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ANALYTICAL AND EXPERIMENTAL STUDIES OF HEAT PIPE RADIATION COOLING OF HYPERSONIC PROPULSION SYSTEMS

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Abstract

Preliminary, research-oriented, analytical and experimental studies were completed to assess the feasibility of using high-temperature heat pipes to cool hypersonic engine components. This new 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 an external heat pipe nacelle. For propulsion systems using heat pipe radiation cooling (HPRC), it is possible to continue to use hydrocarbon fuels into the Mach 4 to Mach 6 speed range, thereby enhancing the economic attractiveness of commercial or military hypersonic flight. In the second-phase feasibility program recently completed, we found that heat loads produced by considering both convection and radiation heat transfer from the combustion gas can be handled with HPRC design modifications. The application of thermal insulation to ramburner and nozzle walls was also found to reduce the heat load by about one-half and to reduce peak HPRC system temperatures to below 2700°F. In addition, the operation of HPRC at cruise conditions of around Mach 4.5 and at 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.

Background and Concept

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, integrated, combined-cycle engines such as the turbofanramjet are favored over multiple engine types for different speed regimes.[1-5] A number of propulsion system critical component technologies for such engines 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 High-Speed Propulsion Assessment (HiSPA) and High Mach Number Technology Engine (HiMaTE).[6]

HPRC is a newly proposed technique for passively cooling the hot section of a hypersonic engine, thus eliminating the need to use the engine fuel as the coolant.[7-9] For an HPRC system application at Mach 5, the entire engine heat load, or about two-thirds of the total 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. 1.

HEAT PIPE RADIATOR CONCEPT:
TRANSVERSE TRANSPORT - NACELLE SIDE REJECTION

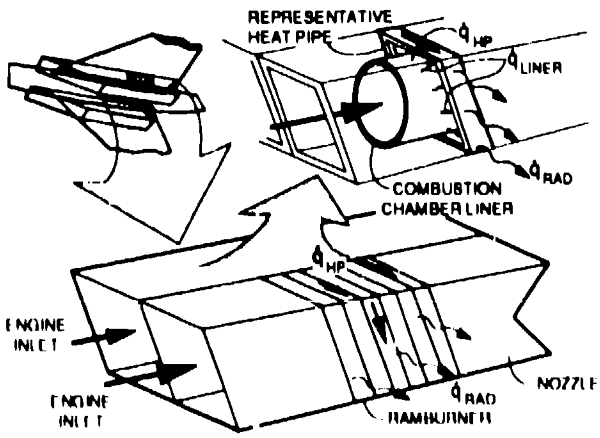

Fig. 1 HPRC concept

As indicated in Fig. 1, 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.

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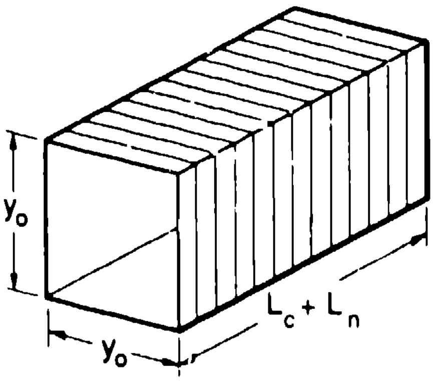
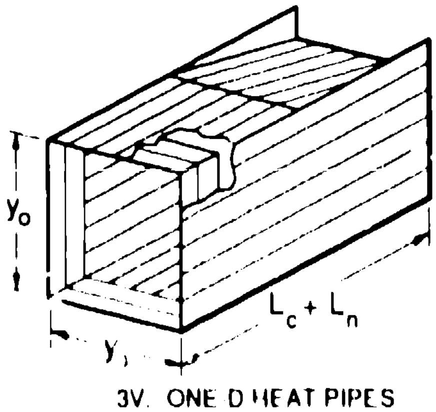
In this configuration, the cylindrical ramburner and nozzle walls are fabricated from special high-temperature materials, such as (but not limited to) carbon-fiber-reinforced carbon matrix (carbon-carbon)[10] or silicon-carbide-whisker-reinforced molybdenum disilicide matrix composites.[11] These walls 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 outerside 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 a later section, this report will show that the HPRC test article consists of one scaled element from one of these L-shaped subsystems.



3F. ONE-D HEAT PIPES



3V. ONE-D HEAT PIPES

Fig. 2 Two HPRC configurations

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 hydrocarbon (HC) fuel prior to combustion. Thus, HPRC is proposed as a cost-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 the hot-section components. Hybrid thermal management systems using heat pipes combined with cryogenic or endothermic fuel cooling are possible.

Previous Work

The HPRC concept, the HPRC thermal analysis computer code, and results from the first (applications) phase of the HPRC program are described elsewhere.[7-8] 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.[12-14] This information defines the representative engine.

