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**HEAT PIPE RADIATION COOLING (HPRC)
FOR HIGH-SPEED
AIRCRAFT PROPULSION**

LOS ALAMOS
National Laboratory

**HEAT PIPE RADIATION COOLING (HPRC)
FOR
HIGH-SPEED AIRCRAFT PROPULSION**

Phase II (Feasibility) Final Report

**Prepared for
NASA Lewis Research Center
Cleveland, OH 44135**

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By

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ABSTRACT

The National Aeronautics and Space Administration (NASA), Los Alamos National Laboratory (Los Alamos), and CCS Associates are conducting the Heat Pipe Radiation Cooling (HPRC) for High-Speed Aircraft Propulsion program to determine the advantages and demonstrate the feasibility of using high-temperature heat pipes to cool hypersonic engine components. This innovative approach involves using heat pipes to transport heat away from the combustor, nozzle, or inlet regions, and to reject it to the environment by thermal radiation from adjacent external surfaces. HPRC is viewed as an alternative (or complementary) cooling technique to the use of pumped cryogenic or endothermic fuels to provide regenerative fuel or air cooling of the hot surfaces. Above Mach 3 engine cooling requirements generally exceed the thermal capacity of hydrocarbon fuels. For HPRC-cooled propulsion systems it is possible to continue to use hydrocarbon fuels into the Mach 4 to Mach 6 speed range, a potential technology breakthrough.

The HPRC program has been conducted through two phases, an applications phase and a feasibility phase. The applications program (Phase I) included concept and assessment analyses using hypersonic engine data obtained from U.S. engine company contacts. The applications phase culminated with planning for experimental verification of the HPRC concept to be pursued in a feasibility program.

The feasibility program (Phase II), recently completed and summarized in this report, involved both analytical and experimental studies. Specific tasks included a continuation of the HPRC concept analyses, test article design, test article fabrication, test facility design and preparation, testing and data analysis, feasibility assessment, management activities, and technology transfer to industry. Meetings with General Electric Aircraft Engines (GEAE) and United Technologies Pratt and Whitney (P&W) during the feasibility program have produced additional ideas for potential heat pipe applications as discussed herein.

During the feasibility program, preliminary HPRC/HiMaTE integration studies were performed. Integration of HPRC with HiMaTE has the potential to eliminate the need for endothermic fuels to cool the engine, but not enough information was available for a quantitative analysis. It was found that the application of thermal insulation to ramburner and nozzle walls can reduce the heat load by about one-half and reduce peak HPRC system temperatures to below 2700°F. It was also found that operation of HPRC at cruise conditions of around Mach 4.5 and an altitude of 90,000 ft lowers peak hot section temperatures to around 2800°F. An HPRC heat pipe was successfully fabricated and tested at Mach 5 conditions of heat flux, heat load, and temperature.

Assessments of the analytical and experimental results from the feasibility program continue to indicate that the HPRC concept is feasible. Therefore, a follow-on technology program (Phase III) is recommended. Successful development of HPRC in a technology program can enable a major technological breakthrough that will enhance the economic attractiveness of commercial or military hypersonic flight.

1.0 INTRODUCTION

1.1 Background

A number of propulsion system critical component technologies for a long-range aircraft capable of taking off at sea level and climbing to hypersonic (above Mach 3) cruise conditions at up to 100,000 feet altitude are identified.¹ These technologies, including engine cycle, cooling method, fuels, and materials, are being developed by private industry and the U.S. Government in programs such as HiSPA and HiMaTE. Integrated, combined-cycle engines such as the turbofanramjet are favored, as opposed to multiple engine types for different speed regimes.^{2,3,4,5,6}

The influence of Mach number (or temperature) on such technologies is shown in Fig. 1. At Mach numbers above Mach 3, the elevated recovery temperature of decelerated, compressed air in the engine inlet results in turbojet engine temperatures that exceed current material temperature limits on rotating machinery. With sufficient ram compression of air at speeds above Mach 3, it is possible to utilize a ramjet engine in which there is no compressor or turbine⁷. However, the hot section (ramburner plus nozzle) materials still require considerable thermal protection in the 5,000 Btu/s to 15,000 Btu/s range to maintain structural integrity. Persistent high temperatures dictate the use of specialized structural materials and coatings that can withstand reactive environments.⁸

Because of the ram temperature rise, air is no longer viable as the primary coolant, so the engine fuel is used either directly as coolant or indirectly as the heat sink in an air/fuel heat exchanger. Above Mach 4 standard JP-grade hydrocarbon (HC) fuels no longer have

^{1,2}Watts, J. D., et al., "Mach 5 Cruise Aircraft Research," Langley Symposium on Aerodynamics, Vol. II, NASA Conference Publication 2398, April 1985, pp. 285-304.

³Kors, D. L., "Combined Cycle Propulsion for Hypersonic Flight," IAF-87263, 38th Congress of International Astronautical Federation, Brighton, United Kingdom, October 1987.

⁴Anderson, G. Y., Bencze, D. P., and Sanders, B. W., "Ground Tests Confirm the Promise of Hypersonic Propulsion," Aerospace America, Vol. 25, No. 9, September 1987, pp. 38-42.

⁵Kunkler, H. and Kuczera, H., "Turbo-Ramjet Propulsion System Concepts for Future European Space Transport (SANGER)," IAF-87-265, 38th Congress of the International Astronautical Federation, Brighton, United Kingdom, October 1987.

⁶Tanatsugu, N., et al., "Analytical Study of Space Plane Powered by Air-Turbo Ramjet with Intake Cooler," IAF-87-264, 39th Congress of the International Astronautical Federation, Brighton, United Kingdom, October 1987.

⁷Hill, P. G. and Peterson, C. R., Mechanics and Thermodynamics of Propulsion, Addison-Wesley, Reading, MA, 1970.

⁸Kors, D. L., "Combined Cycle Propulsion for Hypersonic Flight," IAF-87263, 38th Congress of International Astronautical Federation, Brighton, United Kingdom, October 1987.

sufficient heat capacity (prior to thermal decomposition) to meet the cooling needs of both the aircraft and engine. To achieve hypersonic speeds, it seems necessary to utilize cryogenic fuels such as methane (LNG) or hydrogen (LH₂), or endothermic fuels such as methylcyclohexane (MCH). Such fuels are much more expensive than HC fuels and have special handling and storage requirements. Complex thermal management design studies have shown that cryogenic or endothermic fuel cooling systems require high pressures, heavy and potentially hazardous fuel/air heat exchangers and fuel reactors, and elaborate pumping and distribution passages.^{9,10,11} The use of such fuel cooling systems strongly impacts the propulsion system design and implies extremely costly, new, on-ground fuel handling infrastructures.¹²

The National Aeronautics and Space Administration (NASA), Los Alamos National Laboratory, and CCS Associates, in cooperation with U. S. aircraft engine manufacturing companies, have undertaken a program to assess and develop a new Mach 4 to Mach 6 aircraft propulsion system cooling concept called Heat Pipe Radiation Cooling (HPRC).¹³ Previously, liquid metal heat pipe cooling systems were investigated for the combustion liner and engine inlet leading edges of scramjet engines for missile applications.¹⁴ The ultimate objective of the HPRC program is to bring this concept to a state of technology readiness for use by the aerospace industry.

1.2 HPRC Concept

HPRC is a technique for passively cooling the hot section of a hypersonic engine, which eliminates the need to use the engine fuel as the coolant. For an HPRC system application at Mach 5, the entire engine heat load, or about two-thirds of the total

⁹Hartsel, J. E., "High Speed Propulsion Assessment: HiSPA Program Summary," General Electric Co., Cincinnati, OH, August 1988.

¹⁰Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

¹¹Glickstein, M. R., "High Speed Propulsion Assessment: Thermal Management Analysis," United Technologies Pratt & Whitney, W. Palm Beach, FL, November 1986.

¹²Witcofski, R. D., "Comparison of Alternate Fuels for Aircraft," NASA TM-80155, NASA Langley Research Center, Hampton, VA, September 1979.

¹³Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion," Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

¹⁴Silverstein, C. C., "Heat Pipe Cooling for Scramjet Engines," CR 4036, NASA Langley Research Center, Hampton, VA, December 1986.

Fig. 1 Influence of Mach number (or temperature) on hypersonic propulsion system design (from Ref. 1)

hypersonic aircraft plus engine heat load, is removed from the hot section. This heat is transported by a surrounding, high-temperature, heat pipe nacelle structure to nearby external surfaces, and rejected to the environment by thermal radiation as shown schematically in Fig. 2.

As indicated in Fig. 2, a portion of the engine nacelle consists of built-in heat pipe structures designed to remove heat from between the engines (or between an engine and the wing or fuselage), and to move this heat to a side- or bottom-facing surface for rejection to the environment by radiation heat transfer. In this configuration, the cylindrical ramburner and nozzle walls are fabricated from special high-temperature materials, such as (but not necessarily) carbon-fiber-reinforced carbon matrix (carbon-carbon)¹⁵ or silicon-carbide-whisker reinforced molybdenum disilicide matrix composites.¹⁶ These walls receive heat from the combustion and exhaust gases by convection and radiation heat transfer and are allowed to achieve temperatures on the order of 2800°F or higher depending on advances in the state of the art.

In Fig. 2 the ramburner and nozzle (hot section) are surrounded by a nacelle of square cross section, which is fabricated from flat heat pipe panels. The innerside, and bottom panels constitute one independent, L-shaped, cooling subsystem, and the top and outside panels constitute a second subsystem. Since the hot section is assumed to be part of a two-engine module that is mounted to the wing or fuselage underside, the interior side and upper heat pipe nacelle panels must be insulated. In Sec. 3.7 it will be seen that the HPRC test article consists of one scaled element from one of these L-shaped subsystems.

The ramburner and nozzle walls radiate the engine heat load to the surrounding HPRC portion of the nacelle, which transports the heat load isothermally to side- or bottom-facing heat pipe surfaces. These surfaces lose heat to the environment by convection and radiation heat transfer at temperatures from 2100°F to 2400°F. The required amount of heat pipe nacelle radiator exposed surface area will depend on the specifics of a particular propulsion system cooling design. Design options include restricting the radiator, in length, to the ramburner plus nozzle region or extending it forward on the nacelle or outboard under the wing.

The remaining aircraft heat load, not accounted for by the HPRC system, is absorbed by JP-grade HC fuel prior to combustion. Thus, HPRC is proposed as a cost-

¹⁵Sheehan, J. E., "Oxidation-Resistant Carbon-Carbon Composites," Engineered Materials Handbook, Am. Soc. for Metals International, Metals Park, OH, 1987, pp. 920-921.

¹⁶Carter, D. H., "SiC Whisker-Reinforced MoSi₂," Los Alamos National Laboratory report LA-11411-T, September 1988.

HEAT PIPE RADIATOR CONCEPT: TRANSVERSE TRANSPORT - NACELLE SIDE REJECTION

Los Alamos
HEAT PIPE RADIATOR CONCEPT: TRANSVERSE TRANSPORT - NACELLE SIDE REJECTION - MARTIN - MEE13 - LOS ALAMOS - SEPT89

Fig. 2 HPRC concept

saving alternative, or complementary cooling technique, to the use of expensive, pumped cryogenic or endothermic fuels to provide regenerative fuel or air cooling of hot surfaces. By applying HPRC the preferred use of HC fuels is extended into the Mach 4 to Mach 6 speed range.

An HPRC system is conceptually simple. It requires no pumping and operates at low pressure (around 10 psia). Also, because heat pipes operate nearly isothermally, thermal stresses are reduced, and engine structural designs are simplified. High-strength materials and oxidation-resistant coatings are used to fabricate the HPRC heat pipe nacelle as well as hot-section components. Hybrid thermal management systems using heat pipes combined with cryogenic or endothermic fuel cooling are possible.

The HPRC concept is innovative and visionary. It has the potential for a major technology breakthrough in the acceptance and economic viability of hypersonic flight. As such, HPRC is consistent with the NASA Lewis Research Center vision "*to be the Nation's pathfinder in innovative aerospace propulsion R&T.*"¹⁷

1.3 Review of HPRC Applications Phase (Phase I) Results

A detailed description of the HPRC concept and results from the first (applications) phase of the HPRC program are presented elsewhere.¹⁸ During the applications phase, information on configurations, dimensions, temperatures, and cooling loads of hypersonic engines was obtained through discussions with personnel from engine manufacturing companies. This information defines the representative engine.^{19,20,21}

Heat transfer studies were carried out during the applications phase to determine cooling system and hot-section temperatures, heat loads, and weights for the representative engine cruising at Mach 5 at 80,000 ft. The studies included cooling system requirements for both the ramjet combustor and nozzle. A heat transfer computer code was developed to

¹⁷Saunders, N. T. & Glassman, A. J., Lewis Aeropropulsion Technology: Remembering the Past and Challenging the Future, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, p. 1-11.

¹⁸Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion," Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

¹⁹Hartsel, J. E., "High Speed Propulsion Assessment: HiSPA Program Summary," General Electric Co., Cincinnati, OH, August 1988.

²⁰Hartsel, J. E., "High Speed Propulsion Assessment: HISPA Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

²¹Glickstein, M. R., "High Speed Propulsion Assessment: Thermal Management Analysis," United Technologies Pratt & Whitney, W. Palm Beach, FL, November 1986.

facilitate these calculations. The capability of heat pipe internal fluid transport designs to meet HPRC cooling system heat transport requirements was assessed using results of Los Alamos heat pipe code, HTPIPE,²² calculations.

In the previous applications phase, it was found that application of the HPRC concept would result in reasonable sizes and weights, but relatively high material temperatures up to Mach 5 flight speeds. These preliminary studies suggested that the engine ramburner and nozzle walls be fabricated from a material such as carbon-carbon that could operate at temperatures in excess of 3000°F. However, it is shown in this report that the engine ramburner and nozzle wall temperatures can operate at 2800°F or lower, depending on cruise conditions and HPRC design specifics. This temperature is more consistent with short term advanced high-temperature materials goals.

1.4 Scope of HPRC Feasibility Phase (Phase II)

Analytical and experimental results from the feasibility phase (Phase II) of the HPRC program are reported here. Specific tasks included further investigation of HPRC systems for additional design and operating conditions, test article design, test article fabrication, test facility design and preparation, testing and data analysis, feasibility assessment, management activities, and technology transfer to industry. Meetings with General Electric Aircraft Engines (GEAE) and United Technologies Pratt and Whitney (P&W) during the feasibility program produced additional ideas for potential heat pipe applications (See Sec 2.1.2).

2.0 CONCEPT ANALYSES

2.1. Integration of HPRC with HiMaTE

HiMaTE (High Mach Number Technology Engine) is a joint NASA-Air Force program for demonstrating critical component technologies required for a Mach 4 to 6 hypersonic engine.²³ GEAE and P & W are the prime contractors.

The evolution of a viable thermal management system is one of the critical component technologies for hypersonic flight. The HiMaTE Program is evaluating the role that endothermic fuels can play in such a system.

With HPRC, endothermic fuels would not be required for engine thermal management at flight speeds up to Mach 5. Their high thermal capacity would no longer be needed to

²²Woloshun, K. A., Merrigan, M. A., and Best E. D., "HTPIPE: A Steady-State Heat Pipe Analysis Program, A User's Manual, "Los Alamos National Laboratory Manual LA-11324-M, 1988.

²³Rhode, J. E., Overview of Hypersonic/Transatmospheric Vehicle Propulsion Technology, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, pp. 6-1 to 6-11.

absorb the engine hot section heat load. Cheaper and more readily available JP-7 fuel could then be used. This ability to use JP-type fuels for hypersonic propulsion is the most important characteristic of HPRC, and has major implications for economic, and hence commercially viable, hypersonic flight.

2.1.1 Engine Company Visits. To learn more about HiMaTE, its possible integration with HPRC, and provide for co-ordination of efforts, NASA, LANL, and CCS Associates visited the prime contractors on the following dates.

GEAE	March 7, 1991
P & W	May 14, 1991

These meetings provided a useful exchange of information on HPRC and HiMaTE. Visits were also made to both contractors during the prior HPRC contract. During these visits, detailed information was obtained on many aspects of HiSPA, an earlier Air Force-sponsored study of hypersonic engines in the Mach 4 to Mach 6 speed range.^{24,25,26} Information including engine configurations, dimensions, air flow rates, and engine pressures and temperatures for various cruise conditions was obtained. It was then possible to analyze and evaluate an HPRC system for the representative turbofanramjet engine.

In order to determine how effectively the HPRC concept could integrate with HiMaTE, similar information about a representative HiMaTE engine would be required. Information of this type was not obtained. General Electric supplied a schematic of one engine configuration (see Fig. 3), but no dimensions or other engine design and performance information. General Electric also supplied limited heat flux data in which the effect of thermal radiation from the combustion gas to the ramburner and nozzle walls was included. The reason for the discrepancy in availability of engine information is probably a result of the relative maturity of the HiSPA and HiMaTE programs. HiSPA information was already available because the program had been completed and documented prior to the NASA/LANL/CCS visits. On the other hand, the HiMaTE studies had been under way for less than six months at the time General Electric and Pratt & Whitney were visited. For these reasons, lack of needed

²⁴Hartsel, J. E., "High Speed Propulsion Assessment: HiSPA Program Summary," General Electric Co., Cincinnati, OH, August 1988.

²⁵Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

²⁶Glickstein, M. R., "High Speed Propulsion Assessment: Thermal Management Analysis," United Technologies Pratt & Whitney, W. Palm Beach, FL, November 1986.

information prevented a quantitative assessment of the effectiveness of HPRC as an overall thermal management technique for HiMaTE. However, based on the information which is made available, it is possible to make some qualitative observations about HPRC/HiMaTE integration.

Fig. 3 GEAE HiMaTE engine concept

2.1.2 Qualitative Observations. A GEAE schematic drawing of the elements of a HiMaTE engine is shown in Fig. 3. Although the sketch is not to scale, the following significant points relevant to HPRC integration can be made.

