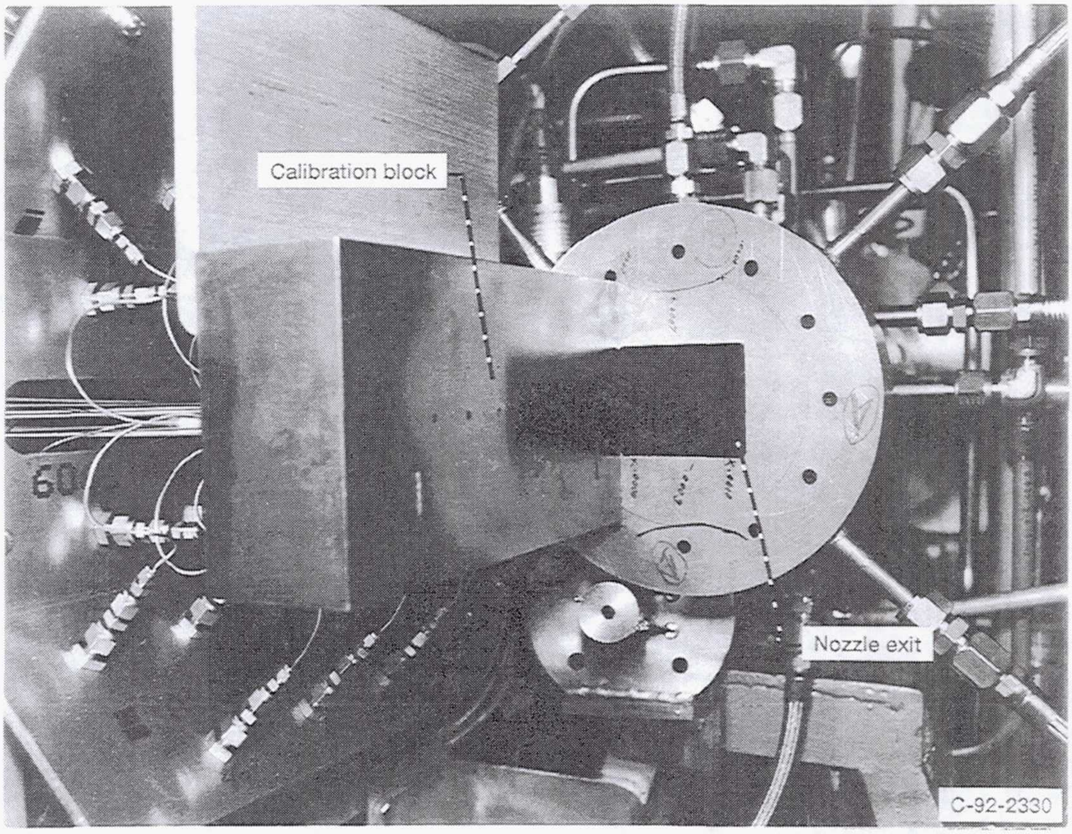


Figure 6.—Front of calibration plate mounted on side of nozzle extension.