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 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,[15] calculations.

In the previous applications phase, application of the HPRC concept resulted in reasonable sizes and weights, but at relatively high material temperatures up to Mach 5 flight speeds. These preliminary conceptual design studies suggested that the engine ramburner and nozzle walls be fabricated from a material such as carbon carbon, which could operate at temperatures in excess of 3000°F. However, as shown in Ref. 9 and in this paper, 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 consistent with short-term advanced high-temperature materials goals.

Analytical and experimental results from the second (feasibility) phase of the HPRC program are reported here.

Analytical Results

At the high, uniform combustion gas recovery temperature of 4200°F assumed in this study, thermal radiation from the combustion gas is assumed to be a significant contributor to the gas-side surface heat load, along with convective heat transfer.[15] Only the convective heat transfer from the combustion gas to the ramburner and nozzle walls was considered in the applications phase.[7-8]

In the present study, HiMaTE ramburner and nozzle liner gas-side surface heat flux data (including the effects of gas radiation) provided by GEAE were used to perform additional calculations with the HPRC thermal analysis code.[9] Results of these calculations are summarized in Fig. 3 and compared with the original results, but not those that included gas radiation. In either case, the calculated ramburner and nozzle wall temperatures exceed the level of about 2800°F currently available with high-temperature, composite materials coated with oxidation-resistant films.[10] However, the development of high-strength materials that can withstand temperatures above 3200°F is a key requirement for NASA-planned propulsion system advances. NASA and the engine companies believe 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.[17-18]

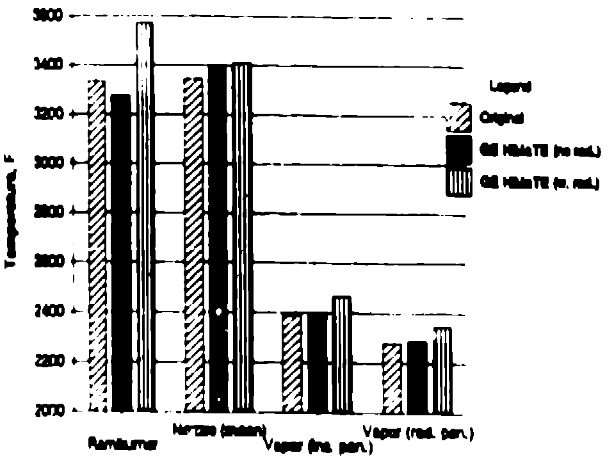


Fig. 3 HPRC system temperatures.

In addition, if necessary, HPRC system and hot-section temperatures can be lowered to 2800°F or less by using one or more of the following options (1) Increasing the size of the HPRC radiating nacelle panels; (2) reducing the thickness of the hot-section walls; (3) increasing the thermal conductivity of the hot-section walls; (4) adding a thermal insulating layer to the gas-side surface of the hot section walls; (5) cruising at Mach 5 or less at an altitude above 80,000 ft (lower dynamic pressure); or (6) cruising at a dynamic pressure of 1000 psf and a Mach number less than Mach 5 (lower altitude). The effects of Options (4) through (6) are examined below.

One method for reducing the hot-section temperatures of the original HPRC system design is to thermally insulate the hot-section walls, thereby increasing the thermal resistance through the heat transfer path. A heat transfer calculation was performed using a thermal barrier coating (TBC) on the gas-side surface of the ramburner and nozzle walls. TBC consists of zirconia, partially stabilized with 8% wt yttria. TBC has been used with considerable effectiveness to reduce the heat load on internally cooled turbine vanes of jet engines.[19] The thermal conductivity of this material is about 0.5 Btu/h-ft-°F at temperatures of interest.[20] The effect of TBC is to increase the thermal resistance and reduce the thermal conductance at the combustion-gas hot-section wall interface. 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/ft^2-h-°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. Using these thicknesses of TBC, the HPRC thermal analysis code was used to calculate the new system temperatures and weights summarized in Table 1. Results from the previous applications phase calculations[7-8] are also shown in Table 1 for comparison.

As shown in Table 1, the addition of TBC reduced the heat load on the radiating panel by almost one-half from 1214 Btu/s to 627 Btu/s. This results in a substantial reduction in temperatures 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/in.^{3} for zirconia,[21] the principal constituent of TBC, 370 lb of fully dense TBC is required on the hot-section inner surface. The trades in using TBC are the added weight of the TBC plus the need for the development of an effective thermally resistant bond between TBC and the liner wall material.[20]

Results from the previous HPRC applications phase heat-transfer analyses[7-8] were based on representative HiSPA engine data for a dynamic pressure of 1000 lb/ft^2, a flight speed of Mach 5, and an altitude of 80,000 ft. The effect of other cruise conditions on HPRC system temperatures was calculated.[9] For the new cruise conditions, new hot-section dimensions and new internal and external heat-transfer coefficients were calculated and input to the HPRC thermal analysis computer code. The maximum ramburner temperatures resulting from these calculations are summarized in Fig. 4. For each of the three Mach number cases in Fig. 4, the length of the HPRC nacelle is equal to the hot section length (112 in. at Mach 5, 104 in. at Mach 4.5, and 90 in. at Mach 4).