1. The augmentor/hyperburner (ramburner) where combustion occurs in ramjet mode appears to be quite short. The length in the sketch is about 25% of the diameter.
2. A Single Expansion Ramp Nozzle (SERN) is used. This nozzle does not utilize a lower surface, and the sidewalls are highly abbreviated.
3. Both of these design features mean that the hot section surface area subject to heating by the combustion gas is reduced substantially compared to the hot sections which were considered in the earlier HiSPA studies. Further, the absence of a lower nozzle surface markedly increases direct radiative heat

transfer from the nozzle to the environment. Consequently, the total heat load to the hot section is reduced compared to HiSPA engines (possibly by as much as 50% or more based on visual examination of Fig. 3), and a significant fraction of this heat load can be dissipated by direct radiation from the nozzle. Because of the lower heat load, an HPRC system for HiMaTE engines will be smaller, lighter, and will operate at lower temperatures than was the case for HPRC systems designed for HiSPA-based engines.

4. If HPRC is used for the entire hot section, the enclosing heat pipe nacelle may be "sculpted" in the nozzle region to avoid blocking the view of radiation leaving the upper surface of the nozzle. The size and shape of the nacelle walls in the nozzle region may then roughly parallel those of the nozzle sidewalls.
5. Depending on the nozzle heat load distribution, it may be more effective to use an HPRC system to cool the ramburner and nozzle throat regions only, with auxiliary conventional cooling methods used for the remaining nozzle heat load.
6. The addition of HPRC can provide all or most of the hot section cooling needs, thereby avoiding the need for endothermic fuels and enabling the engine to operate on JP-grade HC fuel. In addition, there may be localized "hot spots" which are difficult to cool by conventional means, and which could benefit by the application of heat pipe cooling techniques.
7. An assessment of HPRC for HiMaTE will require data on engine dimensions, flow rates, pressures, temperatures, and heat loads for a representative engine and cruise condition.

2.2 Application of HiMaTE Heat Transfer Data To HPRC

At the HPRC applications phase meetings with GEAE and P & W related to their HiSPA programs, information was not provided on heat fluxes incident on the engine ramburner or nozzle walls from the combustion gas, or on convection heat transfer coefficients. Such information was needed to establish that HPRC-calculated heat loads would be consistent with methods used by contractors to establish engine design and performance characteristics.

At the high combustion temperatures involved, thermal radiation could be a significant contributor to the hot section heat load, along with convective heat transfer.²⁷ When asked whether radiation had been included in assessing heat loads, GEAE personnel indicated that radiation had not been included. Since the GEAE analysis of HiSPA engine performance was used as the basis for the HPRC heat transfer analysis, radiation was similarly neglected in evaluating HPRC heat fluxes. At the March 7, 1991 HiMaTE meeting, GEAE presented data showing heat fluxes at the inner surface of the engine hot section for HPRC and HiMaTE (see Fig. 4). The HiMaTE data is given for film cooling and no film cooling cases, and includes the radiative contribution to heat transfer. Only the no-film-cooling data is applicable to HPRC.

The HPRC data in Fig. 4 applies to the ramburner. HPRC system temperatures were then calculated using this information. The HPRC data applies for Mach 5 cruise at 70% of full power, a dynamic pressure of 1000 lb/ft², an altitude of 80,000 ft, and a recovery temperature in the engine of 4200°F.

From other data presented by GEAE, it was assumed that the HiMaTE data applied for a Mach 5.5 cruising speed and a recovery temperature of about 4500°F. Since no information was supplied on the cruise altitude and dynamic pressure, the dynamic pressure was also assumed to be 1000 lb/ft². The corresponding cruise altitude was not calculated, but is estimated to be around 83,000 ft.

²⁷Lefebvre, A. H., *Gas Turbine Combustion*, Hemisphere, New York, 1983, Chapter 8.

Ramburner and Nozzle Liner Gas-Side Surface Heat Fluxes

HPRC vs. HiMaTE

Fig. 4 HPRC/HiMaTE heat flux data from GEAE

2.2.1 Heat Transfer Coefficients. Since the HPRC heat transfer codes use heat transfer coefficients as input, the heat flux data was converted to heat transfer coefficient data for the engine by the equation

$$q = h(T_r - T_w),$$

where q is the heat flux, h is the heat transfer coefficient, T_r is the gas recovery temperature, and T_w is the inner surface temperature of the hot section. If radiative heat transfer was a significant component of the total heat flux, the above equation was used to calculate a "pseudo" heat transfer coefficient that accounts for both convective and radiative heat transfer.

Additionally, the mean heat transfer coefficient for the nozzle h_{mean} is used in the HPRC codes. The overall h_{mean} for the nozzle was obtained by weighting the mean heat transfer coefficient between the ramburner exit and the nozzle throat by the distance between those points, and the mean heat transfer coefficient between the throat and the nozzle exit by the distance between those points.

In the HPRC references design, the distance between the ramburner exit and the nozzle throat is 15 in., the distance between the nozzle throat and the nozzle exit is 65 in., and the total nozzle length is 80 in.

Thus, h_{nmean} was calculated from the equation

$$h_{nmean} = \frac{\frac{(h_c + h_{nt})}{2} 15 + \frac{(h_{nt} + h_{nx})}{2} 65}{80},$$

where h_c is the heat transfer coefficient at the ramburner exit, h_{nt} is the heat transfer coefficient at the nozzle throat, and h_{nx} is the heat transfer coefficient at the nozzle exit.

Since the heat flux at the exit of the inner nozzle surface was not included in the GEAE HiMaTE data, it was necessary to obtain the corresponding heat transfer coefficient by other means. This was done by estimating the ratio of the exit to the throat heat transfer coefficients from GEAE HiSPA data,²⁸ and then using this ratio to calculate the nozzle exit heat transfer coefficient from the HiMaTE nozzle throat data.

²⁸Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

Without Radiation. Results for the heat transfer coefficients (neglecting the contribution of radiation) are given in Table 1.

Table 1. Calculation of Heat Transfer Coefficients from GE HiMaTE Data of Fig. 4			
Parameter	Ramburner Exit	Nozzle Throat	Nozzle Exit
Heat Flux, Btu/ft ² -sec	39	144	
Recovery Temperature, °F	4500	4500	4500
Inner Surface Temperature, °F	3000	3000	
Heat Transfer Coefficient, Btu/hr-ft ² -°F	93.6	346	122
Mean Nozzle Heat Transfer Coefficient, Btu/hr-ft ² -°F		190	

It was then necessary to correct the Mach 5.5, 4500°F HiMaTE data in Table 1 to the Mach 5.0, 4200°F conditions corresponding to the HPRC reference design. The correction factor was estimated by comparing the heat transfer coefficient at Mach 5 with that at Mach 5.5, for dynamic pressure of 1000 lb/ft² and a recovery temperature of 4200°F. Again, this information was generated from General Electric HiSPA data.²⁹ The correction factors are listed below.

	Ramburner	Nozzle Throat	Nozzle Exit
$h_{M5}/h_{M5.5}$	0.905	0.823	0.930

With Radiation. In the HiMaTE data of Fig. 4, the radiative contribution to heat transfer at the nozzle throat for an inner surface temperature of 3000°F is about 7% of the convective contribution. At surface temperatures of 3300 - 3400°F which are characteristic of the HPRC reference design, the radiative contribution will be even smaller. Since the gas temperature drops off considerably aft of the throat, the radiative contribution in this region will also decrease (although the convective contribution also decreases). Therefore, it was assumed that radiation has a negligible effect on the mean heat transfer coefficient in the nozzle.

²⁹Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

The situation is quite different in the ramburner. According to the HiMaTE data of Figure 8, the radiative contribution to heat transfer at the ramburner exit is about 57% greater than the convective contribution. The basis for this calculation is not know, nor are the gas and wall emissivities upon which radiative heat transfer from the gas is strongly dependent³⁰.

For the HPRC ramburner, wall temperatures are in the 3300-3400°F range, and the gas recovery temperature is 4200°F. These temperatures compare with 3000°F and 4500°F, respectively, for HiMaTE. With the smaller gas-to-wall ΔT with HPRC, the radiative contribution should be substantially smaller.

The HPRC heat transfer codes do not presently include provision for radiative heat transfer between the combustion gas and the inner surface of the ramburner and nozzle. For purposes of calculation, it was assumed that the radiative and convective contributions to heat transfer were equal in the ramburner. An equivalent convective heat transfer coefficient was then defined for the ramburner which is twice that for convective heat transfer alone.

Resulting Heat Transfer Coefficient. Resulting heat transfer coefficients applied to the HPRC representative design are listed in Table 2, along with the original heat transfer coefficients. (The original heat transfer coefficients are slightly smaller than earlier values. This is because of an adjustment which was made in the specific heat ratio γ so that the HPRC engine temperatures would be more compatible with the General Electric data [see Sec. 2.4.3].)

Table 2. Heat Transfer Coefficients for HPRC Heat Transfer Calculations		
Case	h_c , Btu/hr-ft ² -°F	h_{nmean} , Btu/hr-ft ² -°F
Original (Ge HISPA Case 176)	98.1	144
GE HiMaTE (no radiation)	84.7	162
Ge HiMaTE (including radiation)	169*	162

*Assumes radiative and convective components of heat transfer are equal.

2.2.2 Revised HPRC Temperatures Calculations. Temperatures calculated with the HPRCc code, using the heat transfer coefficients of Table 2, are shown in Fig. 5. Without the radiative contribution, the ramburner inner wall temperature T_c decreases by 58°F from 3334°F to 3276°F when heat transfer coefficients derived from the HiMaTE data are used, while the mean nozzle inner wall temperature T_{nmean} rises by 52°F from 3343°F to 3395°F. The heat

³⁰Lefebvre, A. H., *Gas Turbine Combustion*, Hemisphere, New York, 1983, Chapter 8.

pipe vapor temperature T_{vi} in the insulated panel rises by 10°F from 2389°F to 2399°F , while T_{vr} in the radiating panel rises by 8°F from 2280°F to 2288°F . When radiation is included, T_c increase by 237°F to 3571°F , $T_{n\text{mean}}$ increases by 66°F to 3409°F , T_{vi} increases by 79°F to 2468°F , and T_{vr} increase by 67°F to 2347°F .

Fig. 5 HPRC temperatures

Since the originally assumed radiation contribution will be reduced at the higher calculated value of T_c , the actual value of T_c is smaller than indicated. Therefore, the HPRC system temperature changes resulting from the inclusion of radiative heat transfer may be regarded as upper limits.

If necessary, because of materials considerations, HPRC system and hot section temperatures can be reduced by implementing one or more of the following design options.

1. Add a thermal insulating layer to the inner surface of the hot section walls.
2. Increase the HPRC radiating area.
3. Reduce the thickness of the hot section walls.

4. Increase thermal conductivity of the hot section walls.
5. Cruise at Mach 5 or less at an altitude above 80,000 ft (lower dynamic pressure).
6. Cruise at a dynamic pressure of 1000 lb/ft² and a Mach number less than 5 (lower altitude).

The first option is considered in Sec. 2.3, and is shown to be extremely effective in reducing hot section temperatures. The last two options are assessed in Sec. 2.4, and also lead to lower combustion gas temperatures. The radiative contribution to heat transfer will be smaller at the lower gas temperatures. For example, if the cruise condition is Mach 4.5 at 90,000 ft, the gas temperature in the ramburner is 3942°F, 258°F below that of the reference HPRC design. The radiative contribution to heat transfer, and its impact on peak hot section temperatures, is then substantially lower than indicated by the above results.

2.3 Impact of Hot Section Thermal Insulation on HPRC Temperatures

The original HPRC system conceptual design required hot section temperatures considerably in excess of 3000°F in order to transfer the hot combustion gas heat load radiatively from the hot section to the surrounding heat pipe nacelle, and then to the environment. One method for reducing hot section temperatures is to thermally insulate the hot section walls, thereby increasing the thermal resistance through the heat transfer path.

An HPRC heat transfer calculation was performed for the configuration 3V reference design with a thermal barrier coating (TBC) on the inner surface of the ramburner and nozzle walls as shown in Fig. 6. TBC consists of zirconia partially stabilized with 8 wt% yttria. It has been used with considerable effectiveness to reduce the heat load on internally cooled turbine vanes of jet engines.³¹ The thermal conductivity of this material is about 0.5 Btu/hr-ft-°F at temperatures of interest.³² The effect of TBC is to increase the thermal resistance and reduce the thermal conductance at the combustion gas-hot section wall interface, according to the relation:

³¹Grisaffe, S. J. and Lowell, C. E., Overview of Lewis Materials Research - Contributions, Current Efforts, and Future Directions, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, p. 23-8.

³²Strangmen, T. E., et al., Thermal Barrier Coating Life - Prediction Model Development, Turbine Hot Section Technology 1987, NASA Conf. Publ. 2493, 1987, pp. 369-375.

Fig. 6 Thermal barrier coating (TBC) on hot section liner

$$\frac{1}{U} = \frac{1}{h} + \frac{\ell}{k},$$

where U is the thermal conductance, h is the heat transfer coefficient in the combustion gas, ℓ is the thickness of TBC, and k is the thermal conductivity of TBC. The thickness ℓ can be solved from the above equation to obtain

$$\ell = \frac{k}{h} \left(\frac{h}{U} - 1 \right).$$

It was assumed that a sufficient thickness of TBC would be added to the inner surface of the hot section to reduce the thermal conductance U in the ramburner to 30 Btu/ft²-hr-°F. The required thicknesses of TBC are then

0.130 in. on the combustor inner wall, and
0.108 in. on the nozzle inner wall.

For the configuration 3V reference design with the exit gap between the heat pipe and the nozzle closed, and a 12 in. transition section length, the results shown in Table 3 were obtained using code HPRCc.

Table 3. Effect of TBC on HPRC Temperatures and Weight	With TBC	Without TBC
Combuster inner wall temperature, °F	2670	3345
Nozzle inner wall temperature, °F	2655	3362
Heat pipe vapor temperature-insulated panel, °F	1964	2399
Heat pipe vapor temperature-radiating panel, °F	1910	2289
Heat load on insulated panel, Btu/sec	270	521
Heat load on transition section, Btu/sec	351	670
Heat load on radiating panel, Btu/sec	627	1214
HPRC weight, lb	630	644
Weight of TBC, lb	370	
Weight of HPRC + TBC, lb	1000	

The addition of TBC reduced the heat load on the radiating panel by almost one-half from 1214 Btu/s to 627 Btu/s.³³ This results in a substantial reduction in temperatures all along the heat transfer path. The peak hot section temperature is now 2670°F, within near-term temperature goals for carbon-carbon. Heat pipe temperatures, now in the 1900°F range, run about 400°F cooler. Based on a density 0.226 lb/in³ for zirconia,³⁴ the principal constituent of TBC, 370 lb of fully dense TBC is required on the hot section inner surface. For 80% dense material, 296 lb of TBC would be needed.

When TBC is applied to the surfaces of gas turbine components, thicknesses are usually in the range of 0.010 in. to 0.030 in. Here, we are considering TBC layers with a

³³Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion," Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

³⁴Deane, C. W., System Study for Mach 5 Turboramjet Engines, Report No. R84-956687-10, United Technologies Research Center, East Hartford, CT, December 1984, p. 19.

thickness as large as 0.130 in. Prior to applying the TBC, a bond coat about 0.005-in.-thick is applied to the substrate. The bond coat is usually composed of nickel or cobalt with varying percentages of chromium, aluminum, and yttrium.³⁵ At operating temperatures in the 2600°F of to 2700°F range, new bond coating techniques will likely be required. It appears that the addition of an insulation layer on the inner surface of the hot section permits the use of current or near term carbon-carbon as the hot section material. The trade-offs for using TBS are the added weight of the TBC, plus the need for development to effectively bond TBC at required temperatures and thicknesses to carbon-carbon.

2.4 HPRC at Other Cruise Conditions

The HPRC heat transfer analysis was based on representative HiSPA engine data for a dynamic pressure of 1000 lb/ft², a flight speed of Mach 5, and an altitude of 80,000 ft. The effect of other cruise conditions on HPRC system temperatures, and methods by which this information was obtained, are described in this section.

2.4.1 Specification of Other Cruise Conditions. The dynamic pressure P_d at a given altitude is given by

$$P_d = \rho V^2 = \rho C^2 M^2$$

where ρ is the air density, V is the flight speed, C is the sonic velocity, and M is the Mach number. Above 40,000 ft, the ambient temperature T_0 remains approximately constant at 392R. The sonic velocity squared, which is proportional to T_0 is then also constant. For a given dynamic pressure, the air density ρ then varies inversely as M^2 . From this relationship, the air density at any Mach number can be found from the known ρ at some specified M . The altitude corresponding to ρ can then be obtained from the Standard Atmosphere Table.³⁶ This procedure was used to find the altitude corresponding to Mach numbers 4.5 and 4 at $P_d = 1000$ lb/ft² from the air density at 80,000 ft (the altitude which corresponds to Mach 5 and $P_d = 1000$ lb/ft²). Results are given in Table 4.

³⁵Strangmen, T. E., et al., Thermal Barrier Coating Life - Prediction Model Development, Turbine Hot Section Technology 1987, NASA Conf, Publ. 2493, 1987, pp. 369-375.

³⁶Keenan, J. H. et al., Gas Tables, 2nd Ed., Wiley, New York, 1980, p. 180.

Table 4. Altitude Versus Mach Number ($P_d = 1000 \text{ lb/ft}^2$)		
Mach Number	Air Density, lb/ft^3	Altitude, ft
5.0*	0.00277*	80,000*
4.5	0.00342	75,600
4.0	0.00433	71,700

*Reference design conditions.

Engine characteristics needed to calculate heat transfer coefficients in the combustion gas stream and in the external air stream were then determined for Mach 4 and 4.5 by extrapolation of characteristics calculated by General Electric for these Mach numbers at altitudes of 70,000 ft. and 80,000 ft.³⁷

In addition to determining HPRC characteristics for Mach 4, 4.5 and 5 at altitudes corresponding to $P_d = 1000 \text{ lb/ft}^2$, HPRC characteristics were also obtained for these same Mach numbers at altitudes of 80,000 ft. and 90,000 ft. The dynamic pressures for these conditions were found by first obtaining the air density corresponding to the various altitudes from the Standard Atmosphere Table,³⁸ and noting that the dynamic pressure is proportional to the air density. The various cases that were considered are summarized in Table 5.

³⁷Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

³⁸Keenan, J. H. et al., Gas Tables, 2nd Ed., Wiley, New York, 1980, p. 180.