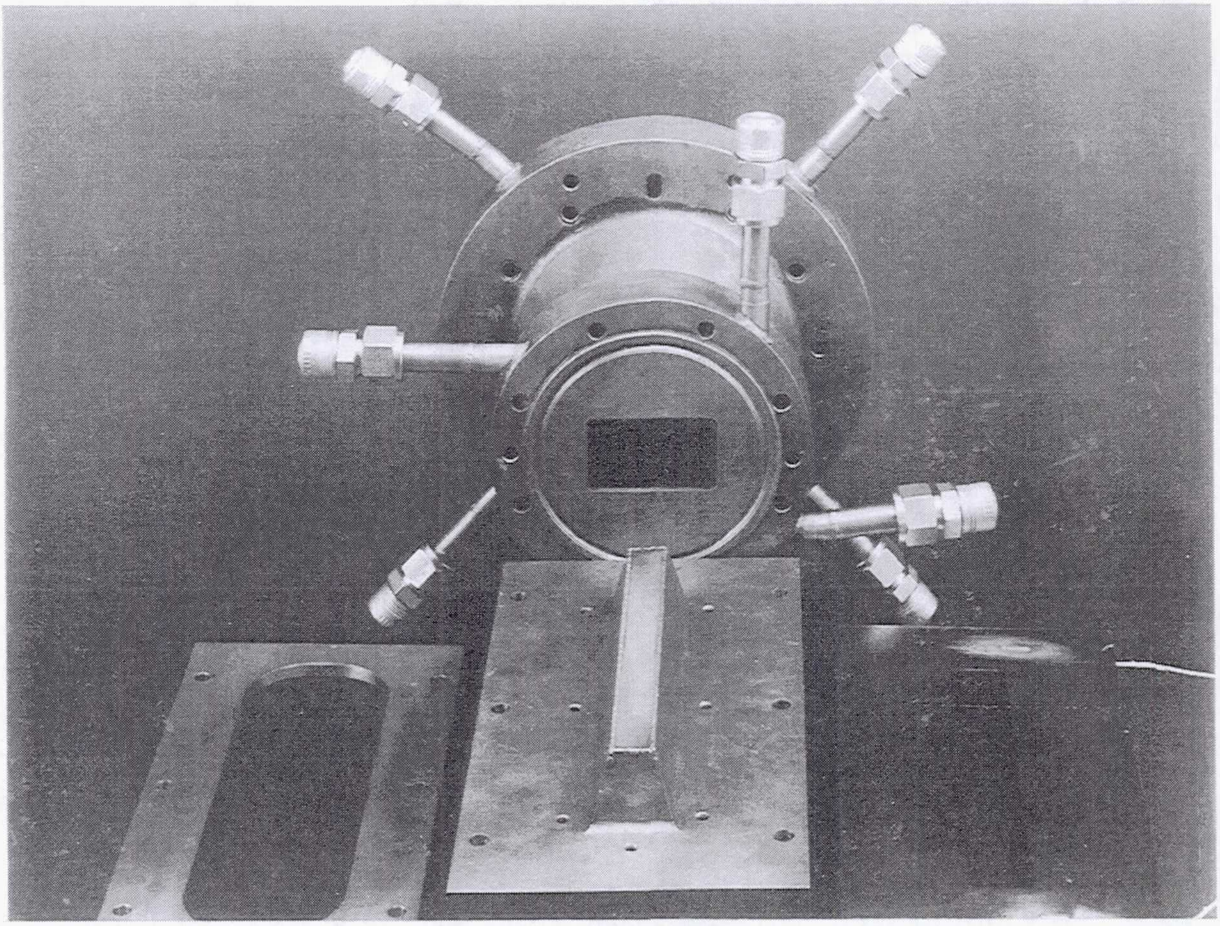


Figure 7.—Transpiration cooling test hardware.