Table 1.
Calculations of Effect of TBC on HPRC Temperatures and Weight

	New	Previous
Combustor 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/s	270	521
Heat load on transition section, Btu/s	351	670
Heat load on radiating panel, Btu/s	627	1214
HPRC weight, lb	630	
Weight of TBC, lb	370	
Weight of HPRC + TBC, lb	1000	

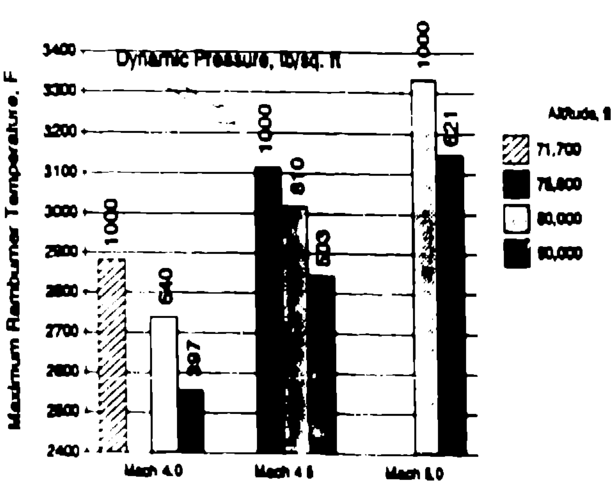


Fig. 4 Effects of altitude and Mach number.

From Fig. 4, it is evident that the cruise Mach number and altitude have a strong effect on the maximum ramburner temperature. For example, 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 (q) of 1000 lb/ft^2 and an altitude of 75,600 ft, drops to slightly over 3100°F. As the altitude increases 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, q = 1000 lb/ft^2, and at an altitude of 71,7000 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. At Mach 4 and Mach 4.5, the maximum ramburner temperatures would be lower if the heat pipe nacelle length had been maintained at the reference design value of 112 in.

Experimental Results

HPRC nacelles consist of adjacent heat pipe elements, as illustrated in Figs. 1 and 2. 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.[22-24] The following procedures were used to optimize the HPRC heat pipe for performance and weight: (1) identified alternative wick structures and evaluated their relative performances, (2) selected and optimized one or more suitable wick structure(s), (3) designed heat pipes for maximum thermal performance and minimum weight, and (4) detailed and evaluated the performance of the selected design. 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.[7-8] Thermal performance analyses were done by means of hand calculations and computer-aided calculations using HTPIPE.[15]

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 shown in Fig. 2 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. 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.

Figure 5 is a schematic drawing of the test article final configuration. This test article was installed in the large vacuum chamber described in Ref. 9 and shown schematically in Fig. 6. A right-angle segment consisting of one element of an HPRC heat pipe nacelle subsystem is shown. For the idealized HPRC system of Fig. 2, 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 are transverse (perpendicular) to the axis of the engine. For the representative engine, 112-in.-long ramburner plus nozzle system, the box includes 140 to 180 heat pipe tubes along the engine axis, depending on the heat pipe cross-section design dimensions. For the feasibility phase test program, the overall length of the heat pipe was scaled to about 78 in. (two 39 in. legs). Figure 6 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.