Table 5. Cruise Conditions Considered for HPRC Calculations

Mach Number	Dynamic Pressure, lb/ft²	Air Density, lb/ft³	Altitude, ft
5.0	621	0.00172	90,000
5.0*	1000*	0.00277*	80,000*
4.5	503	0.00172	90,000
4.5	810	0.00277	80,000
4.5	1000	0.00342	75,600
4.0	397	0.00172	90,000
4.0	640	0.00277	80,000
4.0	1000	0.00433	71,700

***Reference design conditions.**

2.4.2 Hot Section Dimensions. As the cruise Mach number changes, certain hot section dimensions also change. Dimensions for Mach 5 and Mach 4.5 were obtained from General Electric data for a hydrogen-fueled engine with an air flow rate of about 400 lb/sec. Dimensions for Mach 4 were obtained for a JP-7-fueled engine with the same flow rate.³⁹ (There was no hydrogen-fueled data for Mach 4.) A schematic of the hot section is shown in Fig. 7. The dimensions which do not vary with Mach number are also indicated on Fig. 7.

Fig. 7 Hot section dimensions

Values of the three dimensions which do vary - the nozzle throat diameter D_8 , the nozzle length downstream of the throat, L_3 , and the total hot section length, L - are given in Table 6. D_8' (obtained from GE engine performance calculations) differs slightly from D_8 (obtained from the GE dimensional data). D_8' was used in the HPRC calculations.

³⁹Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987.

Table 6. Hot Section Dimensions			
Mach Number	5.0	4.5	4.0
Dg, in.	23.6	25.9	29.2
Dg', in.	23.2	25.2	28.6
L3, in.	65	57	43
L, in.	112	104	90

2.4.3 Internal Heat Transfer Coefficients. The heat transfer coefficient at a given point along the hot section gas flow path was calculated using the following equations: ⁴⁰

$$Re = \frac{4W}{\pi \mu D},$$

$$r = Pr^{0.333},$$

$$\frac{T_r}{T_e} = 1 + \frac{\gamma - 1}{2} r M^2,$$

$$Nu_{\infty} = 0.020 \quad Re^{0.8} Pr^{0.33} \left(\frac{T_r}{T_w} \right)^{0.15}, \quad \text{and}$$

$$h_{\infty} = \frac{k Nu_{\infty}}{D},$$

⁴⁰Kunkler, H. and Kuczera, H., "Turbo-Ramjet Propulsion System Concepts for Future European Space Transport (SANGER)," IAF-87-265, 38th Congress of the International Astronautical Federation, Brighton, United Kingdom, October 1987.

where W is the gas mass flow rate, μ is the gas absolute viscosity, D is the diameter, r is the recovery factor, γ is the ratio of constant pressure and constant volume specific heats, M is the Mach number, T_r is the absolute gas recovery temperature, T_e is the absolute bulk gas temperature, T_w is the absolute wall temperature, Re is the Reynolds number, Pr is the Prandtl number, Nu_∞ is the Nusselt number for a long duct with fully developed flow, k is the gas thermal conductivity, and h_∞ is the heat transfer coefficient for a long duct with fully developed flow. Air tables were used to find the temperature-dependent gas properties.

In the hot section, the combustor is a relatively short duct, and the nozzle is a duct whose diameter varies along its length. The heat transfer coefficient h_c in the ramburner (and at the nozzle entrance) is taken to be $2h_\infty$ to allow for the thinner boundary layer on the inner surface and, hence, lower thermal resistance.⁴¹ Similarly, the heat transfer coefficient h_n in the nozzle is taken to be $1.2h_\infty$, the smaller multiplying factor being used because of the longer duct length in the nozzle. Heat transfer coefficients were calculated for the ramburner (h_c), the nozzle throat (h_t), and the nozzle exit (h_e). The heat transfer coefficient at the ramburner exit (nozzle inlet) was assumed to be equal to the mean ramburner heat transfer coefficient h_c . The mean heat transfer coefficient for the nozzle h_{nmean} was obtained by weighting the various nozzle h 's according to the equation

$$h_{nmean} = \frac{\frac{h_c + h_t}{2} L_2 + \frac{h_t + h_e}{2} L_3}{L_2 + L_3}.$$

In order to calculate heat transfer coefficients at various points along the hot section flow path, the mass velocity and gas temperatures must be known at the points of interest. Equations for calculating these parameters from known conditions at the nozzle inlet are presented in Appendix B, Nozzle Relationships. The Mach number M_i at the ramburner exit (nozzle inlet) was calculated from the equations

$$\rho_i = \frac{P_i}{R_g T_i},$$

⁴¹Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion," Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

$$A_c = \frac{\pi}{4} D_c^2,$$

$$V_i = \frac{W}{\rho_i A_c},$$

$$C_i = \sqrt{\gamma R_g T_i}, \text{ and}$$

$$M_i = \frac{V_i}{C_i},$$

where ρ_i , T_i , and ρ_i are the pressure, absolute bulk temperature, and density at the nozzle inlet, respectively. R_g is the gas constant for air, D_c and A_c are the ram burner diameter and flow area, W is the gas mass flow rate, V_i and C_i are the flow velocity and sonic velocity at the nozzle inlet, γ is the ratio of specific heats at constant pressure and constant volume, and M_i is the Mach number at the nozzle inlet. The pressure P_i at the nozzle inlet (ram burner exit) was taken to be equal to the total pressure at the engine inlet.

The specific heat ratio γ was assumed to be 1.333 during the earlier HPRC calculations. However, the value of γ was adjusted during the present calculations to conform more closely to the GE HiSPA data. The adjustment was made using the following equation.

$$T_0 = \left[1 + \frac{\gamma - 1}{2} M^2 \right] T_e,$$

where the T_0 is the absolute total temperature, T_e is the absolute static (bulk) gas temperature, and M is the Mach number. At the nozzle throat, $M = 1$. Solving for γ from the above equation for this case,

$$\gamma = 2 \frac{T_0}{T_e} - 1.$$

For the reference flight conditions of $M = 5$ at an altitude of 80,000 ft, T_e at the nozzle throat is 4144R from the GE HiSPA data, and T_0 as stated by GE personnel is 4660R. Then, from the above equation, $\gamma = 1.249$. This value was used for all calculations reported here. This smaller value of γ resulted in a slight downward adjustment in combustion gas heat transfer coefficients.

Once the nozzle inlet Mach number is known, along with the inlet temperature, pressure, and flow area, these variables can be found at any other location along the nozzle by the use of standard nozzle equations⁶ (see Appendix B). They were calculated at the nozzle throat and exit. The nozzle efficiency was assumed to be 100% upstream of the throat, and 98% downstream of the throat. Once this information was obtained, the heat transfer coefficients were calculated using the equations listed in Sec. 2.4.3 above. Results obtained for the heat transfer coefficients are summarized in Table 7 for the cruise conditions listed in Table 5. A more detailed breakdown of the data involved in calculation of the heat transfer coefficients are given in Tables C1 through C8 of the Appendix C, under Hot Section Property and Heat Transfer Data. Flow path heat transfer coefficients, and hence HPRC system temperatures, are seen to vary substantially with cruise Mach number and altitude.

2.4.4 External Heat Transfer Coefficients. Heat transfer coefficients on the HPRC nacelle and wing external surfaces also vary with cruise conditions. They were calculated using the following equations

$$T_r = T_a \left(1 + \frac{\gamma - 1}{2} r M^2 \right), \text{ and}$$

$$h_{ext} = \frac{\epsilon \sigma T_{re}^4}{T_r - T_{re}},$$

where T_r is the absolute recovery temperature, T_a is the absolute ambient air temperature = 392°R, γ is the specific heat ratio = 1.4, Pr is the Prandtl number, r is the recovery factor = $(Pr)^{1/3}$. M is the cruise Mach number, ϵ is the surface emissivity = 0.8, σ is the Stefan-Boltzmann constant = 0.476×10^{-12} Btu/ft²-sec-R⁴, T_{re} is the absolute radiation equilibrium temperature, and h_{ext} is the heat transfer coefficient. The Prandtl number is a slowly varying function of temperature. It was evaluated at a temperature of 1880°F, where its value is 0.722. It follows that $r = 0.900$.

Table 7. Internal Heat Transfer Coefficients for Various Cruise Conditions

Mach Number	Altitude,ft	Dynamic Pressure,lb/ft ²	Heat Transfer Coefficient Btu/hr-ft ² -°F			
			Nozzle Inlet (h _c)	Nozzle Throat (h _t)	Nozzle Exit (h _e)	Nozzle Mean (h _{mean})
5.0	80,000	1000	98.1	204	81.6	144
4.5	80,000	810	73.4	136	65.2	101
4.0	80,000	640	56.9	83.2	51.8	68.2
5.0	90,000	621	67.4	145	57.1	102
4.5	90,000	503	51.5	93.3	46.6	70.4
4.0	90,000	397	38.9	58.8	35.3	47.5
4.5	75,600	1000	90.1	163	79.2	122
4.0	71,700	1000	79.1	118	71.3	95.5

Heat transfer coefficients were evaluated at angles of attack of 0 and 10 deg. The value for 0 deg. was used as h_{ext} for the sides of the HPRC nacelle. The value for 10 deg. was used as h_{ext} for the nacelle bottom and the undersurface of the aircraft wing to which the engine is attached. Results are given in Table 8 for the same cruise conditions (Table 5) as were used to calculate the internal heat transfer coefficients of Table 7.

2.4.5 Maximum Ramburner Temperature at Various Cruise Conditions In Fig. 8, the maximum (inner wall) ramburner temperature with HPRC is shown for each of the cruise conditions that have been considered. These temperatures were generated with the HPRCc code for Configuration 3V, with the engine operating at approximately 70% of full power. In general, the temperature of the inner ramburner wall is higher than the mean temperature of the nozzle inner wall. The reverse is true for the Mach 5 cases. At an altitude of 80,000 ft, the nozzle temperature is 9°F higher. At an altitude of 90,000 ft, the nozzle temperature is 28°F higher. In each case, the length of the HPRC nacelle is equal to the hot section length (112 in., at Mach 5; 104 in. at Mach 4.5; 90 in. at Mach 4). Had the nacelle length been maintained constant at the reference design value of 112 in., temperatures at Mach numbers less than Mach 5 would be significantly lower than those of Fig. 8.

Fig. 8 Maximum ramburner temperatures

Table 8. External Heat Transfer Coefficients for Various Cruise Conditions

Mach Number	Altitude ft	Dynamic Pressure, lb/ft ²	Angle of Attack, deg	Radiation Equil. Temp., °R	Recovery Temp., °R	Heat Transfer Coeff., Btu/hr/ft ² -°F
5.0	80,000	1000	0	1460	2156	8.96
			10	1642	2156	19.40
4.5	80,000	810	0	1330	1821	8.74
			10	1468	1821	18.05
4.0	80,000	640	0	1200	1521	8.86
			10	1294	1521	16.95
5.0	90,000	621	0	1402	2156	7.03
			10	1595	2156	15.83
4.5	90,000	503	0	1285	1821	6.98
			10	1435	1821	15.07
4.0	90,000	397	0	1168	1521	7.23
			10	1274	1521	14.63
4.5	75,600	1000	0	1358	1821	10.08
			10	1493	1821	20.78
4.0	71,700	1000	0	2141	1521	11.62
			10	1328	1521	22.11

It is evident that the cruise Mach number and altitude have a strong effect on the maximum ramburner temperature. At Mach 5, the temperature drops from 3320°F to 3150°F when the altitude increases from 80,000 ft. to 90,000 ft. At Mach 4.5, the temperature at a dynamic pressure of 1000 lb/ft² and an altitude of 75,600 ft. drops to slightly over 3100°F. As the altitude increase first to 80,000 ft. and then to 90,000 ft., the temperature drops further to 3020°F and then to 2860°F. At Mach 4.0, $P_d = 1000$ lb/ft², and an altitude of 71,700 ft., the temperature is 2880 °F. As the altitude rises first to 80,000 ft. and then to 90,000 ft., the temperature drops to 2740°F and then to 2570°F. These results show that if the engine is cruising slightly below Mach 4.5 at 90,000 ft., the hot section temperatures will be around 2800°F, a level within near-term expectations for carbon-carbon. Temperatures will be even lower if the nacelle length is maintained at the reference design value of 112 in.

Reference to Table C1 and Table C5 in the Appendix C shows that the ramburner gas temperature is 258°F cooler at Mach 4.5 and 90,000 ft. than at the reference design cruise conditions of Mach 5 and 80,000 ft. Therefore, the radiative contribution to heat transfer from the combustion gas to the ramburner walls should be appreciably less for cruise at Mach 4.5 and 90,000 ft.

2.5 Integration of HPRC Heat Pipe Data With Heat Transfer Studies

In the HPRC Phase I heat transfer studies information on heat transport requirements of the heat pipes comprising the nacelles to which the hot sections radiated was not initially available⁴². Therefore, engineering judgement was applied in estimating heat-pipe-related characteristics used in the configuration 3V heat transfer model. LANL-supplied data on heat pipe and wick demensions required for adequate heat transport through the HPRC heat pipes subsequently became available.⁴³ Heat transfer calculations were consequently repeated for configuration 3V, in order to determine the sensitivity of HPRC temperatures and weight to the required heat pipe and wick dimensions.

2.5.1 HPRC System Configuration. The case selected for re-analysis was configuration 3V, which was used as the basis for the nominal HPRC design (see Fig. 9). In configuration 3V, the HPRC is composed of flat panels which comprise the nacelle surrounding the engine hot section to be cooled. The panels are made up of individual heat pipes cells which extend along the direction of the engine axis. Heat from an insulated interior or underwing panel is transported to an exterior side or bottom panel for radiative disposal,

^{42,43}Martin, R.A., Keddy, M., Merrigan, M.A., and Silverstein, C.S., "Heat Pipe Radiation Cooling for High Speed Aircraft Propulsion," Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

along with heat radiated directly to that panel from the hot section. The transfer is accomplished using a transition section between an insulated and an exterior panel. The transition section consists of a flat panel of heat pipe cells which are oriented along a direction normal to the engine axis. The panel is bent at right angles, so that one half of the transition section panel is in contact with an insulated heat pipe panel and the other half is in contact with the adjacent exterior radiating panel.

The HPRC system for each engine then consist of four flat heat pipe panels and two right-angle heat pipe transition sections. Since each heat pipe cell in the nacelle panels "sees" approximately the same axial heat load, each panel is approximately isothermal. The flat nacelle panels are also capable of being joined by a flexible hinge. The rear sections of the top and bottom panel can then move in accordance with the exit dimensions of the contained engine nozzle, minimizing the gap between nacelle and nozzle.

Configuration 3V may be compared with configuration 3F (see Fig. 9). In this case the nacelle is fabricated from two sets of right-angle panels whose heat pipes are oriented normal to the direction of the engine axis. Since all the heat pipe cells have a right angle bend in the middle, fabrication of the HPRC nacelle is more involved than for configuration 3V, where only the two transition section have right-angle heat pipe cells. Also, each heat pipe cell in the nozzle section sees a different axial heat load. The nacelle is thus considerable more nonisothermal than for configuration 3V. Finally, the rigid construction of configuration 3F precludes the use of movable panels to minimize the nacelle-nozzle gap.

The LANL analysis of heat pipe heat transport requirements considered several wick types. It was concluded that an artery-type wick with two arteries per heat pipe cell was optimum for the intended application, because of the low volume and weight of the artery wick. For configuration 3V, the heat pipe and wick dimensions were determined for a transition section length of 32 in.

3V. ONE-D HEAT PIPES

3F. ONE-D HEAT PIPES

Fig. 9 HPRC configurations 3V and 3F

2.5.2 Transition Section. The heat pipe cross-section and dimensions for the transition section are shown in Fig. 10. The vapor space thickness (rib height) is larger in the transition section to keep its axial heat flux from becoming too large. Too high axial heat flux in the transition section heat pipes could result in an excessive capillary pressure requirement.

Fig. 10 Heat pipe cross-section for configuration 3V

HPRC temperature and weight will vary with the transition section length. Therefore, a range of lengths was considered in order to identify the optimum value. Since heat pipe dimensional data was available only for the 32-in.- length, heat pipe dimensions at other transition section lengths were estimated.

The relative orientation of the transition section with respect to adjacent insulated and radiating panels is shown in Fig. 11. The heat transferred to one-half the transition section from an adjacent insulated panel plus the heat which is absorbed directly by that half-section determines the effective axial heat flux for the transition section.

Fig. 11 Transition section geometry

The heat which is directly absorbed by the half-section adjacent to the radiating panel is transported by evaporating vapor across the thickness of the transition section. The vapor then condenses on the wick section which is in contact with the radiating panel, releasing its heat of condensation. Thus, the transverse-flowing heat does not contribute to the axial heat flux in the transition section. The total axial heat transfer rate in the transition section Q_{ta} may then be expressed as

$$Q_{ta} = Q_i + \frac{Q_t - Q_i}{2},$$

where Q_t is the total rate at which heat is absorbed by the transition section, and Q_i is the rate at which heat is transferred to the transition section from the insulated panel.

It is desirable to maintain the axial heat flux constant in the transition section as the transition section length L_t is varied. It was assumed (and later verified) that Q_{ta} remains essentially constant with variations in the transition section length. For a constant axial heat flux, this requires that the vapor space area also does not change. Then, assuming that the vapor space width is fixed, the vapor space thickness w_{vt} must vary inversely with L_t . The heat transfer characteristics of an HPRC system with configuration 3V were then redetermined, using the above assumptions and heat pipe cross-section and wick

was necessary to modify the configuration 3V code HPRCc to accommodate the artery wicks. The resulting modified code is called HPRCc1.

The variation with transition length L_t of the vapor space thickness w_{vt} , the total heat pipe thickness w_{hpt} , Q_i , Q_t , and Q_{ta} is given in Table 9. The calculations apply for a closed gap between the HPRC nacelle and the nozzle at the nozzle exit. The following values for other heat pipe dimensions were also used in the calculations.

Heat pipe wall thickness	0.015 in.
Heat pipe rib thickness	0.020 in.
Artery distribution wick thickness	0.004 in.
Arteries per heat pipe cell	2
Artery inner diameter	0.0938 in.

It can be seen that Q_{ta} varies from 617 to 611 Btu/sec, about 1%, as L_t decreases from 32 to 12 in. The assumption that Q_{ta} remains essentially constant with variations in the transition section length is thus confirmed.

