Engineering Notes

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Application of Heat-Pipe Technology to Rocket-Engine Cooling

S. E. Stephanou* and T. E. Ward†
McDonnell Douglas Astronautics Company, Western Division, Santa Monica, Calif.

AND

J. S. Holmgren‡
Donald W. Douglas Laboratories, Richland, Wash.

In recent spacecraft and satellite applications, heat pipes have been used for heat rejection from power distribution and electronic systems.[1,2] There has also been interest in the possibility of applying heat-pipe technology to the cooling of rocket nozzle throats.[3]

Figure 1 shows the concept considered herein. The working fluid is boiled or evaporated in the evaporator section (next to the heat source) and the vapor travels to the condenser region where it gives up an amount of heat equal to the heat of vaporization of the working fluid. This heat is rejected to a space radiator. Capillary forces return the liquid to the evaporator through the wicking structures. The device is nearly isothermal since the only temperature drops are those associated with thermal conduction through the thin container wall and liquid layer, and those associated with the boiling and condensing phenomena.

While the heat fluxes through the heat pipe from the evaporator to the condenser for rocket-engine applications are within the present state-of-the art, the fluxes through the evaporator wall are significantly above the highest values demonstrated to date (2 Btu/in.^{2}/sec).[4] Temperatures also are considerably higher than any reported for heat-pipe applications to date, posing material as well as design and operational problems. In this Note, the capability of heat pipes to cool a relatively small thruster (141 lb) using a high-energy propellant system (OF_2/B_2H_6) is evaluated. The rocket is assumed to have a chamber pressure of 100 psia and a throat diameter of 1 in. The thrust chamber is assumed to be film-cooled, the expansion section of the nozzle is radiation-cooled and the throat region is cooled by heat pipes.

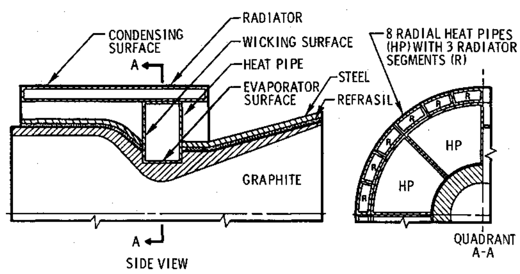

Fig. 1 Heat-pipe cooling of rocket throat using space radiators.

Heat-Pipe Design and Operation

The general heat-pipe configuration shown in Fig. 1 has a total of 8 primary heat pipes located radially around the rocket throat, leading to a cylindrical set of space radiator segments (24) which are also heat pipes. The evaporator sections of the heat-pipe space radiators are adjacent to the condenser sections of the primary heat pipes connected to the nozzle throat. The radiator is segmented in such a manner that the puncture of a single segment will not result in failure of the cooling system. Other radiator designs are possible; e.g., the engine throat heat pipes could be extended radially until sufficient radiator surface is provided in the condensing sections using a single toroidal radiator tube. The design in Fig. 1 can accommodate about 30 Btu/sec from the nozzle throat region. A radiator operating at about 3000°F with this configuration will require a surface area of about 0.60 ft^2.

Consideration of the physical properties and heat-transfer characteristics of candidate high-temperature working fluids led to the selection of silver. Silver has a relatively low vapor pressure at 3500°F, is less corrosive than indium (the other prime candidate), and data are available on heat pipes operating at 3600°F with silver.[5] In addition, the thermal conductivity of liquid silver is an order of magnitude higher than that of indium; high thermal conductivities are required in order to achieve the high evaporator fluxes which occur in the throat region of the rocket nozzle [5 to 8 Btu/sec-in.^{2}]. These high fluxes represent the major constraint on the heat pipe. The highest recorded evaporator heat flux (about 2 Btu/in.^{2}/sec) was obtained with a silver tungsten heat pipe operating at 2000°C.[5]

The containment of molten metal at temperatures above 3500°F is a difficult problem. Long-duration (>1000 hr) high-temperature tests using tungsten, tungsten-rhenium, and rhenium as the container material, and lithium, silver, and indium as working fluids have shown tungsten to be the

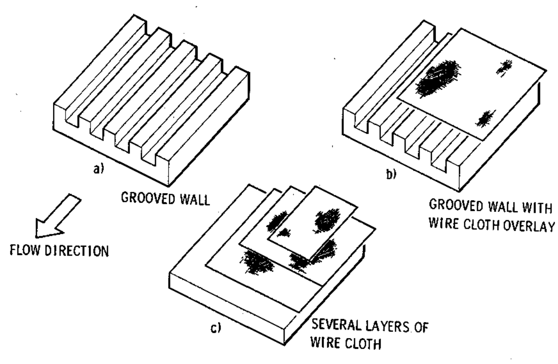

Fig. 2 Typical wicking material configurations.