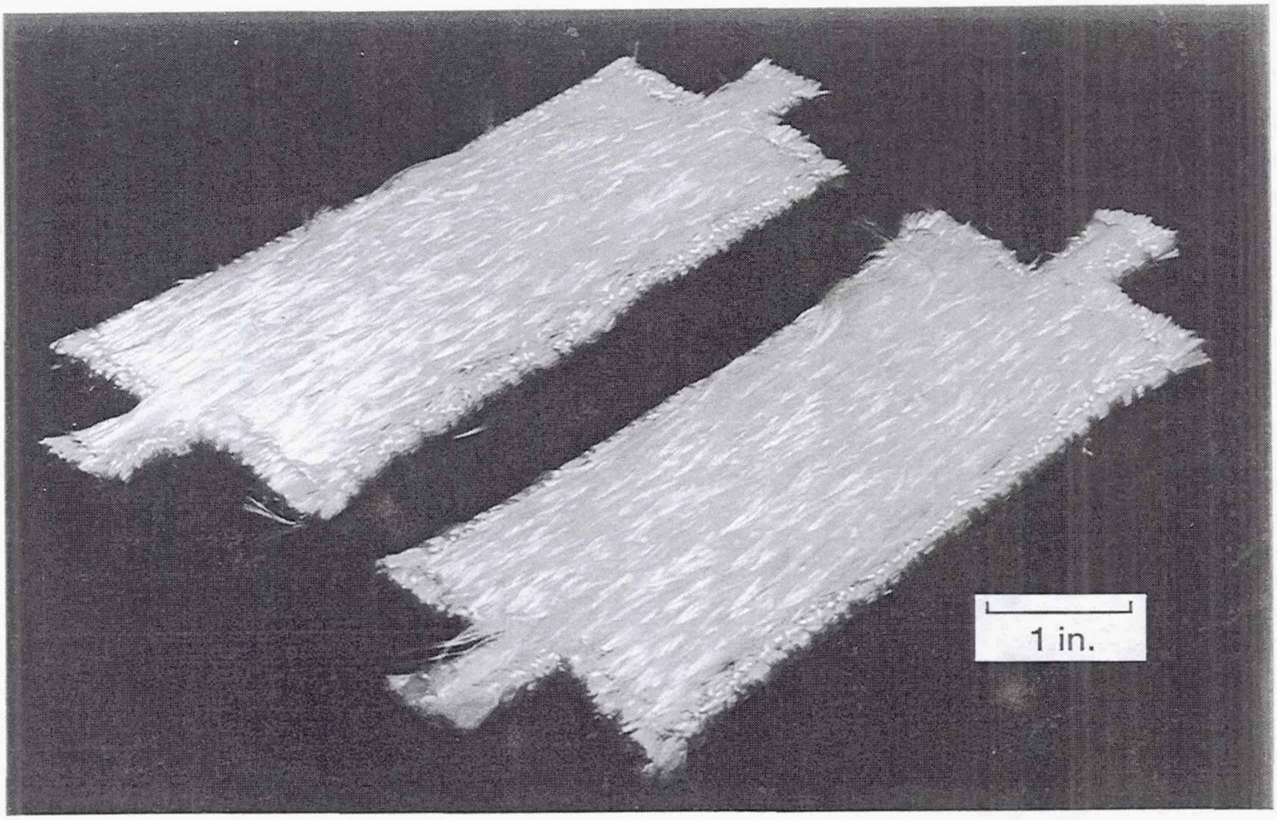
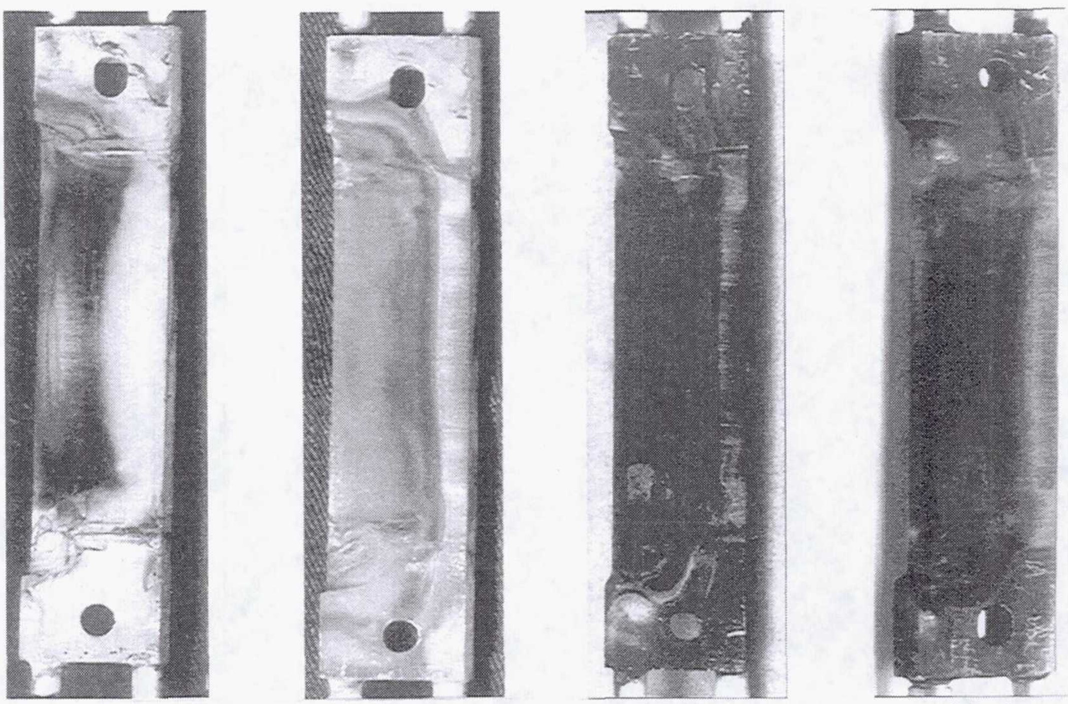