2.5.3 Revised HPRC Calculations. In Fig. 12, the ramburner inner wall temperature, the nozzle inner wall temperature, the heat pipe vapor temperature in the radiating panel, and the HPRC system weight are plotted against the transition section length. The heat pipe weight decreases from 710 lb to 640 lb as L_t drops 32 to 12 in., while the hot section and vapor temperatures remain almost constant. Thus, a small transition section length not only minimizes HPRC system weight, but also reduces the size and, hence, the fabrication effort of the only configuration 3V heat pipe component which is not a planar panel. This conclusion is similar to that reached in the original calculations.

Table 9. Effect of Transition Section Length on Heat Pipe Dimension and Heat Loads					
L_t , in.	w_{vt} , in.	w_{hpt} , in.	Q_i , Btu/sec	Q_t , Btu/sec	Q_{ta} , Btu/sec
32	0.792	0.830	412	821	617
24	1.056	1.094	462	769	616
16	1.584	1.622	511	715	613
12	2.11	2.148	534	687	611

Fig. 12 Effect of transition section length on HPRC characteristics

In Table 10, key HPRC temperatures and system weights are shown at a transition section length of 12 in. for the revised and original heat transfer calculations. L_t is the transition section axial length, T_{ci} is the mean temperature on the inner surface of the

ramburner, T_{ni} is the mean temperature on the inner surface of the nozzle, T_{vip} is the vapor temperature in an insulated heat pipe panel, T_{vrp} is the vapor temperature in a radiating heat pipe panel, and W_s is the HPRC system weight.

Table 10. Heat Transfer Characteristics of HPRC for Original and Revised Calculations

Case	L_t , in.	T_{ci} , °F	T_{ni} , °F	T_{vip} , °F	T_{vrp} , °F	W_s , lb
Revised	12	3327	3350	2338	2289	640
Original	12	3345	3362	2399	2289	644

The hot section temperatures for the revised calculation run 12°F to 18°F cooler than the original design. The vapor temperature in the insulated heat pipe panel, runs 61°F cooler, primarily because of the much thinner wick through which heat must flow in passing from one panel to another. The vapor temperature in the radiating panels is essentially the same in both cases, primarily because the radiating heat load doesn't change significantly. The weight is also about the same, being 4 lb lower for the revised case. There is little change in system weight because the much lighter artery wick in the revised design is largely offset by the thicker, longer, and consequently heavier heat pipe ribs. The surface heat flux at the transition section where heat is transferred from adjacent insulated panels does increase markedly with a decrease in the the transition section length, since more heat is then absorbed from the hot section by the insulated panels and less surface area is available at the insulated panel-transition section interface.

In Table 11, the variation of the surface heat flux with transition section length is shown, along with the boiling limit corresponding to the vapor temperature in the transition section. While the surface heat flux increase by almost a factor of 4 as the transition length decrease from 32 to 12 in., it remains far below the boiling limit. This is a consequence both of the lithium fluid parameters which favor a high boiling limit and the very thin distribution wick along the heat pipe walls. Thus, heat pipe operation in the transition section will not be

disrupted by the occurrence of boiling in the wick. It may be concluded that the artery wick and revised heat pipe cross-sectional dimensions do not significantly affect the HPRC system weight or temperatures. However, the revised design is necessary to assure adequate heat transport capability in the various heat pipe panels.

Table 11. Variation of Transition Section Surface Heat Flux with Transition Section Length

Transition Section Length, in.	Transition Section Vapor Temperature, °F	Surface heat Flux, Btu/ft ² -sec	Boiling Limit, Btu/ft ² -sec
32	2306	26.5	11,270
24	2307	39.6	11,250
16	2312	65.7	11,120
12	2317	91.7	10,990

2.6 Applicability of JP-4 Fuel to HPRC

With HPRC, the hot section heat load on the engine is radiated to the surrounding heat pipe nacelle and passively dissipated to the surrounding environment. The auxiliary heat load on the aircraft is absorbed by the JP-7 fuel. JP-7 is thermally stabilized, with an upper temperature limit of about 550°F.

Fig. 13 JP-4 fuel temperature versus Mach number

Conventional JP-4 fuel is not thermally stabilized, and has an upper temperature limit of around 300°F-350°F. It is of interest to determine maximum cruising speeds if HPRC were employed on an engine fueled with JP-4. Assuming that the specific heat of JP-4 and JP-7 are comparable, the bulk JP-4 fuel temperature after absorbing the aircraft heat load at 70% of full thrust is derivable from data in Ref. 7. The variation of fuel temperature with Mach number is shown in Fig. 13. From Fig. 13, the upper limit on cruise Mach number with HPRC when JP-4 is used as fuel is about Mach 3. The fuel would then be heated to about 350°F after absorbing the aircraft heat load. At flight speeds above Mach 4, nonconventional, high thermal capacity fuels are normally required for direct fuel cooling of the engines. It appears that HPRC would be a suitable alternate cooling approach for such situations, if the engine were fueled with JP-4.

3.0 TEST ARTICLE DESIGN

3.1 Heat Pipe Concept

A heat pipe is a device that transfers large amounts of heat through a small area with a small temperature difference and without external power or control. The principle of operation

of a heat pipe is described in detail elsewhere.^{44,45,46} In its most basic form, a heat pipe consists of an evacuated and sealed tube of impervious material lined inside with a thin layer of porous material. Before sealing the tube, a working fluid chosen for the specific application is admitted in sufficient quantity to saturate the porous liner or wick.

When a source of heat is placed in contact with a portion of the heat pipe external surface (the evaporator), the working fluid vaporizes while absorbing its latent heat of vaporization per unit mass of fluid. Internal to the heat pipe, the liquid and vapor phases of the working fluid are in equilibrium at the surface of the wick structure. The vapor moves through the central void space and condenses producing liquid and liberating heat in (the condenser) regions where the wall-wick-liquid matrix is at lower temperature than the evaporator. The conversion of fluid to vapor at the surface of the wick structure in the evaporator region causes the liquid-vapor interface to develop surface curvature concave toward the liquid space. Surface tension effects at this curved liquid-vapor interface result in a fluid pressure decrease that is proportional to the surface tension of the fluid and inversely proportional to the radius of curvature of the liquid-vapor interface. This decreased fluid pressure causes flow of liquid through the wick structure from the cooler regions of the heat pipe back to the evaporator region. Continuation of this process provides a two-phase flow loop transferring heat from the evaporator to the condenser regions of the device with capillary pumping of the liquid provided by the liquid surface tension.

For a heat pipe to operate, the maximum capillary pumping pressure must be able to overcome the total pressure drop in the heat pipe, which consists of three parts. One pressure drop is required to return the liquid from the condenser to the evaporator and another to move the vapor from the evaporator to the condenser. Finally, a pressure drop may result from the difference in elevation between the evaporator and the condenser. The capillary pressure pumps condensed liquid from the condenser back to the evaporator for re-evaporation.

Thus, a heat pipe can continuously transport the latent heat of vaporization from evaporator regions to condenser regions without drying out the wick. The process continues as long as the flow passage for the working fluid is not blocked and sufficient capillary pressure is maintained.

Heat pipes that operate at high temperatures utilize working fluids with high latent heats of vaporization and can, therefore, achieve very high heat transfer rates. In addition, the

⁴⁴Dunn, P. D. and Reay, D. A., Heat Pipes, 3rd ed., Pergammon Press Ltd., 1982.

⁴⁵Chi, S. W., Heat Pipe Theory and Practice: A Source Book, Hemisphere Publishing Corp., 1976.

⁴⁶Brennan, P. J. and Krolczek, E. J., "Heat Pipe Design Handbook," Vol. II, prepared by B & K Engineering, Inc., for NASA Goddard Space Flight Center under contract no. NAS5-23406, 1979.

thermal resistances associated with the fluid flow and vaporization within the heat pipe are commonly low enough that the process can be considered to be isothermal. To illustrate these points, at liquid-metal temperatures of about 2240°F (1500 K), heat pipes have been operated at Los Alamos with axial power densities up to 20,246 Btu/ft²-s (23 kW/cm²) over distances of 9.8 ft to 13 ft (3 m to 4 m), with total internal temperature differences of only a few degrees.⁴⁷

Heat pipes can operate over a range of temperatures; however, at each operating temperature, there is a maximum axial heat transport capacity. The thermal-transport performance of a heat pipe is subject to a number of limitations depending on the physical configuration of the heat pipe, the properties of the working fluid, the thermal fluxes, and the heat pipe orientation with respect to body forces. The limitations of concern for most applications are the sonic, entrainment, capillary (or wicking), and boiling limits.⁴⁸

3.2 General Design Principles

The approach taken to design an HPRC heat pipe optimized for performance and weight was as follows: (1) identify alternative wick structures and evaluate their relative performances, (2) select and optimize one or more suitable wick structure(s), (3) design heat pipes for maximum thermal performance and minimum weight, and (4) detail and evaluate the performance of the selected design(s).⁴⁹ Criteria for selecting and evaluating alternative heat pipe designs included weight and volume, ease of fabrication, and reliability. Design optimization was done on the basis of mass, power transport, and vapor temperature drops.

Thermal performance analyses were done by means of hand calculations and computer-aided calculations. The relations used in the heat pipe design effort were reviewed in elsewhere.^{48,50,51} The computer-aided design effort utilized HTPIPE, the Los Alamos steady-state, heat pipe analysis computer program described in another report.⁵² Thermal

⁴⁷Merrigan, M. A., Martinez, E. H., Keddy, E. S., Runyan, J. R., and Kemme, J. E., "Performance Demonstration of High Power Space Heat Pipe Design," 18th Intersociety Energy Conversion Engineering Conference (IECEC), Orlando, FL, August 21-26, 1983.

⁴⁸Dunn, P. D. and Reay, D. A., Heat Pipes, 3rd ed., Pergamon Press Ltd., 1982.

⁴⁹Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion," Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

⁵⁰Chi, S. W., Heat Pipe Theory and Practice: A Source Book, Hemisphere Publishing Corp., 1976.

⁵¹Brennan, P. J. and Kroliczek, E. J., "Heat Pipe Design Handbook," Vol. II, prepared by B & K Engineering, Inc., for NASA Goddard Space Flight Center under contract no. NAS5-23406, 1979.

⁵²Woloshun, K. A., Merrigan, M. A., and Best E. D., "HTPIPE: A Steady-State Heat Pipe Analysis Program, A User's Manual," Los Alamos National Laboratory Manual LA-11324-M, 1988.

performance for the selected design was predicted under conditions of one-g adverse gravitational force over the temperature range of 2100°F to 2400°F. Results of these calculations were presented⁵³ in and formed the basis for the specification of the heat pipe geometry and annular wick design used in the HPRC feasibility program.

3.3 Design Objective

In the feasibility phase the objective was to design a test article that could be used to demonstrate the principle of HPRC in the ground test program. The test article configuration represents one element from one of two L-shaped cooling subsystem sections of the HPRC Phase I study configuration 3F shown in Fig. 9 for a representative hypersonic engine operating at cruise conditions at Mach 5 and 80,000 ft altitude. This configuration was selected for the initial experiments on the basis of performance, operational, and fabrication criteria.

In the selection of the test article configuration, a number of trade-offs were considered. These included consideration of fixed versus variable geometry, and elimination of thermal contact resistances inherent in concepts using segmented heat pipe structures as in HPRC Phase I study configuration 3V shown in Fig. 9. Thermal coupling of axial heat pipe nacelle segments to transverse transition sections entails potential thermal contact resistances at the segment joints. The test configuration can be physically modeled with parallel heat transfer paths without the problem of contact resistance, albeit with a right-angle bend in the middle. This configuration has the advantages of being lighter in weight, more reliable, and easier to fabricate because of its simplicity. However, fabrication of the test article presumes the capability of constructing an L-shaped heat pipe or bending a circular heat pipe.

It was shown that along the propulsion system nozzle, an axial heat flux gradient and, hence, an axial temperature gradient can be expected.⁵³ Such a temperature gradient is expected to lead to the transverse heat pipes operating at different temperatures. These temperature differences require that consideration be given to minimization of adverse thermal stresses within the heat pipe nacelle. This can be accomplished using supplemental convective cooling with compressor bleed or ram air. Accordingly, it is expected that the first applications of HPRC systems to hypersonic engine cooling will involve the use of supplemental convective cooling to minimize the axial temperature gradients, as well as to bring down the ramburner and nozzle wall temperatures. (An analysis of the use of supplemental convective cooling to cool nozzle hot spots to the mean temperature was previously presented.⁵³ The analysis showed that only 0.3% of the engine airflow rate would be required at the Mach 5

⁵³Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion, " Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

engine operating point.) Significant axial temperature gradients are not expected in the application of HPRC to the ramburner.

For ease of fabrication, the test article configuration uses transverse, circular heat pipes of 0.625-in.-diameter in the heat pipe nacelle instead of heat pipes with a rectangular cross section. (In an actual application, a flat heat pipe nacelle surface would be exposed to the airflow. A number of design options could be developed to accomplish this.) Test article design criteria were established to allow for worst-case heat flux and temperature conditions expected in the representative engine heat pipe nacelle. Based on the Phase I studies, the test article is being designed to prototype operating conditions in the heat pipe temperature range 1149°C (2100°F) to 1316°C (2400°F), a heat load of about 7 kW to 10 kW per heat pipe per system, a radial heat flux of from 13 W/cm² to 15 W/cm² and an axial heat flux of about 5 kW/cm² to 6 kW/cm², depending on whether a rectangular or equivalent circular heat pipe cross section is assumed.

3.4 Alternative Materials Considerations

Materials availability was factored into the test article design because of the high cost of custom-made refractory metal components. The Thermo Electron division of Tecomet, Wilmington, MA and the Schwarzkopf Development Corp., Holliston, MA division of Plansee, Reutte, Austria were contacted to determine materials availability and costs.

In addition, the alternative materials and materials issues shown in Table 12 were considered. Under the heading "Heat Pipe Nacelle," two families of materials are represented, molybdenum and niobium (Columbium). The high-temperature strength and creep properties of the molybdenum family are better than those for niobium. However, niobium alloys have been used in aircraft engines since the mid-1960's because they have superior room temperature workability (ductility) and weldability. Niobium materials are routinely welded using tungsten/inert gas techniques; whereas, molybdenum materials must be joined by electron-beam welding. In either case the refractory metals will have to be coated with some specialized protective material to inhibit high-temperature oxidation and corrosion. The preliminary materials effort has led to the decision to fabricate HPRC test article heat pipe shells out of LCAC molybdenum.

Table 12

HPRC Materials

HEAT PIPE NACELLE

- LCAC Molybdenum
- PM Molybdenum
- TZM (0.5%Ti, 0.08%Zr, 0.01-0.04%C)
- Mo-41Re (PM)
- Niobium (Columbium)
- Nb-1Zr
- C-103 (Nb-10Hf-1Ti)
- CB-752 (Nb-27Ta-10W-0.8Zr)

RAMBURNER AND NOZZLE WALL

- Carbon-Carbon
- MoSi₂/SiC Composites

PROTECTIVE COATINGS

- Iron-Chromium Silicide (R512E) for C-103
- Silicon Carbide, Silicon Nitride, or Iridium for Carbon-Carbon
- Molybdenum Disilicide for Molybdenum

TECHNICAL ISSUES

- Workability (ductility)
- Recrystallization
- Creep Strength and Modulus of Elasticity
- Compatibility with Liquid Metals
- Oxidation

Powder-pressed molybdenum tubing products produced by powder metallurgy (PM) techniques were avoided in favor of LCAC molybdenum because of known problems with e-beam welding the PM products, namely, cracking in the weld region and formation of spherical voids in the weld nugget by the liberation of contaminant gases trapped in the material.

3.4.1 Heat Pipe Nacelle Alternative Materials. Niobium alloys are generally easier to work with (machine, weld, bend, etc.) than the low-carbon, arc-cast (LCAC) molybdenum or the molybdenum alloy TZM (0.4% to 0.55% by weight titanium, 0.06% to 0.12% zirconium, and molybdenum) materials that were discussed in another report.⁵⁴ Certain niobium alloys have been used since the mid-1960's in relatively high-temperature aerospace applications.

In particular, the niobium alloy C-103 (Nb-10%Hf-1%Ti) has been used to fabricate panels in the afterburner section of gas turbine engines. Despite its inherent lack of resistance to oxidation, C-103 has been used in these severely oxidizing environments at temperatures in excess of 1600 K (2420°F) through the use of surface protecting coatings. When exposed to combustion gases, C-103 is often coated with iron-chromium silicide (R512E) for oxidation protection. At temperatures above approximately 1400 K (2060°F) creep properties of C-103 are relatively poor. In the afterburner applications, which involve many thermal cycles without extensive total time (50 hrs to 100 hrs), the structural parts are stiffened through design features such as convolutions, ribs, welds, etc., to compensate for relatively poor creep properties.

The niobium alloy FS-85 (Nb-27%Ta-10%W-0.8%Zr) was also considered as an alternative heat pipe structural material for the HPRC application with the assumption that similar design features will allow their use at the anticipated temperatures. The niobium alloy FS-85 has better high-temperature creep properties than C-103, but retains good room temperature ductility. Test data and evaluations of the strength characteristics of these materials has been obtained from NASA technical reports produced at the Lewis Research Center by Bob Titran and others.^{55,56}

A request was made to Teledyne Wah Chang Albany (TWCA), OR, to quote on rectangular cross-section tubing fabricated in the niobium alloys C-103 or FS-85. These tubes

⁵⁴Martin, R. A., Keddy, M., Merrigan, M. A., and Silverstein, C. S., "Heat Pipe Radiation Cooling for High-Speed Aircraft Propulsion, " Phase I Final Report, Los Alamos National Laboratory report LA-UR-91-586, February 14, 1990.

⁵⁵Titran, R. H., and Klopp, W. D., "Long-Time Creep Behavior of the Niobium Alloy C-103," NASA-TP-1727, October 1980.

⁵⁶Titran, R. H., and Hall, R. W., "High-Temperature Creep Behavior of Columbium Alloy, FS-85," NASA TN D-2855, June 1965.

would include right-angle bends and would have niobium porous metal wicks (with arteries) produced by porous powder metallurgy (PPM) techniques.