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* Chief of Propulsion Research. Associate Fellow AIAA.
† Research Engineer.
‡ Research Engineer. Member AIAA.

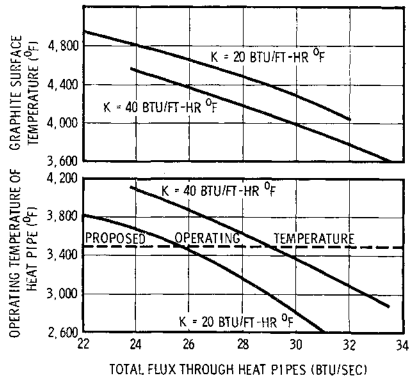

Fig. 3 Heat-pipe operating temperature and graphite surface temperature for high-Q design.

most durable. Indium was corrosive and completely dissolved the tungsten wicking material in 1000 hr; silver and lithium were considerably less corrosive. The combination of liquid silver as the working fluid with a tungsten container and tungsten wicking material appears to be the most advantageous combination for rocket-engine applications. Typical wicking configurations are shown in Fig. 2. Open grooves, screen-covered grooves, and layers of fine screen have all been used successfully as wicking structures. Since liquid flow lengths are relatively short in this application, a screen wick is the easiest and most straightforward approach from a design and implementation standpoint.

Graphite and pyrographite were selected as the candidate throat materials, based on their ability to withstand the corrosive action of fluorine-containing combustion species, their high heat conductivity, and their structural properties at high temperature.

Analytical Basis

The heat-transfer calculations were made for the steady-state conditions attained in a motor firing of several seconds. A three-dimensional model was utilized for the analysis based on the treatment of Mayer[6]; heat-transfer coefficients were calculated assuming that the rocket exhaust gases were at chemical equilibrium throughout the nozzle. The use of graphite as the exposed throat material, refrasil as the insulator, and steel as the outer structural material for the "high-Q" design for use with a space radiator is depicted in Fig. 1. To minimize the size of the radiator, the temperature of the radiating surfaces is as high as the thermal properties of the materials used will allow. An alternative "low-Q" design was developed for use with heat exchangers and other cooling methods where the total heat rejection capability is more limited. In this design, the heat flux through the heat pipes is reduced by placing a layer of insulation between the heat pipe and graphite, reducing the temperature differential between exhaust gases and nozzle surfaces, and making more efficient use of the high-ablation temperature of graphite. In both designs, the insulating refrasil layer protects the outer steel case from the hot graphite.

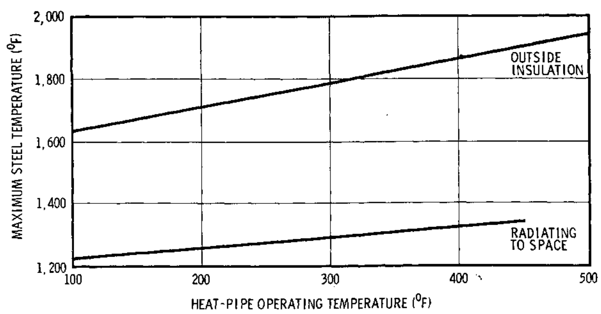

Fig. 4 Maximum temperature of the steelcase (low-Q design).

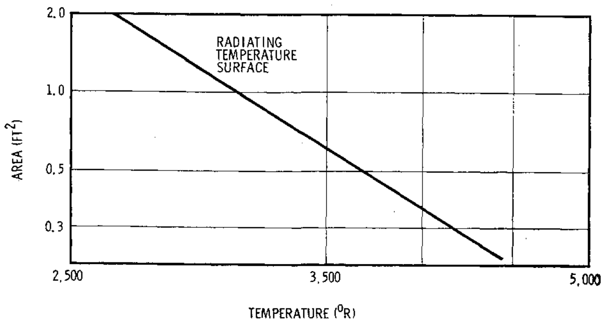

Fig. 5 Required radiator area vs radiator surface temperature.

Heat-Transfer Analysis

Operating temperatures of the heat pipe for the high-Q design are shown in top part of Fig. 3 for a range of graphites with conductivities of 20 and 40 Btu/ft-hr-°F and for various heat flux values. For the assumed maximum heat-pipe operating temperature of 3500°F, the total heat flux through the heat pipes is 25 to 29 Btu/sec, which is equivalent to a heat-transfer rate of 4 to 5 Btu/in.^{2}-sec. For motor firings of unlimited duration (steady-state conditions) the steel case requires additional cooling of 0.4 Btu/sec in the throat area. This can be accomplished by regenerative cooling techniques, or additional heat pipes upstream and downstream of the throat area.