Figure 8.—Braided Nextel seal material.



1 Cycle
1 Second

13 Cycles
57 Seconds

29 Cycles
150 Seconds

49 Cycles
270 Seconds

Figure 9.—Macro photographs taken during oxygen rich tests shows coating degradation.

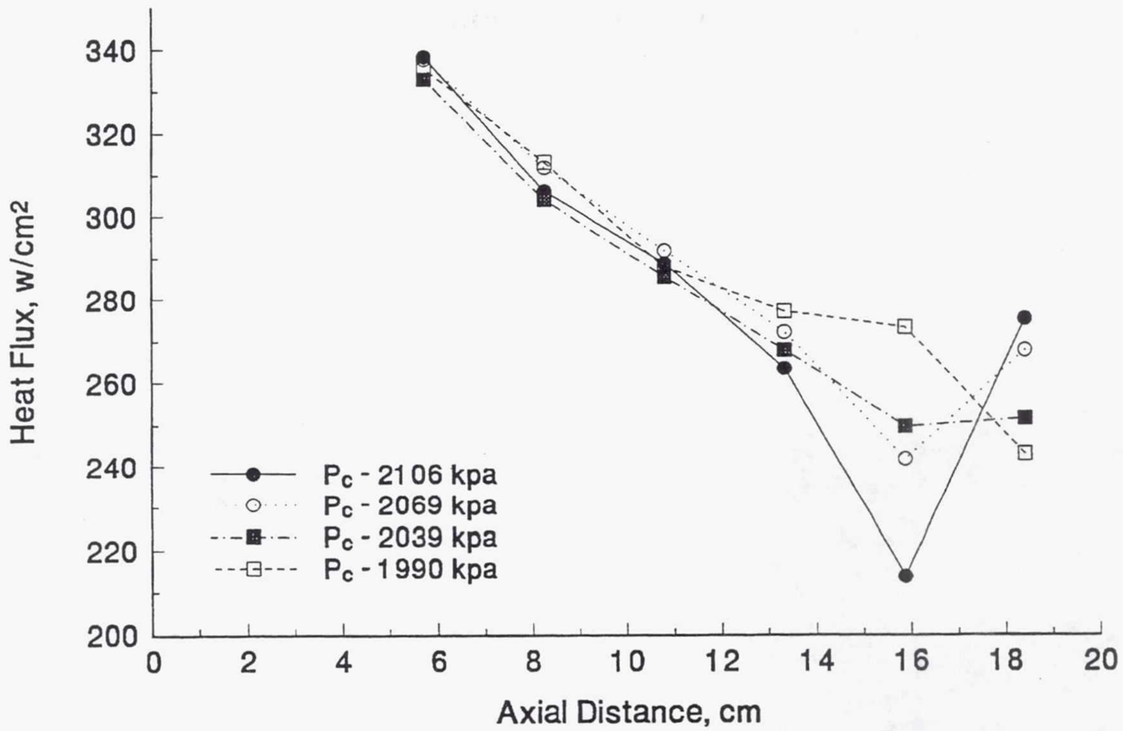


Figure 10.—Heat flux distribution on calibration plate as a function of chamber pressure. Note over/under expansion effects downstream.