TWCA indicated that FS-85 was not available, and that four to five months alone would be required to obtain C-103 tubing with the desired rectangular cross-section, with an additional two months required for TWCA to fabricate the PPM wicks. Therefore, use of these materials in the Phase II effort was ruled-out. However, because lower HPRC system operating temperatures appear feasible as a result of Phase II calculations (see Sec. 2.3 and Sec. 2.4.5), niobium materials should be considered for possible future use in HPRC applications.

3.4.2 Ramburner or Nozzle Wall Alternative Materials. Successful application of an HPRC system requires the availability of high-temperature materials suitable for use as ramburner or nozzle inner wall materials. In Phase I the use of oxidation-resistant carbon fiber reinforced carbon matrix (carbon-carbon) composites was assumed. Carbon-carbon materials coated with vapor deposited silicon carbide or silicon nitride for oxidation protection have been successfully tested for hundreds of hours of thermal cycling at temperatures up to 2550°F, and sustained exposure at 3000°F is expected by the year 1995.⁵⁷

Other high-temperature materials are being developed. These include molybdenum disilicide/silicon carbide composites.⁵⁸ These composites are considered a significant new development in high-temperature structural materials. Molybdenum disilicide (the matrix material) is an intermetallic compound that is extremely resistant to high-temperature oxidation, but ductile and highly fracture resistant at high temperatures. Silicon carbide (the filler material) is a ceramic reinforcement that strengthens the molybdenum disilicide at high temperatures.

Molybdenum disilicide/silicon carbide (MoSi_2/SiC) composite materials can be used as structural components at temperatures up to 3270°F.⁵⁹ These new composites are stronger than current superalloys at elevated temperatures, and more fracture resistant at high temperatures than current structural ceramics. In oxidizing environments, these composites remain intact because molybdenum disilicide forms a protective silica layer barrier that limits oxidation.

3.4.3 Oxidation Protection Coatings. Developments and improvements in the areas of high-temperature structural materials and oxidation protection coatings for high-temperature

⁵⁷Sheehan, J. E., "Oxidation-Resistant Carbon-Carbon Composites," Engineered Materials Handbook, Am. Soc. for Metals International, Metals Park, OH, 1987, pp. 920-921.

^{58,59}Carter, D. H., "SiC Whisker-Reinforced MoSi_2 ," Los Alamos National Laboratory report LA-11411-T, September 1988.

structural materials are relevant to broadening the hypersonic operating envelope of the HPRC technique. Materials research and development tasks were not included in the current HPRC feasibility effort. However, materials development will be required for the implementation of molybdenum/lithium or niobium/lithium HPRC heat pipe systems capable of operating in air. (The current program included heat pipe testing at about 2300°F in a vacuum environment to prevent oxidation of the refractory casings.) A review of current oxidation protection coatings for refractory metals and for refractory composite materials research and development work is available.⁶⁰ Another report ⁶¹, presents test results for silicide coatings on molybdenum, and a third report⁶², presents results using silicon carbide.

3.5 Test Article Scaling

The test article size has been determined by the HPRC Phase I study for a realistic, representative, variable-cycle hypersonic jet engine operating at Mach 5 and 80,000 ft altitude conditions. The test article was scaled to demonstrate that the heat pipe nacelle can in fact operate at the design heat pipe temperature, provide adequate axial and radial heat transport capacity, and reject the required heat throughput by radiation heat transfer.

To determine the proper size of the model test article, an analytical scaling procedure was developed. The procedure will be outlined here. During operation of any heat pipe, the capillary pumping pressure must be sufficient to overcome the operational pressure drops in the pipe caused by gravitational and acceleration forces, as well as the pressure drops owing to vapor flow and liquid flow. The capillary pumping pressure is inversely proportional to the effective pore radius in the wick material. Thus, the wick geometric properties play a critical role in heat pipe design. For the HPRC program, the wick material had to be chemically compatible with lithium at elevated temperatures (up to 2400°F).

The very fine mesh wire cloth fabricated from refractory metal material needed for HPRC wick fabrication is not readily available from commercial sources. Los Alamos has a limited supply of suitable Mo-41Re material in 400-mesh by 400-mesh size with a theoretical or geometric pore size of 32 mm (see Sec. 4.2). Selection of this material determined the capillary pumping pressure available for the scaled heat pipe.

⁶⁰Glass, David E., et al., "Design Study of a Refractory-Composite/Heat-Pipe-Cooled Wing leading Edge," NASA Langley Research Center, NASA-CR-1157, in press.

⁶¹Tuominen, S. M. and Dahl, J. M., "Cyclic Oxidation Testing of Molybdenum Protected by Silicide Coatings," *J. Less Common Metals*, Vol. 81, 1981, pp. 249-260.

⁶²Merrigan, M. A., Dunwoody, W. E., and Lundberg, L., "Heat Pipe Development for High-Temperature Recuperator Application," *Advances in Heat Pipe Technology*, Fourth International Heat Pipe Conference, Reay, D. A., Ed., Pergamon Press, September 1981, pp. 245-257.

To perform the scaling analysis, it was assumed that the model would operate at the same temperature and heat flux conditions as the prototype, namely, in the heat pipe temperature range 1149°C (2100°F) to 1316°C (2400°F), and with a heat load of about 7 kW to 10 kW per heat pipe per system, with a radial heat flux of from 13 W/cm² to 15 W/cm², and with an axial heat flux of about 5 kW/cm² to 6 kW² (see Sec. 3.3). It was further assumed that the model and prototype have the same cross sectional dimensions. Finally, it was assumed that the model heat pipe wick performance was restricted to that obtainable with 400-mesh screen with a minimum geometric pore size of 32 mm.

Using the functional relationship of heat pipe pressure drops to length, mass-flow rate, cross-sectional area, and heat throughput and using results from the Los Alamos heat pipe design code HTPIPE, while allowing for a design margin factor of 2, it was determined that the scale heat pipe leg lengths should be 39 in. It can be shown that a model with 39-in.-leg-lengths using 400-mesh wick material corresponds a full-scale prototype using a wick material with an effective pore size of 18 mm with 70-in.-legs operating at the worst-case tilt (45°) in a 1-g gravitational field. The test article was, therefore, fabricated in the shape of an "L" with leg lengths of approximately 39 in.

3.6 HTPIPE Calculations for the HPRC Test Article

Results from the steady-state heat pipe computer code HTPIPE are shown in Table 13 for the geometric and physical input parameters characterizing the scaled HPRC test article heat pipe, including an effective pore radius of 37.5 microns (0.00375 cm), the value achieved with the HPRC test article wick during the final pore testing. For modeling purposes, in this computer run the heat pipe was assumed to be tipped to an angle of 45° and operating against gravity. It can be seen from the table at a test article operating

Table 13

**HPRC Heat Pipe Predicted Performance
For the Measured Effective Pore Radius
Using the Los Alamos Computer Code HTPIPE**

6	geometry: annular wick
1	working fluid: Li
99.10000	evaporator length (cm)
0.00000	adiabatic length (cm)
99.10000	condenser length (cm)
-45.00000	tilt angle (0 to 90, positive for assist)
0.70500	pipe inside radius (cm)
0.01000	distribution screen thickness (cm)
0.00127	half of wire diameter (cm)
0.07240	annulus thickness (cm)
0.00375	effective pore radius of liq-vap interface (cm)
0.67000	wick surface porosity (0.<poro<1.)
1.00000	thermal conductivity of pipe wick (W/cm K)
0.00127	nucl. site radius (cm), if 0.0 defaults to .00127
1.21778	vapor passage area (cm**2)
0.30424	total liquid flow area (cm**2)

tee	-----pressure drops-----		temperature		power limits
	capillary liquid hydrostatic	evaporator adiabatic condenser	tbe tbeson	tbc tbcson	qsonic qentrn qcap
1422.	129357.	59634.	1448.	1422.	40545.
	2953.	0.	1505.	1422.	98562.
	57578.	9193.			17482.
1533.	120595.	57066.	1545.	1533.	100325.
	4487.	0.	1631.	1533.	146063.
	56016.	3026.			27246.

temperature of 1533K (2300°F), that a maximum heat throughput in excess of 27 kW can be achieved through a vapor space of 1.22 cm², according to the model. Since the needed heat throughput capability is no more than 10 kW (based on the Phase I reference design for Mach 5 operation at an altitude of 80,000 ft), the test article has a design power-margin-factor of 2.7.

3.7 Description of Test Article

Fig. 14 is a schematic drawing of the test article final configuration. This test article was installed in a large vacuum chamber described in Sec. 5.7 below and shown schematically in Fig. 15. A right-angle segment consisting of one element of an HPRC heat pipe nacelle subsystem is shown. In an actual HPRC system, a second parallel right-angle heat pipe is used to form a square box around the hypersonic engine ramburner and nozzle such that the axes of the heat pipes were transverse (perpendicular) to the axis of the engine. In an actual 112-in.-long ramburner plus nozzle system, the box includes from 140 to 180 heat pipe tubes along the engine axis, depending on the heat pipe cross-section design dimensions. For the Phase II test program, the overall length of the test article along the axial direction of the central heat pipe was scaled to about 78 in. (two 39 in. legs) using the analytic procedure discussed in Sec. 3.5. Fig. 15 also shows that high-frequency, rf-induction coil heating was used to simulate heat input to the exterior surface of the heat pipe nacelle from the hypersonic engine.

Engineering drawings of the HPRC molybdenum-lithium, high-temperature, test article heat pipe are provided in Fig. 16a and Fig 16b. Shown in the figures are the (approximately) 78-in.-long, 0.625-in.-outside diameter, 0.035-in.-wall-thickness, low-carbon, arc-cast (LCAC) molybdenum heat pipe enclosure itself, and details of the electron-beam-welded end-cap closure and, on the other end, the E-beam-welded tube fitting for filling the pipe with the working fluid, lithium metal. Also shown in Fig. 16a and Fig. 16b are details of the inhomogeneous annular screen wick (see Sec. 4.2) and the interior spring. Fig. 16a and Fig. 16b show additional details, such as, extra pipe length and "jelly roll" for lithium overfill, and the inclusion of Zr and Hf gettering material in the form of dimpled disks mounted on the end of the wick plug.

Fig. 14 Schematic drawing of HPRC heat pipe test article

Fig. 15 Schematic drawing of large vacuum chamber used to test HPRC heat pipe test article

Fig. 16a Engineering drawing of HPRC heat pipe test article

Fig. 16b Engineering drawing details of HPRC heat pipe test article

4.0 TEST ARTICLE FABRICATION

4.1 Enclosure Material Selection

The enclosure material chosen for the HPRC heat pipe was low-carbon arc-cast (LCAC) molybdenum. A number of factors influenced the decision to use this material. The primary considerations were compatibility with the Li working fluid, and consistency with HPRC applications phase requirements for high temperature capability and high strength. Additional considerations included weldability, previous experience in working with this material at Los Alamos, cost, and availability. As it turned out, the selected material was obtained within a 3 month period.

Other materials considered for the heat pipe enclosure were the molybdenum alloys TZM and Mo-41Re. Niobium alloys, also considered, are much easier to machine and weld than Mo alloys because of their room temperature ductility. Niobium alloy tubing is used routinely as a heat pipe enclosure at Los Alamos, but this tubing has poorer high-temperature strength and creep properties than Mo alloys. In addition, niobium alloy tubing is not readily available, nor is Mo-41Re tubing, except in the powder-pressed form. Finally, TZM was excluded because previous heat pipe development experience at Los Alamos has shown that if Li is the working fluid in a TZM enclosure, the Li can pull the low, but appreciable (0.01% to 0.04%), carbon content out of the alloy. The consequence of this is TZM strength reduction and carbide buildup in the working fluid.

The best circular cross section match in available LCAC tubing sizes was 0.625-in.-diameter. A wall thickness of 0.035 in. was selected for ease of handling and because Los Alamos had previously bent this size molybdenum tubing in the tube bending device described in Sec. 5.6.

During February 1991, a request for quotation was issued by Los Alamos for 56 ft of drawn, seamless, LCAC molybdenum, 5/8-in.-outside-diameter, 0.035-in.-wall tubing cut in seven 8-ft-lengths. The RFQ included detailed specifications on tubing chemical composition certification, exterior and interior surface cleaning and finish, and packaging for shipment. The RFQ solicited bid responses from three refractory metal supply companies: Rembar Co., Inc., Dobbs Ferry, NY, Schwarzkopf Development Corp., Holliston, MA, and Thermo Electron Co., Wilmington, MA. The anticipated cost was about \$170 per foot (for a total of about \$9520) with an expected delivery of four to five weeks.

During March 1991, a purchase order for the LCAC molybdenum tubing was issued to Thermo Electron Co., Wilmington, MA. This material was received at Los Alamos in good

condition during the week of April 21. Metallurgical processing records were obtained for the molybdenum enclosure material.

4.2 Wick Development

The HPRC wick development effort took place over a period of four months from April 1991 to July 1991.

4.2.1 History of Wick Material. Some comments about the history of the screen material that was used to fabricate the HPRC heat pipe wick are appropriate. The wick material used was overage from previous high-temperature heat pipe research and development programs conducted at Los Alamos. This 400-by-400-mesh, Mo-41Re wire cloth wick material is not commercially available "off-the-shelf." The 0.001-in.-diameter wire strand in this material was drawn to Los Alamos specifications by Metallwerk Plansee GmbH of Reutte, Austria in 1981 to 1982, and woven into screen wick material by Durner Corp., FRG. Duplication of this material is possible, but would involve a long delivery time and would be very expensive.

4.2.2 Wick Design. A bias-wrap wick design was considered because bias wicks are more flexible than rigid sintered wicks and, therefore, bend more reliably. The bias-wrap fabrication technique involved winding narrow strips of the wick material in a spiral fashion on a 0.490-in.-mandrel, while overlapping and spot welding the edges of the seam together. To prevent buckling of the wick in compression on the inner radius during the bending process, or during operation of the heat pipe, a spiral-wound spring was inserted inside the wick. Bias wrap wick samples were prepared for evaluation in 400-mesh stainless steel and in 400-mesh molybdenum (Mo)-41% rhenium (Re) alloy wire cloth. Both materials were successfully joined using a spot welder. There was concern about the possibility of burn-through with the Mo-41Re material.

4.2.3 Spot Welding of the Wick Seams. There were early indications that the Mo-41Re wick material could be successfully joined along a seam using a spot welder. However, there was concern about the possibility of burn-throughs with the Mo-41Re material that could cause a larger than desirable pore. Considerable effort was expended to develop a reliable, repeatable, joining technique for the Mo-41Re wire cloth. Numerous wick spot weld samples were examined under a microscope.

A 55.5-cm-long, four-wrap, sample wick was prepared and pore checked to determine the capillary pumping pressure performance. An effective pore radius of 32 mm to 40 mm is

desired based on design calculations. A systematic approach to determining the most favorable weld parameters, namely, screen cleaning methods, spot welder power setting, spot welder tip geometry, purge gas flow rate, weld location with respect to overlap edge, and spacing between welds was pursued to improve the effective pore radius.

4.2.4 Wick Developmental Samples. A total of six HPRC heat pipe wick samples were fabricated and tested for effective pore radius using a helium (He) and ethyl alcohol bubble point porosity method. An acceptable value of porosity was first achieved using the Number 4 sample, a two-layer bias wrap design. The Mo-41Re wire cloth material for this sample was cleaned but not vacuum-fired at 1830°F (1000°C) as for subsequent samples. An effective pore radius of 38 mm was measured. At the He pressure corresponding to 38 mm, a single leak was observed along the helical seam. From the observations made during testing, it was concluded that the Number 4 wick sample achieved close to the geometric pore radius of 32 mm (expected for 400-mesh screen).

Two more wick designs were investigated before finalizing the wick fabrication procedure. The purpose of these designs was to reduce the length of the welded seam closure and, hence, the number of welds and yet to retain good wick bending properties. Wick sample Number 5 was pore tested successfully and involved a single bias "anchor" wrap to which was attached a 2-1/4 turn straight wrap cut along the screen wire directions. Because the Number 5 design was rather stiff and could buckle during bending, it was concluded that the double bias wrap of Number 4 was a preferred design.

The Number 6 wick design was similar to Number 5 except that the 2-1/4 turn straight wrap was cut out of the sheet screen supply at a 45° angle to the screen wire directions. This idea produced a more flexible wick than Number 5 while retaining the advantage of at least one complete wrap of unwelded screen. However, there was concern about slipping between layers in the bend zone.

After preparation, testing, and evaluation of six HPRC heat pipe wick samples of different design, the Number 4 wick sample design was selected for the HPRC test article. This design features simplicity and flexibility and achieved close to the geometric pore radius of 32 mm (for 400 mesh screen) as discussed previously.

4.2.5 Wick Fabrication. Fabrication of an 88-in.-long wick for the test article using the Number 4, two-layer, bias-wrap design in 400-mesh Mo-41Re alloy screen was completed in July 1991. The 88-in.-length included the two 39 in. legs, an 8 in. dead-space for lithium overfill, and 2 in. extra length to allow for pore checking and trimming as the fabrication progressed. The screen material was inspected for flaws, cut in two 162-in.-strips, each about 1.1-in.-wide, cleaned using an HCl bath, caustic cleaner bath with alcohol rinse, and then

vacuum high-fired at 2600°F (1425°C). When spot welding of the helical seam was complete and the wick passed the pore check, a structurally supporting inner spring of molybdenum wire was wound on a mandrel, inserted inside the wick, and released. The pore check was then be repeated. Next, a specially designed wick end plug, fabricated out of molybdenum was installed in one end of the wick using a Zirconium braze technique. Finally, the wick/plug assembly was rechecked for adequately small effective pore radius, and was then ready for installation in the molybdenum heat pipe enclosure tube.

The evaporator end of the wick was plugged, as discussed later; the condenser end was left open but packed with an 8-in.-long, loosely-packed, "jelly-roll" of 150-mesh molybdenum screen wound on a 0.188-in. mandrel. The purpose of the coarser screen in the bottom of the condenser end (vertical leg) is to help redistribute the lithium working fluid during cool-down of the HPRC heat pipe from 1500 K to 300 K when the lithium specific volume is decreasing, and to hold the lithium "pool" when the pipe is operating at 1500 K. The axial hole in the center of the coarse screen allowed the molten lithium to enter the pipe when it was filled.