The calculated heat flux from the exhaust gases to the graphite surface exhibits a sharp peak of 5.1 Btu/in.^{2}-sec at the throat. It falls to 0.5 Btu/in.^{2}-sec at points 1.2 in. upstream and 1.8 in. downstream of the throat. The heat pipe cools this 3-in. region between these points. The lower part of Fig. 3 shows the maximum surface temperature of the graphite throat insert vs the flux through the heat pipe.

The investigation for the low-Q design showed that the heat flux was reduced to approximately 10% of the high-Q design by the changes imposed on the system. It was also found that in this low-Q design, the steel case could not be cooled sufficiently by the heat pipes located at the throat with the insulation thicknesses chosen for this model (0.4 in. of refrasil). As with the high-Q design, the case can be cooled with a relatively small amount of additional cooling (0.55 Btu/sec). Variations of the maximum steel case temperature as a function of heat-pipe operating temperature are shown in Fig. 4; also shown is the effect of providing insulation outside the case.

Heat-Rejection Techniques

Figure 5 shows required radiator area vs operating temperature for the 30 Btu/sec heat load. A surface emissivity of 0.8 and an area factor of 1.0 have been assumed. If 3000°F is taken as the average temperature for the radiating surface in the high-Q case, 0.55 ft^2 (510 cm^2) is the required radiating area. For the low-Q design, where heat fluxes are $\frac{1}{10}$ and temperatures are about $\frac{1}{5}$ of those for the high-Q design, the radiator area is about 62 times greater.

An alternative method of heat rejection is to transfer the heat from the condenser section of the heat tube to a heat exchanger utilizing propellant from the propellant tanks. This is feasible if the propellant can be maintained below its boiling or decomposition temperature, and if the full cooling capability of the propellant is not used to cool the injector or film-cool the chamber. For the OF_2/B_2H_6 propellant system considered in this investigation, at an O/F ratio of 4.0 and a propellant temperature rise of 50% of the difference between the melting and boiling temperature, the heat that can be reasonably absorbed by the propellants is 3.1 Btu/sec for the B_2H_6 and 3.4 Btu/sec for the OF_2, giving a total of 6.5 Btu/sec. This is considerably below the 25 to 30 Btu/sec heat-rejection rate required for throat-area cooling. If decomposition of the propellants is not considered, then the use of the propellants for heat absorption is more feasible, particularly if all the propellant is used in the heat exchanger rather than only that flowing to the engine during firing. However, the chemical instability of B_2H_6, for example, at the elevated temperatures that could be attained locally or in the bulk liquid in such a heat exchanger, could present a serious problem.

Conclusions

The possibility of using heat pipes for cooling the throat area of a small rocket engine (141-lb thrust) was investigated analytically. For space operation, use of heat pipes for transfer and space radiators for rejection of the heat appears feasible. Some additional cooling must be provided, however, in order to maintain the structural characteristics of the motor case (if steel), particularly if firing times are long. Further demonstration of the technical feasibility of this concept must come from experiments and tests using actual high temperature heat sources and various heat-pipe configurations.

References

[1] Waters, E. D., "ATS-E Heat Pipe Development Test Report," DAC-60766, Aug. 1968, Donald W. Douglas Labs., Richland, Wash.

[2] "The GEOS-11 Heat Pipe System and Its Performance in Test and In Orbit," S2P-3-25, April 29, 1968, Applied Physics Lab., John Hopkins Univ., Silver Springs, Md.

[3] "Heat Pipe Technology for Advanced Rocket Thrust Chamber," proposed NASA-JPL Rept. 68-549, July 1968.

[4] Deverall, J. and Kemme, J., "High Conductance Devices Utilizing the Boiling of Lithium or Silver," LA-3211, Los Alamos Scientific Lab., April 1969, Los Alamos, N.Mex.

[5] Ranken, W. and Kemme, J., "Survey of Los Alamos and Euratom Heat Pipe Investigations," Los Alamos Scientific Lab., Los Alamos, N.Mex.

[6] Mayer, E., "Analysis of Convective Heat Transfer in Rocket Nozzles," *ARS Journal*, July 1961.

Relation of Meteoroid Protection to the Luminous Efficiency

JOHN W. VORREITER*

NASA Ames Research Center, Moffett Field, Calif.