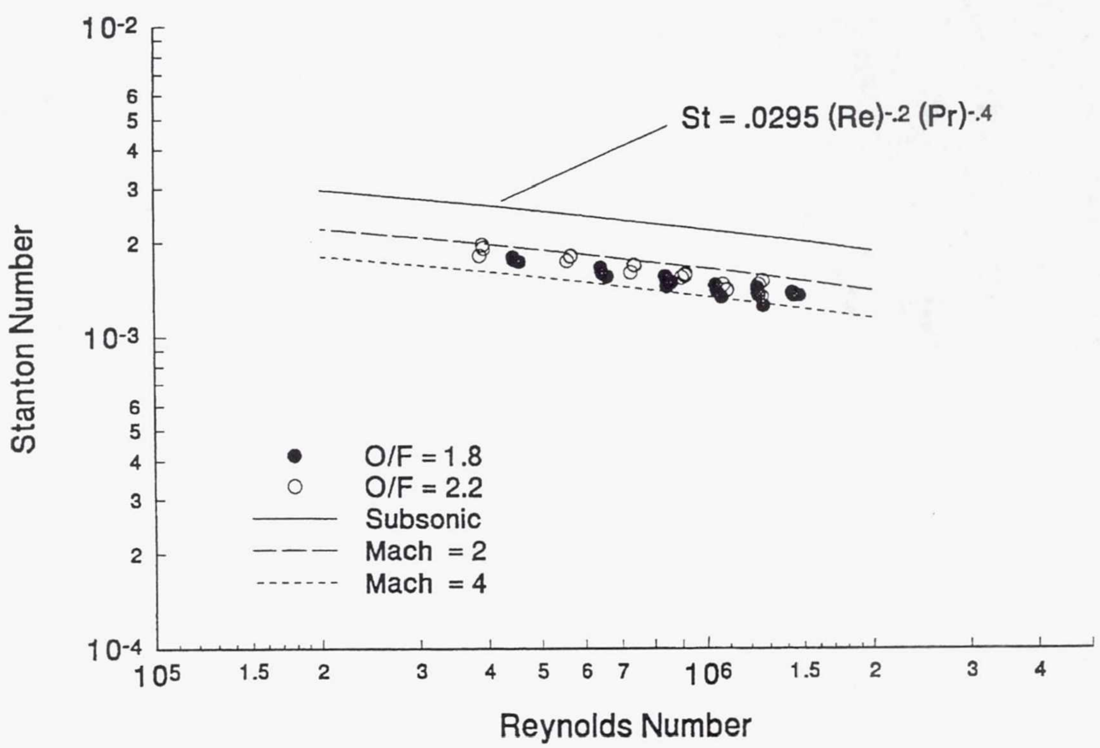


Figure 11.—Stanton number distribution on the calibration plate.