4.2.6 Spring Installation. A 0.020-in.-diameter molybdenum wire spring was wound on an undersized 0.437-in.-diameter mandrel (to allow for spring-back to the wick inside diameter of 0.490 in.), inserted into the wick, and released. The purpose of the spring is to prevent distortion or collapse of the wick during phase changes of the lithium working fluid during start-up or shut-down, and during bending.

4.2.7 End Plug Brazing. An HPRC wick end-plug was designed and fabricated. While the design of the molybdenum end-plug was essentially cylindrical, several features required careful consideration. The end-plug must form a leak-tight closure on the evaporator end of the wick. Therefore, the plug diameter is machined to give a tight fit inside the nominal 0.490-in.-inside-diameter of the wick. In addition, the edges of the plug are rounded so as not to damage the wick. The plug was fabricated with a threaded shaft on one end. This shaft accommodated 2 g each of dimpled (for spacing) hafnium (Hf) and zirconium (Zr) gettering disks that are intended to deplete the lithium working fluid of any residual gaseous contaminants such as oxygen and nitrogen that remain in the heat pipe after it is processed, filled, and sealed. (Chemical cleaning, Ar backfilling, protective gloves, and high-temperature bake-out procedures during heat pipe fabrication are used to reduce such contaminants to a minimum prior to sealing the pipe.) Finally, a groove was cut in the circumference of the plug, two wraps of 0.015-in.-diameter Zr wire (cleaned using an aqua regia acid dip) were installed

in the groove, and the plug was inserted and spot welded into one end of the wick. The latter spot welds were done only to keep the plug in place during Zr brazing.

The "best" end of the wick was selected for the plug, that is, for the "evaporator" end of the pipe, because this horizontal leg experiences the greatest pressure difference between the lithium gas and liquid phases during operation.

Zirconium (Zr) brazing of the HPRC wick end plug into the evaporator end of the wick was successfully completed during August 1991, using the apparatus discussed in Sec. 5.1. This task proved to be surprisingly difficult and involved two redesigns of the solid molybdenum plug, eleven developmental braze trials using extra wick material, and two attempts with the actual wick before success was achieved. The large number of braze trials are attributed to the unique wick design and the need to simultaneously satisfy the following critical braze parameters for this situation: good contact between the wick and the plug, the correct braze temperature, sufficient dwell time at the correct braze temperature, enough braze material, cleanliness, and the absence of grooves in the plug under the screen wick.

Good contact between the wick and the plug was obtained by over-sizing the plug and using 0.020-in.-diam. Mo wire wraps around the outside of the wick; however, several braze trials were required to establish good contact. The braze temperature was inferred from remote measurement of the susceptor surface temperature using an optical pyrometer, and was determined from several braze trials to be about 1770°C or about 190°C above the Mo-71%Zr alloy eutectic temperature of 1580°C at which the Zr wire will braze the end-plug to the surrounding wick. (Previous work at Los Alamos suggested that brazing would occur at a lower susceptor temperature of 1690°C instead of 1770°C.) A five minute dwell time was used for all braze trials. Several early braze trials utilized a design with a circumferential groove in the wick plug, which was sized to contain two wraps of 0.015-in.-diam. Zr wire. Each of these early trials resulted in a burn-through of the wick material, so grooved designs were rejected.

4.2.8 Wick Assembly Pore Check. The entire wick plug assembly satisfied a careful experimental pore check at an equivalent capillary radius of 37.5 microns with the spring installed. An analysis of the theoretical performance implications of this data was performed using the Los Alamos code HTPIPE (see Sec. 3.6).

The wick/plug assembly was pore checked and cleaned by baking at 750°C in a furnace and was installed in the molybdenum heat pipe enclosure. Because of the need to oversize the wick end plug, and because the Mo wire wraps were left on the plug, it was necessary to design a short transition section with an inside diameter of 0.690 in. This transition section added one more electron beam circumferential weld to close the heat pipe.

The 0.625-in.-diam. enclosure tube was cut to size, prepared on the ends to accommodate the end caps, chemically cleaned, and baked-out. Photographs of the HPRC heat pipe components were obtained prior to installing the wick in the enclosure tube. The components were then delivered to the Los Alamos electron-beam (e-beam) weld facility for final joining.

4.3 Electron-Beam Welding

A total of eight developmental e-beam weld samples for closure of the test article heat pipe ends were prepared and evaluated during the seven month period from March 1991 to September 1991. While electron-beam welding is done routinely at the Lab, experience has shown that it is advisable to perform and perfect sample welds using the actual material prior to attempting work with the final product.

Meetings were held with technical staff of the Los Alamos e-beam facilities to discuss the weld schedule and weld parameters such as e-beam voltage, current, vacuum, focus condition, travel speed, beam oscillation, as well as precision of fit of the parts, foreign materials (contamination), and preheating of the parts with an unfocused e-beam. For LCAC molybdenum welding, such parameters are important to produce a weld with structural integrity and alkali metal leak tightness.

After the wick/plug assembly was installed in the 0.625-in.-diameter molybdenum heat pipe enclosure tube, end caps were e-beam welded onto the ends of the tube. These welds had to be vacuum tight and structurally sound at 2400°F. One end cap has a fill tube passing through the center of it.

Existing tubing and end cap samples were delivered to the e-beam weld shop during April 1991. The initial e-beam weld, using previously successful parameters for LCAC molybdenum tubing, yielded cracks along the triangular-shaped weld joint. It was believed that the e-beam power was too high and that the weld specimen was being held hot for too long. A second developmental weld test was performed with adjusted parameters and improved results. However, some microcracking was still observed along the enlarged grain structure at the point of the weld. Development in this area continued with a design modification of the welded parts for a third weld test.

Photomicrographs of the third developmental closure welds indicated good weld quality. The samples appeared to be leak tight with no stress cracking. Some flaws (voids and discolorations) were observed, however, and were believed to be the result of contaminants at the machined interface.

A fourth developmental weld test was completed using the new "coffee-can lid" or "cap" design (as opposed to the previously used "plug" design) with e-beam weld parameters

of 125 kV, 6 ma, sharp focus, 60 inches per minute, and 3.4 s total weld time. Photomicrographs of the closure weld indicated good weld quality, but not quite sufficient weld penetration. The fourth sample appeared to be leak tight without stress cracking, and without the flaws observed in the previous weld test, which are believed to be caused by contamination and over-focus of the e-beam. It was decided to continue perfecting the weld parameters for the "coffee-can lid" design.

The fifth weld test included two welds, a circumferential end cap weld and a standoff-annular weld of a fill tube to a fill cap. Inadvertantly, this test utilized raw stock material that had been fabricated using powder metallurgy (PM) techniques. Both welds showed evidence of voids (bubbles) believed to be caused by the release of trapped gases in the PM molybdenum material.

The sixth weld test also involved a PM molybdenum part and, in addition to the presence of voids, the weld had a crack running completely through it.

Weld tests seven and eight utilized LCAC molybdenum components and were judged to be completely successful, clearing the way to perform the HPRC closure welds. These final welds allowed the e-beam current to be fine tuned to 6.5 ma, which produced about 80% weld penetration. Final e-beam welds were successfully performed and leak checked, and the heat pipe was back-filled with argon (Ar) gas to prevent contamination, and weighed.

4.4 Processing, Distillation, and Filling

All HPRC heat pipe components were chemically cleaned, handled with protective rubber gloves, and degassed by baking-out at high temperature. Chemical cleaning of molybdenum parts consisted of caustic cleaning with a sodium hydroxide-peroxide-water mixture, followed by acid cleaning with a hydrochloric acid mixture, and then rinsing with deionized water and then ethanol. The molybdenum parts were then vacuum high-fired at a temperature determined by the application temperature plus about 100°C.

The assembled heat pipe was vacuum degassed on the tilt stand at temperatures above 1300°C. It was then filled with a total of 61 g of pure lithium (Li) using the distillation apparatus (transfer pot, distillation pot, and known-volume pot) and procedures outlined below. The 61 g of Li were inserted into the heat pipe in two increments of 29 g and then 32 g. Preliminary operation of the heat pipe with only 29 g of Li was only partially successful.

HPRC heat pipe lithium (Li) fill calculations were performed to determine the required lithium charge mass. It was desired to supply enough Li to fill the heat pipe annulus and the wick screen material over the entire operating length of the wick plus 15% overfill based on previous experience. In addition, it was determined that an 8-in.-length of extra tubing was required to handle the increase in specific volume of Li when heated to 1500 K. The first

attempt to fill the heat pipe with the calculated value of 43 g of Li lead to only 29 g entering the pipe. This was caused by insufficient heating of the known-volume pot and, hence, an incomplete dump of Li from the known-volume pot into the heat pipe.

The total quantity of Li inserted into the HPRC test article after a second filling was 61 g or 120 cc at about the Li melt temperature. This quantity of Li was inserted to insure complete filling of the extra 8-in.-length, as well as the primary wick and annulus, and to cover the possibility of an incomplete dump of Li from the known volume pot into the heat pipe. The effect of excess Li is to reduce the effective operating length of the heat pipe. (An overfill of 32 cc or 16 g of Li would reduce the operating length about 8 in.) The effect of insufficient Li is to render the heat pipe incapable of full performance.

The apparatus shown schematically in Fig. 17 was used for distillation of Li from the distillation pot to the known-volume pot, and for filling of the heat pipe with Li from the known-volume pot. To begin the distillation process, the entire distillation pot, known-volume pot, and heat pipe system was evacuated and the water-cooled, copper, chill block located below the known volume pot was activated. Next, the known volume pot was heated to a temperature of about 215°C using an electrical heating tape. This temperature exceeds the 181°C melting point of Li. Finally, the distillation pot was heated slowly to about 950°C with an rf coil to produce a Li vapor pressure of about 10 Torr and to initiate a vapor flow of Li from the distillation pot into the known-volume pot. (A valve above the distillation apparatus was closed when the distillation pot reached about 400°C.) After about 2 hours, a visual confirmation of the filling of the known-volume pot was obtained as the Li liquid/vapor interface could be observed rising in the exposed stainless-steel tube connecting the distillation pot to the known-volume pot.

For filling, the rf heating was kept on, maintaining the distillation pot at 950°C. The known-volume pot temperature was raised to about 470°C, and the heat pipe was heated to 250°C over its entire length. Then, the chill block was removed allowing the Li to flow out of the known-volume pot and into the heat pipe.

After filling, the chill block was installed on the inlet stem of the heat pipe, the heat pipe was tipped to an angle of about 20° with the closed end of the pipe above the inlet stem, and heated to 250°C to melt the Li and allow it to flow to form a Li plug in the inlet stem above the chill block. The pipe was then inserted into a portable plastic glove box in an Ar environment so that the distillation apparatus could be removed and a modified valve installed on the fill stem. The valve was then attached to a vacuum pump and opened under vacuum to evacuate the Ar-filled space between the valve seat and the Li plug. The purpose of this procedure is to insure that no gas enters the heat pipe.

Fig. 17 Schematic drawing of lithium distillation apparatus

Finally, the heat pipe was "wet in" for about 12 hours on the tilt stand using a nearly full-length, 82-in.-long, rf coil to heat the pipe to the desired operating temperature of about 1225°C. The purposes of wetting-in are to allow the Li to wet and clean the wick surfaces and to fill the annulus.

4.5 Heat Pipe Bending

Tests using an existing, refurbished, Los Alamos tube bending apparatus were planned and performed after the molybdenum heat pipe enclosure tubing was received. While refractory metal tubing can be bent by skilled commercial machine shops such as the one at Thermo Electron, Wilmington, MA, bending a refractory metal heat pipe with a screen wick installed involves precautions to avoid altering the wick properties. After heat pipe processing and filling with lithium, the pipe was installed in a special apparatus for bending (see Sec. 5.6). The heat pipe was bent at controlled strain rates after being heated above the brittle-to-ductile transition temperature of the recrystallized molybdenum tubing to achieve ductility. Bending was performed at a rate of about 1/3 degree per second.

Three developmental tube bending tests were performed. First, to check out the mechanical integrity of bender components, a piece of 0.625-in.-O.D. stainless tubing was bent without heating. Next, an existing piece of LCAC molybdenum, 5/8-in.-O.D., 0.035-in.-wall tubing was instrumented with 9 chromel-alumel thermocouples, heated to about 320°F (160°C), and successfully bent to a 90° angle. Some spring-back of the tubing was observed. The tube was cooled, removed from the tube bender, and helium leak checked. In a standard set-up, the tube was found capable of sustaining a vacuum pressure of at least 3×10^{-9} Torr, indicating acceptability.

The third and final tube bending test was performed just prior to bending the actual completed heat pipe. A 45 in. length of HPRC stock LCAC molybdenum tubing was cut and heated to 1350°C with an rf coil for about 1 hr to simulate the grain growth and embrittlement of the HPRC heat pipe enclosure material. This bend test served to establish a procedure for bending the actual heat pipe. In particular, Tempilstick brand heat sensitive markers were found to be useful in determining heat pipe and bending apparatus surface temperatures prior to starting the bending process.

After preliminary performance tests in the straight configuration (see Sec. 6.1), the HPRC test article was successfully bent to an angle of 89° using the tube bending apparatus. Prior to bending, the tube bending apparatus and the mounted heat pipe were heated to a temperature in excess of 400°C (752°F) using an oxygen-acetylene torch in order to exceed the brittle-to-ductile transition temperature for the molybdenum enclosure tubing, which had spent

about 8 hr operating above its recrystallization temperature of about 1150°C (2100°F). While some molybdenum surface oxidation occurred in the heated region as a result of the high bending temperature, it was easily cleaned off with ethanol and a fine filament abrasive pad.

5.0 TEST FACILITY APPARATUS AND INSTRUMENTATION

The HPRC feasibility phase testing was conducted in both the North and South Bay experimental areas of Building 1 at Technical Area 46, the main building occupied by the Advanced Engineering Technology Group at Los Alamos. These experimental areas contain adequate floor space, a machine shop, high-frequency rf induction heating power, cooling water, a large-sized vacuum chamber, and other miscellaneous but essential apparatus discussed below, all needed to conduct the HPRC feasibility phase experimental program.

5.1 Wick Development and Testing Apparatus

A number of specialized pieces of equipment were used in the wick development and testing effort. A Weldmatic Unitek Model 1-132-02 spot welder was used to join the helical seams of the wick at an energy setting of about 6 Joules. A small Centorr furnace was used to vacuum bake (clean) small molybdenum heat pipe components such as the wick material, heat pipe end caps, fill stem, etc. at 1300°C. To vacuum bake the 87 in. wick, a 9 1/2-ft.-long, Lindberg Hevi-Duty, multiple unit, Type M-2024, electric furnace was used. The maximum usable temperature of this furnace is 1010°C.

A special-purpose set-up was used to measure the effective pore radius of the wick at various stages of development. In this set-up the wick is plugged on both ends and immersed in an ethanol liquid bath while the wick is internally pressurized with helium to form a gas-liquid interface in the pores of the wick. It can easily be shown that the effective pore radius is inversely proportional to the gas pressure, which is measured using a sensitive pressure gage. The procedure is to slowly pressurize the wick with helium and to record the pressure for which bubbles are first observed. This pressure can be correlated with the largest pore and, hence, the effective pore size of the wick. This pore size is inversely proportional to the maximum capillary pressure capability of the wick and, as such, is a measure of wick performance.

A special Zr vacuum braze facility was designed and fabricated for joining the end-plug to the wick using the procedure discussed in Sec. 4.2.7. The Zr wire in the end-plug groove serves as braze material to join the plug to the wick. The braze facility consists of a 106-in.-long, vertical, 1.5-in.-outside-diameter quartz tube with one sealed end closure and one end closure connected to a vacuum system. Suspended inside the quartz tube are the wick and a surrounding molybdenum susceptor enclosing the wick at the location of the plug. The

susceptor was heated by a water-cooled copper rf-coil, surrounding the quartz tube, to a temperature above the Mo-Zr eutectic temperature of 1580°C at which the Zr wire brazed the end-plug to the surrounding wick.

A Veeco Model MS-17 AM Leak Detector was used on the most sensitive scale, namely, 3×10^{-9} Std cc/s air equivalent, to check for the vacuum tightness of heat pipe components. A 'standard leak' was used to check the performance of the detector prior to each test. The leak detector apparatus was modified to allow backfilling with Ar gas when desired after completion of a leak test.

5.2 Electron-Beam Welding Facility

The electron-beam welding machines at Los Alamos are operated by group MST-6, Materials Technology: Metallurgy, which is located at Technical Area 3. The machine used to perform the HPRC molybdenum tubing and endcap welds was manufactured by Hamilton Standard and had specifications of 3 kW, 150 v, and 50 ma. A special extension to the weld chamber of this machine was used to accommodate the HPRC heat pipe.

5.3 RF Power

High-frequency (100 kHz to 450 kHz) rf induction power was supplied to the test article heaters by an existing new Pillar Industries 50 kW rf generator that was installed on a mezzanine in the North Bay above the HPRC test area.

5.4 Tilt Stand, Quartz Vacuum Tube, and RF Coil

A tilt stand was used to facilitate Li filling wetting-in, and preliminary testing of the 86-in.-long HPRC heat pipe in the South Bay, while the heat pipe was still in the straight configuration. The 12-ft-tall tilt stand with an attached rotatable frame was used to securely mount the HPRC test article for convenience, to allow tipping, and to prevent breaking the fragile 0.25-in.-diameter molybdenum tube part of the fill stem. Whenever the heat pipe was removed from the tilt stand, the fill stem was supported by an attached piece of aluminum angle iron.

Modifications to the tilt stand were made and a special quartz tube for vacuum degassing, wet-in, and testing on the tilt stand was fabricated by the LANL glass blowing shop. The 106-in.-long quartz vacuum tube was installed inside a copper, 5/16-in.-diameter, 82-in.-long, water-cooled rf coil. Both the vacuum tube and the rf coil were fabricated specifically for the HPRC heat pipe. The rf coil featured a shunt at 39 in. so that with the shunt connected, a 39-in.-length of the heat pipe could be heated, and with the shunt disconnected, the heat pipe could be heated over the full 82 in. of the coil.