THE design of shielding to protect spacecraft from meteoroids requires a knowledge of the frequency, mass, density, and velocity of the impacting bodies. Velocity, altitude, frequency, and luminous intensity may be obtained from meteor photographs. Assumptions as to the shape and knowledge of the luminous efficiency yield the mass and density of the meteoroid. Knowledge of this mass, density, frequency, and velocity permits the spacecraft designer to select the proper thickness of meteoroid bumper for a given probability of success. However, the luminous efficiency is somewhat uncertain. In this Note the sensitivity of meteoroid protective shield thickness upon the value used for luminous efficiency is derived and is shown to be small.

The luminosity equation[1] enables one to infer the initial mass as follows:

$$I = -(\tau/2)V^2(dm/dt) \tag{1}$$

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* Research Scientist. Member AIAA.

where I = meteor luminosity, τ = luminous efficiency (exact value uncertain), V = meteor velocity, and dm/dt = ablation rate or mass loss rate.

If τ is assumed to be independent of I and V, this equation can be integrated over the path of the meteor to obtain the initial mass, m_0. This integration also yields the relationship between m_0 and τ, namely that $m_0 \propto \tau^{-1}$.

The luminous efficiency is often considered to be proportional to velocity or, i.e., $\tau = \tau_0 V$, where τ_0 is called the luminous efficiency factor. If this relation is used instead of the more basic luminosity equation

$$\int_{m_0}^{0} dm = -\left(\frac{2}{\tau_0}\right)\int_0^\infty \left(\frac{I}{V^3}\right) dt = -m_0 \tag{2}$$

and $m_0 \propto \tau_0^{-1}$ instead of $m_0 \propto \tau^{-1}$.

If the meteoroid is "spherelike," one can calculate its density from the data obtained from photographic measurements. Observations of the deceleration, velocity, and altitude when extrapolated to the initial portion of the flight can yield the parameter $m_0/C_D A_0$, where C_D = drag coefficient and A_0 = initial frontal area. Assuming the body to be a sphere of uniform density enables one to express the frontal area in terms of mass and density, $A_0 = \pi[(m_0/\rho_0)3/4\pi]^{2/3}$, where ρ_0 = initial density. Substituting this A_0 and $m_0 \propto \tau^{-1}$ into the observed value of $m_0/C_D A_0$ and solving for density yields $\rho_0 \propto \tau^{1/2}$.

Several experimenters have investigated the relationship between the meteoroid shield thickness (or penetration depth) and V, m, and ρ of the impacting projectile. These relationships, along with $m_0 \propto \tau^{-1}$ and $\rho_0 \propto \tau^{1/2}$ enable the sensitivity of the required shield thickness or penetration depth to τ to be established.

Arenz[2] correlated penetrations on double-walled 2024-T3 aluminum targets with optimum shield thickness and spacing at a fixed velocity with the expression

$$(t_s + t_b)/d_{BL} = 0.287\rho^{0.6}[1 + 0.0067m^{(0.9+0.28\rho)}] \tag{3}$$

where t_s = shield or outer wall thickness; t_b = backup sheet thickness; d = projectile diameter; subscript BL = ballistic limit, value required to resist mechanical failure on inner surface; and m and ρ for the projectile are in g and g/cm^3.

The ratio $(t_s + t_b)/d$ is only slightly dependent upon the mass term, especially for the lower density material, and the meteoroid size range of interest. The expression thus can be simplified to

$$t_s + t_b = 0.287\rho^{0.6}d \tag{4}$$

and, for a spherical shape, $d = (6m/\pi\rho)^{1/3}$. Then, with $m_0 \propto \tau^{-1}$ and $\rho \propto \tau^{1/2}$ one can write

$$(t_s + t_b)_{BL} \propto \tau^{-1/5} \tag{5}$$

In another investigation Cour-Palais[3] developed an equation for the backup sheet thickness for glass particles impacting, multiwalled 2024-T3 aluminum structures with practical values of shell spacing and front-sheet thickness. His equation has the form

$$t_b = 0.055(\rho\rho_t)^{1/6}m^{1/3}V(\text{cm}) \tag{6}$$

where t_b = backup sheet thickness, cm; ρ and ρ_t are particle and backup sheet densities, g/cm^3; and V is in km/sec. A similar analysis to that for the previous penetration equation yields $t_b \propto \tau^{-1/4}$ for a given structure and velocity.

In a third study Summers[4] demonstrates a relation for the penetration of a projectile into a solid body

$$p/d = 2.28(\rho_m/\rho_t)^{2/3}(V/C)^{2/3}$$

where C = speed of sound in the body, and p = penetration depth. Here, a similar analysis yields

$$p \propto \tau^{-1/6} \tag{7}$$

In summary, then, the relationship presented by Arenz yields $t \propto \tau^{-1/5}$ for V = 7.2 km/sec, $0.7 \leq \rho \leq 2.78$ g/cm^3,