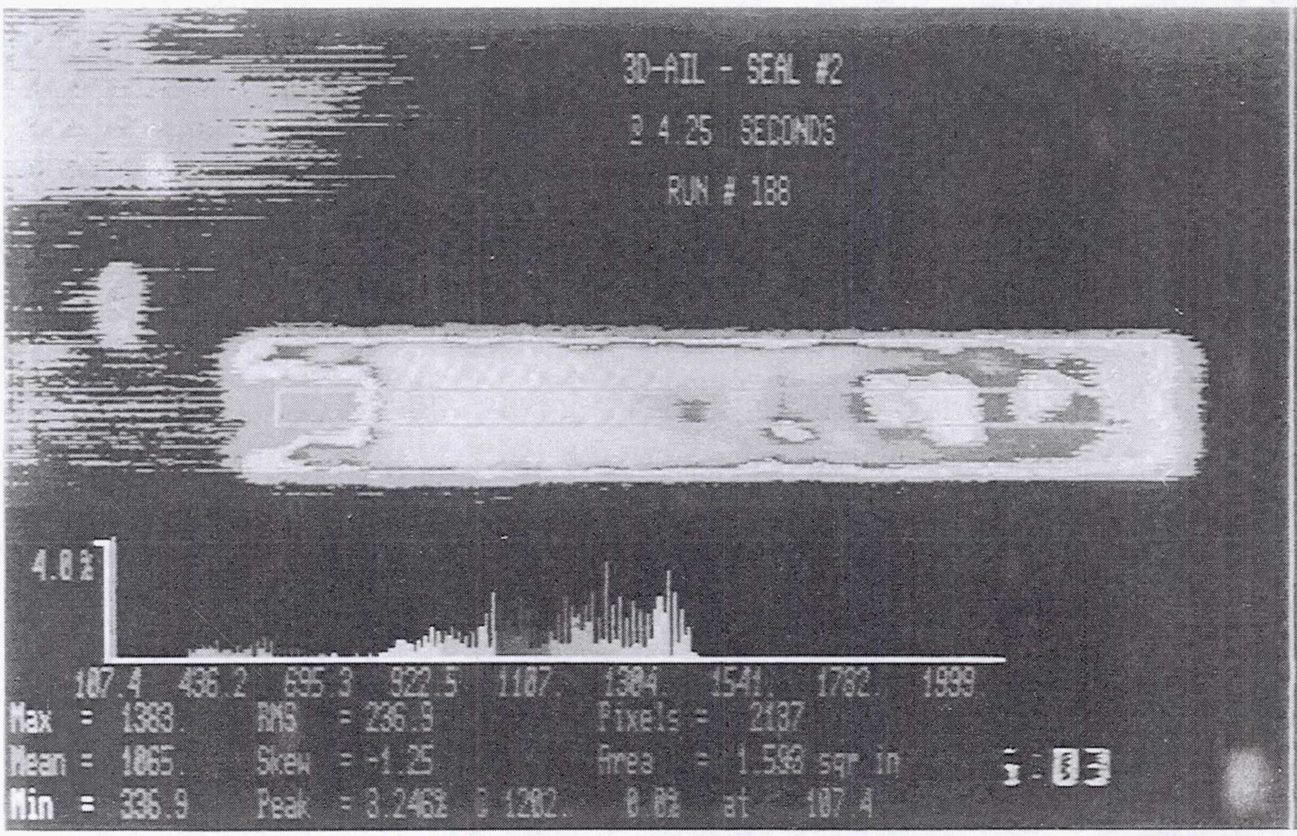


Figure 12.—Infrared seal temperature photograph.