Next, the HPRC heat pipe was installed inside the quartz tube. Then, a vacuum head for the open end of the quartz tube was assembled from a vacuum "tee" and ultra-high vacuum fittings with 0.25 in. feed-throughs for cooling water inside the quartz tube. A water-cooled "chill block" was attached to the heat pipe fill stem during heating. The entire assembly was mounted on the rotatable frame of the tilt stand as shown in Fig. 18 .

5.5 Processing, Distillation, and Filling Apparatus

Chemical cleaning, Ar backfilling, protective gloves, and high-temperature bake-out procedures during heat pipe fabrication were used to reduce contaminants to a minimum prior to filling or welding the HPRC heat pipe.

An existing stainless steel Li transfer pot was used to transfer Li from the storage container to the distillation pot. This cylindrical container was installed in a special liquid metal transfer rig that includes a high-purity Li storage container. A known charge of Li was moved from the storage container to the transfer pot by evacuation, heating, and inert gas pressurization.

A stainless steel Li distillation pot and known volume pot assembly was fabricated for filling the HPRC heat pipe. The known charge of Li in the transfer pot was transferred to the heavy wall distillation pot, which was rf-heated to 950°C during distillation of Li into the known volume pot. A copper, water-cooled, chill block located on the HPRC heat pipe fill stem prevented entry of Li into the heat pipe until after the known volume pot was filled. The distillation process, heat pipe filling in two steps, and heat pipe wet-in (rf heating of the entire heat pipe at 1400°C to distribute the working fluid throughout the wick and annulus) were completed in September 1991.

Except where rf heating was required, Sybron-Thermolyne, 'BriskHeat,' 120v, 300W, flexible electric heating tapes were used to heat cylindrical pots, valves, and tubing.

5.6 Tube Bending Apparatus

The tube bender, shown in Fig. 19 , consists of a support stand, horizontal fixture to clamp the tubing, cylindrical tube follower, cylindrical guide surface (that determines the final tube bend radius), electric motor and geared speed reduction system, chain drive, and electrical heaters to heat both the tubing and the bender parts that are in contact with the tubing. (During the actual bending of the HPRC heat pipe, the electrical heaters were removed, and an oxygen-acetylene torch was used to heat the components to temperatures much higher than the 160°C obtainable with the electric heaters.) This apparatus was used

Fig. 18 Photograph of HPRC test article on the tilt stand showing vacuum enclosure

Fig. 19 Photograph of LANL refractory metal tube bending apparatus

previously to bend heat pipe tubing for SP-100 space-based nuclear reactor cooling systems.

To bend refractory metal tubing, the tubing is first clamped in the horizontal fixture and then heated above the brittle-to-ductile transition temperature. Finally, the motor is turned on and the tube follower is automatically pulled around the tubing outer surface forcing it to conform to the shape of the guide surface at controlled strain rates.

5.7 Large Vacuum Chamber

An existing large Torr Vacuum Products vacuum chamber was used for testing the HPRC heat pipe in the bent configuration. It is an all stainless steel pressure vessel with approximately 0.25 in.-wall-thickness and measures about 7-ft in diameter and 6-ft in length. The chamber has been fabricated in two sections. One section is stationary and has multiple vacuum-sealed feed-through ports designed for instrumentation, cooling water, or power feed-throughs. The second main volume section rides on four casters and can be rolled away from the stationary section for access to the interior. These two sections are bolted together with a large (about 7-ft-diameter) O-ring seal in-between flanges.

A vacuum pressure of below 1×10^{-6} Torr will be maintained in the test chamber during HPRC testing using a CVI Inc., TorrMaster TM250, model CGR511, cryopump mounted on the stationary section. This large unit is rated at pumping speeds of 9,500 l/s for water vapor and 4,000 l/s for nitrogen and has a capacity of 2,000 standard liters for condensable gases. A Leybold-Heraeus Vacuum Products, Inc., Trivac Model D30A, oil-filled, rotary-vane, mechanical pump, hooked in parallel with the cryopump, was used to rough down the vacuum chamber to a pressure below 100 mm Hg prior to opening a valve connecting to the cryopump.

5.8 Instrumentation

The thermocouples used were Type K (chromel-alumel, color code yellow). These thermocouples were attached to Omega Engineering cold junction compensated, model 650-KF-X-OSS, 10-channel, digital thermocouple readouts. The range of these instruments was -50°C to 125°C , with a resolution of 1°C , and an accuracy of $\pm 0.9^{\circ}\text{C}$.

The ionization gauges used to measure vacuum pressures in various test set-ups were connected to Granville-Phillips, Series 260, Gauge Controllers.

A Pyro, Micro Optical Pyrometer, Model 95, was used to make remote measurements of molybdenum surface temperature. Temperatures were easily readable and repeatable to within $\pm 10^{\circ}\text{C}$. The indicated temperatures were corrected to actual values using tables and assuming a spectral emissivity for unoxidized molybdenum of 0.37.

To facilitate the tube bending operation (see Secs. 4.5 and 5.6) temperature indicating markers called Tempilsticks were used to advantage. These crayon-like markers are applied to a hot surface and indicate that the surface is at least as hot as the indicated value on the Tempilstick if the substance liquifies.

6.0 EXPERIMENTAL RESULTS

6.1 Preliminary Testing of Straight Heat Pipe on the Tilt Stand

Testing of the 86-in.-long HPRC heat pipe was completed during September 1991 in the South Bay experimental area. First tests were run with the heat pipe in the straight configuration, to obtain preliminary data prior to bending to establish a performance (but not performance limits) baseline. These data were obtained to compare with performance data obtained later in the bent L-shaped configuration as there was some risk of damage to the wick as a result of imposed stresses during the bending process. The heat pipe was mounted on the tilt stand (see Sec. 5.4) for these preliminary tests, as shown in Fig. 18

Operational performance testing of the straight HPRC test article began on September 20, 1991. Subsequently, a total of about 11 hr of heat pipe operation was logged on the test article at HPRC design conditions, including both horizontal and tipped orientations. Tipped orientations were used to simulate the effect of gravitational head. Excellent performance was achieved. The heat pipe transported about 7.2 kW (6.8 Btu/s) of power at the operating temperature of 1500K (2240°F). This estimate is based on radiation from the entire 86-in.-long heat pipe at an emissivity of 0.23 (measured at Los Alamos for clean, as-received, molybdenum). The thermal power input of 7.2 kW corresponds to a radial heat flux of 14.5 W/cm^2 ($12.8 \text{ Btu/ft}^2\text{s}$), and an axial heat flux based on the vapor space (wick inside) diameter of 0.491 in., of 5.9 kW/cm^2 ($5200 \text{ Btu/ft}^2\text{s}$). These numbers meet HPRC system theoretical requirements based on applications phase calculations of $2475/2 = 1238 \text{ Btu/s}$ total heat load for the ramburner plus nozzle, carried by one of two HPRC L-shaped nacelle systems consisting of $112 \text{ in.}/(0.625 \text{ in. per heat pipe}) = 179$ adjacent heat pipes.

In additional testing, the straight HPRC test article was run tipped at an angle of 30° , showing a capacity to run with a 39 in. gravitational head. Further, the 30° tipped heat pipe was shut down and successfully restarted from a cold condition, demonstrating a transient

startup. The HPRC test article is shown running at a surface temperature of 1500K (2240°F) in the tipped configuration in Fig. 20.

After these performance tests, the HPRC test article was successfully bent to an angle of 89° using the Los Alamos refractory metal bending apparatus (see Sec. 4.5 and Sec. 4.6). Prior to bending, the tube bending apparatus and the mounted heat pipe were heated to a temperature in excess of 400°C (752°F) using an oxygen-acetylene torch in order to exceed the brittle-to-ductile transition temperature for the molybdenum enclosure tubing, which had spent about 8 hr operating above its recrystallization temperature of about 1150°C (2100°F).

Final performance testing of the bent, L-shaped test article occurred in the large vacuum chamber during October 1991.

6.2 Testing of Bent Heat Pipe in Large Vacuum Chamber

A total of three tests of the bent HPRC heat pipe were conducted in the large vacuum chamber. Rf power was supplied by the new generator described in Sec. 5.3. The first two tests involved starting the heat pipe in a horizontal configuration (the two legs of the heat pipe formed a horizontal plane). The heat pipe started during the first test, but the test was terminated at a heat pipe surface temperature of 1083°C to adjust the power output trim controls on the generator. For the second test, the heat pipe was successfully started and brought up to operating temperature of 1224°C (1497K) and held for about 5 minutes prior to shutting down. The heat pipe is shown installed in the large vacuum chamber in Fig. 21.

Prior to the third bent HPRC heat pipe test, eight Type K (chromel-alumel) thermocouples were fed into the vacuum tank via vacuum tight Conax feed-through hardware and spot welded to the heat pipe from end-to-end in selected locations. The heat pipe was installed in the vertical configuration (the heated leg of the heat pipe was horizontal and the unheated leg was vertical) and rf-heated along 33 in. of the horizontal leg. About 2 hours into the test, a hot spot developed about 6 in. to 8 in. from the end of the horizontal leg at a heat pipe surface temperature of 943°C, and the heat pipe was shut down.

This vertical configuration poses the most severe performance challenge of the heat pipe because in an actual HPRC application, the heat pipe will be heated over the entire active length. In an actual application, some of the working fluid vapor will merely flow across the heated vertical leg and condense on the outer cool radiating surface. Such fluid will then be wicked back to the other side of the heat pipe, rather than be transported back to the horizontal leg through the wick annulus. Future testing of the HPRC heat pipe in the vertical configuration will include more realistic distribution of the heat input.

Fig. 20 Photograph of HPRC heat pipe running at a surface temperature of 2240°F with a 30°C tilt

Fig. 21 Photograph of HPRC bent heat pipe installed in the large vacuum tank

7.0 ASSESSMENT OF HPRC

7.1 Materials Issues

While materials development is not within the scope of the HPRC program, because of the importance of advanced materials and coatings to the concept, a consideration of materials issues is appropriate. Preliminary HPRC applications studies to determine system temperatures, size, and weight assumed that the hot section walls could be fabricated from a material with an operational temperature capability in excess of 3000°F. For the purposes of preliminary analysis and design, carbon-carbon was assumed as a representative hot-section material for the ramburner and nozzle walls because of its light weight and its good strength at the temperatures of interest. It is recognized that considerably more research work is required to develop the necessary oxidation protection coatings for carbon-carbon at such operational temperatures.

However, since the HPRC system is radiatively coupled with the hot section, it is readily adaptable to whatever suitable high temperature material becomes available. This material could be carbon-carbon, a ceramic matrix composite, or some other breakthrough high temperature material. To not research hypersonic propulsion systems that can take advantage of such materials as they become available will guarantee technological stagnation.

HPRC Phase I results indicated that the system size and weight were reasonable for the design basis conditions, but that hot section temperatures would be well in excess of 3000°F. Carbon-carbon materials have adequate strength at such temperatures, but the temperature capability of the required oxidation protection coatings for carbon-carbon have not yet been demonstrated beyond the range 2600°F to 2800°F.

Nevertheless, the development of high-strength materials that can withstand temperatures above 3200°F is a key requirement for NASA-planned propulsion system advances. It is believed by NASA and the engine companies that composite structural materials and oxidation protection coatings will be available for use in aircraft engines at such temperatures within the next 10 to 20 years.^{63,64}

In addition, it was shown in this report (see Secs. 2.3, 2.4) that hot section temperatures can be 2800°F or lower depending on cruise conditions (Mach 4.5 at 90,000 ft altitude) and the specifics of HPRC design (using, for example, a zirconia thermal barrier coating to reduce the heat load).

⁶³Shaw, R. J., Overview of Supersonic Cruise Propulsion Research, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, p. 2-25.

⁶⁴Plencner, R. M. & Knip, Jr., G., High Efficiency Core Technology, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, pp. 32-1 to 32-12.

7.2 Feasibility Assessment

7.2.1 General Assessment. HPRC application studies in Phase I determined that with HPRC and the use of JP-7 fuel, hypersonic aircraft cruise speeds in excess of Mach 5 at 80,000 ft altitude could be attained. The JP-7 fuel is used to absorb the miscellaneous aircraft heat load, while HPRC is used to absorb and dispose of the entire waste heat load of the engine hot section (by radiation to the environment).

Design basis operating conditions at Mach 5 at 80,000 ft altitude and carbon-carbon hot section wall material were selected to enable calculation of system size, weight, and temperatures. However, the reference HPRC design was never intended to be a definitive statement of HPRC system requirements. Rather, the purpose of the reference design was to provide a basis for developing an analytical model with which to obtain preliminary feasibility calculations and to assess flight condition and/or design modifications.

High hot section temperature capability is a goal for several NASA advanced programs, including the High Temperature Materials Technology Program (HITEMP), the Enabling Propulsion Materials Program (EPM) of the High Speed Civil Transport (HSCT) program, and high efficiency gas turbine cores for advanced subsonic propulsion.^{65,66} The EPM program calls for testing of an uncooled ceramic matrix composite combustor liner and an intermetallic matrix composite nozzle.⁶⁵ Recent NASA Lewis Research Center sponsored studies of high efficiency cores for advanced subsonic propulsion systems, performed by General Electric Aircraft Engines and United Technologies Pratt and Whitney, are based on the use of uncooled ceramic matrix composite turbines operating at peak temperatures of about 3200°F.⁶⁶ These studies predicted the availability of these materials by about the year 2010.

The HPRC concept was conceived and assessed in this same spirit. As a completely passive cooling system, HPRC is a revolutionary component technology for hypersonic flight, which can be developed through a reasonable extension of existing heat pipe technology. The ultimate implementation of HPRC can occur as complementary high temperature materials technology reaches a state of technology readiness. The availability of suitable materials depends on the intensity of materials development programs.

⁶⁵Shaw, R. J., Overview of Supersonic Cruise Propulsion Research, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, p. 2-25.

⁶⁶Plencner, R. M. & Knip, Jr., G., High Efficiency Core Technology, Aeropropulsion '91, NASA Conf. Publ. 10063, 1991, pp. 32-1 to 32-12.

7.2.2 Assessment of HPRC Feasibility Phase Results. The Phase II HPRC feasibility study reported here had insufficient information to quantitatively evaluate the integrability of HPRC with HiMaTE engine designs. However, qualitative review of the information that was available shows the potential of HPRC for handling the majority, if not all, of the HiMaTE ramburner and nozzle heat loads.

Radiation heat transfer from the hot combustion gas can increase the heat load in the ramburner of a hypersonic engine.⁶⁷ However, the extent of this increase and its impact on hot section temperatures in an HPRC system require a more detailed thermal analysis than the approximate analysis described here.

Higher heat loads that might result from taking gas radiation into account can be handled through suitable HPRC design modifications including increasing the size of heat rejection panels, using a thermal barrier coating, using thinner and/or higher thermal conductivity hot section walls, and cruising at higher altitudes and/or lower Mach numbers.

The effects of using thermal barrier insulation on the interior walls of the ramburner and nozzle, and of cruising at lower Mach number and higher altitudes were investigated numerically. It was found that these conditions could reduce hot section wall temperatures to and below the 2800°F level, which is consistent with near term carbon-carbon temperature goals. Such temperatures are obtained either by insulating the liner wall with approximately 0.1 in. of thermal barrier coating (TBC), or by cruising at about Mach 4.5 at an altitude of 90,000 ft. The addition of TBC would enable an increase in cruise Mach number and/or a decrease in cruise altitude.

More detailed consideration of heat pipe heat transport requirements, undertaken in Phase II, led to selection of thinner, artery-type wicks and vapor spaced dimensions substantially larger than were used in the initial HPRC heat transfer studies. However, it was found that these design changes have little effect on HPRC system weight and temperatures.

Thermally stabilized JP-7 fuel appears to be best suited as the fuel in a hypersonic engine using the HPRC system. Its thermal stability and capacity is adequate to absorb the aircraft heat load at cruise speeds at least up to Mach 5.

Experiments reported herein demonstrated that HPRC heat pipes can be designed and fabricated, and that such heat pipes can provide the required heat transport and thermal characteristics at the design basis conditions of Mach 5 and 80,000 ft.

Based on calculations and experiments using a newly developed HPRC heat pipe, it was found that the HPRC concept is feasible, with reasonable size, weight, and material temperatures (projected to be available within the next 10 to 20 years), up to Mach 5 flight

⁶⁷Lefebvre, A. H., *Gas Turbine Combustion*, Hemisphere, New York, 1983, Chapter 8.

speeds. The capability of using JP-7 fuel rather than high thermal capacity endothermic or cryogenic fuels increases the economic attractiveness, and hence the commercial viability, of hypersonic flight.

8.0 CONCLUSIONS AND RECOMMENDATIONS

A new concept is proposed for cooling hypersonic aircraft propulsion systems and is discussed in this report.^{68,69} This HPRC concept is attractive, not because it provides more effective cooling than a fuel-cooled engine, but because it shifts the burden of cooling to a passive radiation system. Consequently, HC fuels, such as JP-7, can be used at hypersonic speeds rather than specialized high thermal capacity fuels, such as, hydrogen, methane, and endothermic hydrocarbons. These fuels are much more expensive, and require a completely new fuel supply and fuel handling infrastructure on the ground, as well as complex and heavy pumping systems and heat exchangers or conversion reactors on the aircraft.

Preliminary analytical and experimental results of heat transfer, heat pipe, and system studies reveal that adequate heat transport capability is available using molybdenum-lithium heat pipe technology. Analytical results show that the HPRC system radiator area can be limited in size to the ramburner-nozzle region of the engine nacelle; reasonable system weights are expected; hot section temperatures are consistent with advanced structural materials development goals for the next 10 to 20 years; and system impact on engine performance is minimal.

While it is recognized that the HPRC studies presented in this report are research-oriented and preliminary in nature, assessments of the analytical and experimental results from the feasibility program continue to indicate that the HPRC concept is feasible. Therefore, a follow-on technology program (Phase III) is recommended and outlined in Appendix A. Successful development of HPRC in a technology program can enable a major technological breakthrough that will enhance the economic attractiveness of commercial or military hypersonic flight.

⁶⁸Martin, R., Keddy, M., Merrigan, M., and Silverstein, C., "Heat Pipe Radiation Cooling of Advanced Hypersonic Propulsion System Components," Paper No. AIAA-91-2358, AIAA 27th Joint Propulsion Conference, June 24-26, 1991, Sacramento, Ca.