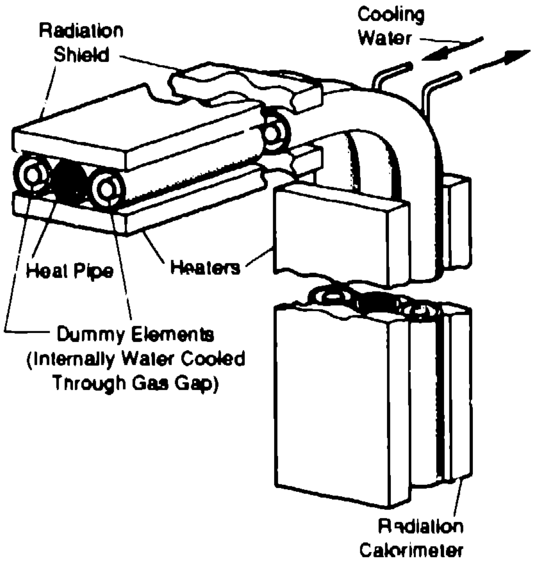


Fig. 5 HPRC test article configuration

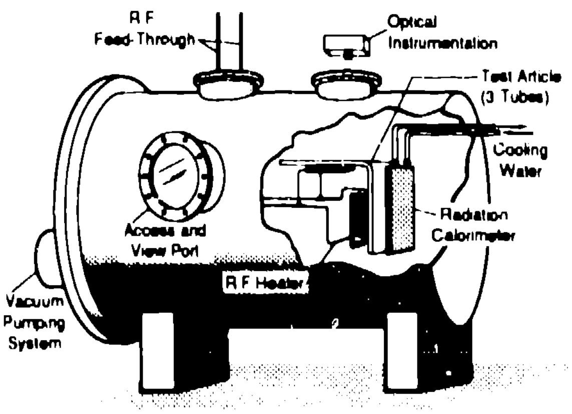


Fig. 6 HPRC test facility

The enclosure material chosen for the HPRC test 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 lithium working fluid, and consistency with HPRC applications phase requirements for high-temperature capability and high strength. Additional considerations included electron-beam weldability, previous experience in working with this material at Los Alamos, cost, and availability.

A bias-wrap, annular wick design was used 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 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. The wick was fabricated from 400-mesh by 400-mesh molybdenum (Mo)-41% rhenium (Re)-alloy wire cloth. The completed wick satisfied an experimental pore check at an equivalent capillary radius of 37.5 mm with the spring installed.

The heat pipe handling, cleaning, bake-out (above 2400°F), filling with 61 g of pure lithium (by distillation at 1740°F), and "wet-in" procedures are described in Ref. 9. Heat pipe fill calculations were performed to determine the required lithium charge mass. Enough lithium was desired 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, an 8-in.-length of extra tubing was required to handle the increase in specific volume of lithium when the lithium was heated to the operating temperature of 2240°F.

Reference 9 provides detailed descriptions of the apparatus and instrumentation used for wick development and testing, electron beam welding, suppling rf-induction power, supporting and tilting the heat pipe, producing the vacuum, processing, distillation, filling, and tube bending.

Operational performance testing of the straight 86-in.-long HPRC heat pipe began on September 20, 1991. First tests were run with the heat pipe in a 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 because there was some risk of damage to the wick as a result of imposed stresses during the bending process. The heat pipe was installed inside a special vacuum tube mounted on a tilt stand for these preliminary tests.

Subsequently, a total of about 11 h of heat pipe operation was logged on the test article at HPRC design conditions, including both horizontal and tipped orientations. Heating was supplied by a spiral, 39-in.-long, water-cooled, copper rf coil. 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 Btu/ft^{2}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 Btu/ft^{2}s). These numbers meet HPRC system theoretical requirements based on applications phase calculations of 2475/2 = 1238 Btu/s total heat load for the ramburner plus nozzle, carried by one of two HPRC L-shaped nacelle systems consisting of 112 in./(0.625 in. per heat pipe) = 179 adjacent heat pipes.

In additional testing, the straight HPRC test article was run while tipped at an angle of 30°, with the evaporator section at the top, 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.

After these performance tests, the HPRC test article was successfully bent to an angle of 89° using special refractory metal, 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 h operating above its recrystalization temperature of about 1150°C (2100°F).

Final performance testing of the bent, L-shaped test article occurred in a large vacuum chamber during October 1991. A total of three tests of the bent HPRC heat pipe were conducted in the large vacuum chamber. 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 rf generator. For the second test, the heat pipe was successfully started and brought up to operating temperature of 1224°C (2235°F) and held for about 5 minutes prior to shutting down.

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 feedthrough hardware and were 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 to 8 in. from the end of the horizontal leg, at a heat pipe surface temperature of 943°C (1730°F), and the heat pipe was shut down.

This vertical configuration posed the most severe performance challenge to 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 being transported against gravity back to the horizontal leg through the wick annulus. Future testing of the HPRC heat pipe in the vertical configuration should include a more realistic distribution of the heat input.

Concluding Remarks

A new, research-oriented cooling concept is proposed for cooling hypersonic aircraft propulsion systems. 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 can be used at hypersonic speeds rather than specialized high thermal capacity fuels, which are much more expensive and require a completely new fuel supply and fuel handling infrastructure. Preliminary results of heat transfer, heat pipe, system, and experimental 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.

As shown, radiation heat transfer from the hot combustion gas can increase the heat load in the ramburner of a hypersonic engine. 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. These conditions reduced hot- section wall temperatures to 2800°F or below, which is consistent with near-term coated 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.

The results of experiments reported herein demonstrate 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. However, much development work remains to be done to bring the HPRC concept to a state of technology readiness.

Acknowledgments

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