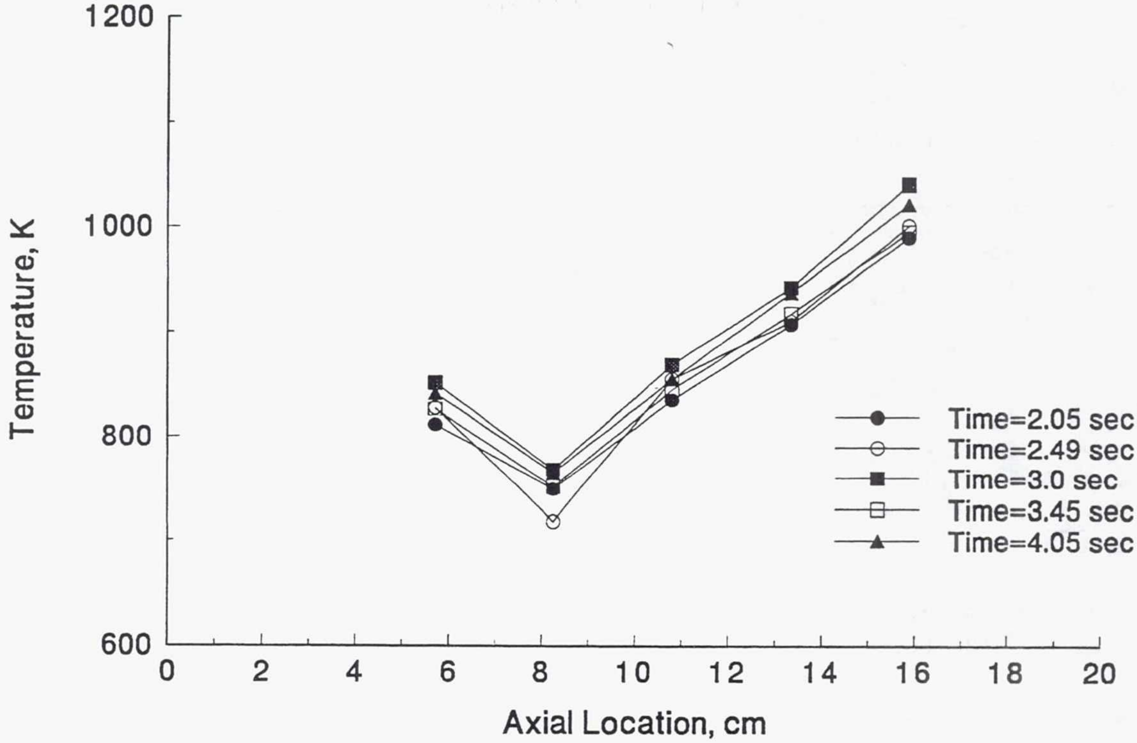


Figure 13.—Infrared temperature data as a function of time and location.

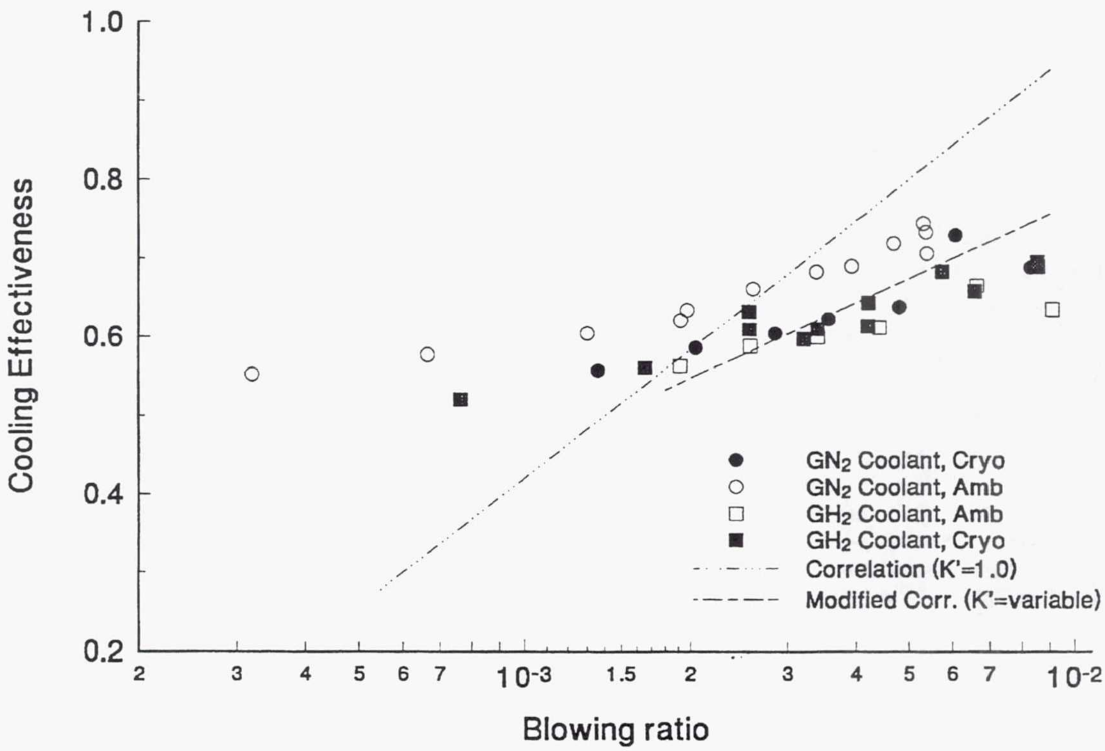


Figure 14.—Transpired cooling effectiveness of Nextel seal material.

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13. ABSTRACT ***(Maximum 200 words)***

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