⁶⁹Martin, R., Merrigan, M., Elder, M., Sena, T., Keddy, E. and Silverstein, C., "Analytical and Experimental Studies of Heat Pipe Radiation Cooling of Hypersonic Propulsion Systems," Paper No. AIAA-92-3809, AIAA 28th Joint Propulsion Conference, July 6-8, 1992, Nashville, TN.

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APPENDICES

APPENDIX A

OUTLINE OF RECOMMENDED FOLLOW-ON TECHNOLOGY PROGRAM (PHASE III)

A.1 Objectives

The technical objective of the proposed technology program is to bring the HPRC concept to technology readiness by performing research and development on critical areas identified during the feasibility phase. In addition, it is proposed to define and develop joint heat pipe applications tasks with the engine companies based on preliminary HiMaTE program coordination meetings previously held. For heat pipe applications such as cooling of hypersonic Single Expansion Ramp Nozzles, resolution of critical HPRC issues are complementary.

A.2 Scope of Work

The HPRC technology program will consist of laboratory simulation of HiMaTE hypersonic engine environments and testing using the existing test article or equivalent. In addition, it will consist of assessment and testing of oxidation protection coatings for high-temperature heat transfer surfaces, evaluation and testing of alternative high-temperature heat transfer surface materials and wicks, and determination of the effects of vibration on heat pipe performance. Further, the program will include definition and preliminary analysis of alternative HiMaTE compatible heat pipe applications, management activities and reporting, and technology transfer to industry.

A.3 Technical Tasks

A.3.1 Task 1 - Laboratory Simulation of HPRC Operation in HiMaTE Engine Environments. HPRC experimental work will be extended to laboratory simulation of HiMaTE engine environments. Discussions with the engine companies will determine HiMaTE temperatures and heat fluxes to be simulated using the existing HPRC test article or equivalent. Radio frequency (rf) induction concentrators will be utilized to facilitate these simulations of HiMaTE engine radiative heat input to an HPRC heat pipe nacelle. Heat transfer operational performance data will be obtained as a function of aircraft operating conditions to validate HPRC capabilities under HiMaTE conditions.

A.3.2 Task 2 - Assessment and Testing of Oxidation Protection Coatings for High-Temperature Heat Transfer Surfaces. The molybdenum heat pipe structure proposed for the HPRC system will require protection against oxidation exposure to the atmosphere. Bench tests will be performed to determine the performance of molybdenum test specimens coated with chemical vapor deposited silicon carbide and/or molybdenum disilicide under conditions of atmospheric oxidation. If niobium materials are evaluated, an alternative protective coating such as iron-chromium silicide (R512E) will be evaluated. The selected coating will be applied to an HPRC heat pipe, and the effectiveness, reliability, and longevity of coatings suitable for hypersonic flight operational conditions will be evaluated. The coated heat pipe will be operated first in a vacuum environment to ensure coating adhesion and heat pipe operation and, subsequently, in an oxidizing environment to simulate HiMaTE engine conditions.

A.3.3 Task 3 - Evaluation and Testing of Alternative HPRC High-Temperature Materials, Working Fluids, and Wicks. In this task, evaluations and tests of alternative heat pipe enclosure materials, working fluids, and wicks will be performed. Discussions with staff at the NASA Langley Research Center and HPRC concept analyses during the feasibility program have suggested using a high-temperature Zirconia liner in the ramburner. Such an insulated system results in calculated carbon-carbon temperatures of below 2700°F and reduced HPRC heat pipe nacelle operating temperatures to about 1900°F. These lower heat pipe operating temperatures suggests the use of alternative enclosure materials and working fluids such as niobium and sodium, respectively. Niobium alloys have been used in aircraft engines since the mid-1960's because they have superior room temperature workability (ductility) and weldability to molybdenum alloys. In either case the refractory metals will have to be coated with a specialized protective material to inhibit high-temperature oxidation and corrosion. Porous powder metallurgy methods of fabricating wicks will be evaluated.

A.3.4 Task 4 - Determination of the Effects of Vibration on HPRC Heat Pipe Performance. The operational adequacy of the HPRC system in a dynamic environment characteristic of hypersonic flight will be demonstrated⁷⁰. Earlier work in this area will be reviewed.⁷¹ The coated heat pipe developed in Task 2 will be operated in the atmosphere with

⁷⁰Deverall, J. E., "Effect of Vibration on Heat Pipe Performance," Los Alamos Scientific Laboratory report LA-37987-MS, 1967

⁷¹Martin, R., Keddy, M., Mettigan, M., and Silverstein, C., "Heat Pipe Radiation Cooling of Advanced Hypersonic Propulsion System Components," Paper No. AIAA-91-2358, AIAA 27th Joint Propulsion Conference, June 24-26, 1991, Sacramento, Ca.

a vibration input designed to evaluate heat pipe performance during exposure to the vibration spectrum created by an operating HiMaTE engine.

A.3.5 Task 5 - Determination of HiMaTE Compatible Heat Pipe Applications.

HiMaTE applications meetings will be held with NASA, GEAE, and P&W to further explore HiMaTE compatible heat pipe cooling applications, to define joint tasks, and to begin preliminary concept analyses and system integration studies for new applications. If directed by NASA, this task will include detailed thermal analyses and system integration studies to evaluate the impact of HPRC on HiMaTE engine designs. (This task could lead to a Los Alamos joint effort with one or both of the engine companies, or engine company funding under the existing NASA HiMaTE program tasks.)

A.3.6 Task 6 - Management Activities and Reporting. Los Alamos will provide all necessary project management activities needed to supervise, direct, and effectively manage this program in an effective and timely manner and within budget constraints. Monthly progress letters and a final summary report will be submitted.

A.3.7 Task 7 - Technology Transfer to Industry. Technology transfer briefings will be provided by Los Alamos to interested engine contractors at the completion of the program.

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APPENDIX B

NOZZLE RELATIONSHIPS⁷²

In order to calculate the heat transfer coefficient at some point along the hot section nozzle, the Reynolds and Prandtl numbers at that point must be determined. This in turn requires that the temperature and mass velocity at that point be known. Thermodynamic principles of nozzle flow are used to calculate this information, given the state of the gas at the nozzle inlet. Consider the converging-diverging nozzle shown schematically in Fig. B1. Conditions at the nozzle inlet are denoted by the subscript *i*, and at a point "e" downstream of the throat. If the downstream point is at the nozzle exhaust, then the subscript "e" denotes conditions at the nozzle exhaust. The subscript "o" denotes stagnation conditions where the gas has been brought to rest isentropically.

Fig. B1 Nozzle Nomenclature

The stagnation enthalpy and temperature in a nozzle are constant as long as the flow is adiabatic (heat is not transferred into or out of the nozzle). Since the nozzle walls are being cooled, the flow with HPRC is not adiabatic. However, since the heat removed by cooling is generally only a small fraction of the total enthalpy content of the nozzle gas flow, the flow is assumed adiabatic for purposes of calculating nozzle heat transfer coefficients. If the gas is isentropic (reversible without frictional losses), the stagnation pressure is also constant.

⁷²Hill, P. G. and Peterson, C. R., Mechanics and Thermodynamics of Propulsion, Addison-Wesley, Reading, MA, 1970.

However, in the less restrictive case of adiabatic flow, the stagnation pressure will drop below the adiabatic value.

The following variables are defined:

$$\alpha = \frac{\gamma - 1}{\gamma}, \quad \text{and}$$

$$\beta = \frac{P_e}{P_{oi}}.$$

The nozzle efficiency η_n is defined

$$\eta_n = \frac{T_{oi} - T_e}{T_{oi} - T_{es}}$$

where T_{es} denotes the absolute temperature at point "e" if the gas had expanded isentropically to that point. Upon solving the above equation for T_e/T_{oi} ,

$$\frac{T_e}{T_{oi}} = 1 - \eta_n \left(1 - \frac{T_{es}}{T_{oi}} \right).$$

Now for an isentropic expansion from the nozzle inlet state,

$$\frac{T_{es}}{T_{oi}} = \beta^\alpha.$$

Therefore, T_e/T_{oi} may be written

$$\frac{T_e}{T_{oi}} = 1 - \eta_n (1 - \beta^\alpha)$$

For a specified downstream pressure ratio β and nozzle efficiency, the corresponding T_e can be found from the above equation.

The associated Mach number M_e is given by the expression

$$M_e^2 = \frac{2}{\gamma - 1} \left[\frac{\eta_n (1 - \beta^\alpha)}{1 - \eta_n (1 - \beta^\alpha)} \right].$$

From the previous equations the above equation may be written

$$M_e^2 = \frac{2}{\gamma - 1} \left(\frac{1 - T_e / T_{oi}}{T_e / T_{oi}} \right)$$

The ratio of stagnation pressures at the nozzle inlet and at point e can be expressed in the form

$$\frac{P_{oi}}{P_{oe}} = \frac{1}{\beta} \left(\frac{T_e}{T_{oi}} \right)^{1/\alpha}.$$

The ratio of nozzle flow area at point e to that at the nozzle inlet can then be calculated from the equation

$$\frac{A_e}{A_i} = \frac{P_{oi}}{P_{oe}} \frac{M_i}{M_e} \left(\frac{1 + \frac{\gamma - 1}{2} M_e^2}{1 + \frac{\gamma - 1}{2} M_i^2} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}}.$$

The inlet Mach number M_i in the above equation can be obtained from the equations in Sec. 2.4.3.

The mass velocity G_e at point "e" is then found by dividing the nozzle mass flow rate W by A_e . Once G_e and the temperature T_e have been obtained from the above equations, the temperature-dependent gas properties and the Reynolds number Re at point "e" may be evaluated. The corresponding Nusselt number and heat transfer coefficient h may then be

calculated, using the equations listed in Sec. 2.4.3. At the nozzle throat, the Mach number is 1.0, which considerably simplifies the above equations. Also, if the nozzle flow is assumed to be isentropic up to and including the throat, the nozzle efficiency η_n to that point is 1.0, which yields further simplification.

APPENDIX C

HOT SECTION PROPERTY AND HEAT TRANSFER DATA

In Tables C1 through C8, data on engine hot section properties and heat transfer characteristics is given for engines cruising at approximately 70% of full power as a function of Mach number and altitude. Some of the data was obtained from another report.⁷³ The remainder was calculated from the nozzle equations presented in Appendix B, and the heat transfer equations in Sec. 2.4.3. The nozzle efficiency was taken to be 1.0 upstream of the throat and 0.98 downstream of the throat. The data listed in Tables C1 through C8 was used in the HPRCc code to calculate HPRC system temperatures for various cruise conditions.

⁷³Hartsel, J. E., "High Speed Propulsion Assessment: Customer Performance and Installation Data," General Electric Co., Cincinnati, OH, 1987

Table C1. Hot Section Properties and Heat Transfer Coefficients

Mach number	5		
Altitude, ft	80,000		
Dynamic pressure, lb/ft ²	1000		
Air Flow rate, lb/sec	381		
Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.085
Gas temperature, °F	4200	3681	2427
Recovery temperature, °F	4211	4175	4031
Pressure, psia	116.8	64.6	9.93
Flow area, ft ²	12.56	2.94	7.24
Diameter, ft	4.00	1.936	3.035
Mach number	0.1395	1.003	2.22
Prandtl number	0.972	0.863	0.741
Reynolds number	2.55x10 ⁶	5.56x10 ⁶	4.36x10 ⁶
Nusselt number	2766	4952	3857
Heat transfer coeff.(inf. tube), Btu/hr-ft ² -°F	49.03	169.6	67.99
Heat transfer coef.,Btu/hr-ft ² -°F	98.1	204	81.6
Mean heat transfer coef. (nozzle), Btu/-hr-ft ² - °F	144.4		

Table C2. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.5		
Altitude, ft	80,000		
Dynamic pressure, lb/ft ²	810		
Air flow rate, lb/sec	286		
Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.114
Gas temperature, °F	3928	3439	2417
Recovery temperature, °F	3943	3898	3785
Pressure, psia	70.9	39.2	8.08
Flow area, ft ²	12.56	3.48	7.08
Diameter, ft	4.00	2.10	3.003
Mach number	0.1653	1.003	2.05
Prandtl number	0.944	0.831	0.741
Reynolds number	1.945x10 ⁶	3.94x10 ⁶	3.30x10 ⁶
Nusselt number	2185	3678	3061
Heat transfer coeff.(inf. tube), Btu/hr-ft ² -°F	36.71	113.1	54.33
Heat transfer coef.,Btu/hr-ft ² -°F	73.4	135.7	65.2
Mean heat transfer coef. (nozzle), Btu/- hr-ft ² -°F	101.3		

Table C3. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.0
Altitude, ft	80,000
Dynamic pressure, lb/ft ²	640
Air Flow rate, lb/sec	214

Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.173
Gas temperature, °F	3680	3219	2483
Recovery temperature, °F	3702	3646	3569
Pressure, psia	40.2	22.2	6.95
Flow area, ft ²	12.56	4.45	7.07
Diameter, ft	4.00	2.38	3.001
Mach number	0.214	1.003	1.808
Prandtl number	0.863	0.793	0.744
Reynolds number	1.510x10 ⁶	2.70x10 ⁶	2.45x10 ⁶
Nusselt number	1717	2654	2396
Heat transfer coeff.(inf. tube), Btu/hr ft ² -°F	28.46	69.36	43.20
Heat transfer coef., Btu/hr-ft ² -°F	56.9	83.2	51.8
Mean heat transfer coef. (nozzle), Btu/ hr-ft ² -°F	68.2		

Table C4. Hot Section Properties and Heat Transfer Coefficients

Mach number	5		
Altitude, ft	90,000		
Dynamic pressure, lb/ft ²	621		
Air Flow rate, lb/sec	238		
Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.0870
Gas temperature, °F	4215	3694	2449
Recovery temperature, °F	4226	4200	4045
Pressure, psia	73.4	40.6	6.39
flow area, ft ²	12.56	2.92	7.08
Diameter, ft	4.00	1.929	3.002
Mach number	0.1384	1.003	2.21
Prandtl num,ber	0.972	0.916	0.741
Reynolds number	1595x10 ⁶	341x10 ⁶	2.75x10 ⁶
Nusselt number	1901	3409	2669
Heat transfer coef.(inf. tube), Btu/hr-ft ² -°F	33.70	120.7	47.57
Heat transfer coef.,Btu/hr-ft ² -°F	67.4	144.8	57.1
Mean heat transfer coef. (nozzle), Btu,- hr-ft ² -°F	101.9		

Table C5. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.5		
altitude, ft	90,000		
Dynamic pressure lb/ft ²	503		
Air Flow rate, lb/sec	178		
Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.113
Gas temperature, °F	3942	3452	2421
Recovery temperature, °F	3956	3913	3811
Pressure, psia	44.5	24.6	5.03
Flow area, ft ²	12.56	3.45	7.08
Diameter, ft	4.00	2.097	3.002
Mach number	0.1643	1.003	2.06
Prandtl number	0.930	0.831	0.763
Reynolds number	1.218x10 ⁶	2.46x10 ⁶	19.02x10 ⁶
Nusselt number	1496	2524	1991
Heat transfer coeff.,(inf. tube), Btu/hr-ft ² -°F	25.73	77.75	38.67
Heat transfer coef.,Btu/hr-ft ² -°F	51.5	93.3	46.4
Mean heat transfer coef. (nozzle), Btu/ hr-ft ² -°F	70.4		

Table C6. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.0		
altitude, ft	90,000		
Dynamic pressure, lb/ft ²	397		
air flow rate, lb/sec	133		
Parameter	Combustor	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.164
Gas temperature, °F	3680	3219	2452
Recovery temperature, °F	3701	3646	3564
Pressure, psia	25.3	13.99	4.15
Flow area, ft ²	12.56	4.31	7.06
Diameter, ft	4.00	2.34	2.999
Mach number	0.207	1.003	1.840
Prandtl number	0.863	0.793	0.741
Reynolds number	0.939x10 ⁶	1.711x10 ⁶	1539x10 ⁶
Nusselt number	1174	1842	1649
Heat transfer coeff.,(inf. tube), Btu/hr-ft ² -°F	19.46	48.96	29.42
Heat transfer coef.,Btu/hr-ft ² -°F	38.9	58.8	35.3
Mean heat transfer coef. (nozzle), Btu/- hr-ft ² -°F	47.5		

Table C7. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.5		
Altitude, ft	75,600		
Dynamic pressure, lb/ft ²	1000		
Air flow rate, lb/sec	363		
Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.000	0.553	0.114
Gas Temperature, °F	3921	3433	2412
Recovery temperature, °F	3936	3889	3778
Pressure, psia	89.7	49.6	10.23
Flow area, ft ²	12.56	3.48	7.08
Diameter, ft	4.00	2.10	3.003
Mach number	0.1653	1.003	2.05
Prandtl number	0.927	0.817	0.741
Reynolds number	2.49x10 ⁶	5.06x10 ⁶	4.19x20 ⁶
Nusselt number	2646	4467	3704
Heat transfer coeff.(inf. tube), Btu/hr-ft ² -°F	45.45	135.9	65.99
Heat transfer coef.,Btu/hr-ft ² -°F	90.9	163.1	79.2
Mean heat transfer coef. (nozzle), Btu/ hr-ft ² -°F	122.4		

Table C8. Hot Section Properties and Heat Transfer Coefficients

Mach number	4.0
Altitude, ft	71,700
Dynamic pressure, lb/ft ²	1000
Air flow rate, lb/sec	323

Parameter	Combustor & Nozzle Inlet	Nozzle Throat	Nozzle Exit
Pressure/total pressure	1.00	0.553	0.166
Gas temperature, °F	3669	3209	2451
Recovery temperature, °F	3691	3635	3553
Pressure, psia	60.4	33.4	10.03
Flow area, ft ²	12.56	4.37	7.11
Diameter, ft	4.00	2.36	3.009
Mach number	0.210	1.003	1.83
Prandtl number	0.863	0.793	0.741
Reynolds number	2.28x10 ⁶	4.12x10 ⁶	3.72x10 ⁶
Nusselt number	2387	3720	3340
Heat transfer coeff., (inf. tube) Btu/hr-ft ² -°F	39.56	98.04	59.39
Heat transfer coeff., Btu/hr-ft ² -°F	79.1	117.6	71.6
Mean heat transfer coef. (nozzle), Btu/- hr-ft ² -°F	95.46		

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