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Surface Cooling of Scramjet Engine Inlets Using Heat Pipe, Transpiration, and Film Cooling

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This article reports the results of applying a finite-difference-based computational technique to the problem of predicting the transient thermal behavior of a scramjet engine inlet exposed to a typical hypersonic flight aerodynamic surface heating environment, including type IV shock interference heating. The leading-edge cooling model utilized incorporates liquid metal heat pipe cooling with surface transpiration and film cooling. Results include transient structural temperature distributions, aerodynamic heat inputs, and surface coolant distributions. It seems that these cooling techniques may be used to hold maximum skin temperatures to near acceptable values during the severe aerodynamic and type IV shock interference heating effects expected on the leading edge of a hypersonic aerospace vehicle scramjet engine.

Nomenclature

a	=	speed of sound
HP	=	heat pipe
I	=	dimensionless transpiration coolant injection parameter
L	=	heat pipe length
M	=	mach number
Q	=	heat flux, W/m²
T	=	temperature, K
u	=	flow velocity, m/s
W_{cool}	=	film coolant injection parameter
x	=	chordwise direction
β	=	shock wave angle
γ	=	specific heat ratio
θ	=	flow direction angle

Subscripts

NOSE	=	aircraft nose
STAG	=	stagnation conditions
2	=	conditions after nose shock
3	=	conditions after engine inlet shock
∞	=	surroundings

Introduction

IT is generally accepted that the proposed aerospace plane will rely primarily on air-breathing propulsion provided by supersonic combustion ramjet engines (scramjets) for trans-atmospheric hypersonic flight.[1] An important design aspect that distinguishes the aerospace plane design from conventional subsonic or supersonic aircraft design is the integrated airframe-propulsion concept. For conventional aircraft, the components that provide lift (wings), thrust (engines), and volume (fuselage) are separate and distinct. They are easily identifiable on the aircraft and can generally be treated as separate aerodynamic bodies with moderate interaction when combined for a total aircraft system design/analysis.

In contrast, the proposed hypersonic aerospace plane involves a careful integration of the scramjet with the aerodynamic shape of the vehicle itself to enhance propulsion system performance (Fig. 1). This integrated airframe-propulsion concept utilizes the entire vehicle underbody as part of the scramjet. Initial compression of the air occurs as a result of passage through the vehicle nose bow shock. Additional compression and the combustion process take place inside a series of engine modules located near the rear of the vehicle. Primary expansion of the combustion gases occurs over the vehicle's nozzle shaped bottom rear surface. The scramjet engine modules are part of an annular inlet area that traverses the vehicle undersurface and is designed to capture enough air to provide the required thrust.

An inevitable result of the integrated airframe-propulsion concept, however, is the intersection of the vehicle's nose oblique shock with the engine's leading-edge bow shock. Edney had defined six types of shock interference patterns that this intersection creates.[2] These patterns can result in highly localized and intense surface heat transfer rates on the engine cowl leading edge.[3–6] All hypersonic vehicle leading-edge surfaces may experience intense stagnation point pressures and heating rates. For engine leading edges, these loads can be amplified by an order of magnitude when the leading-edge bow shock is impinged upon by an oblique shock wave and results in a type IV interference pattern.[4] Thus, shock interference heating is an additional problem encountered when

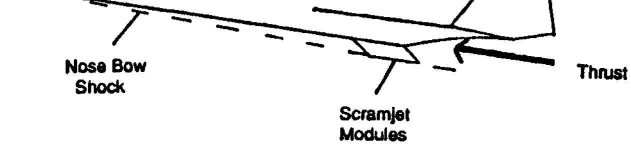

Fig. 1 Hypersonic aerospace plane, integrated airframe-propulsion concept.

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considering hypersonic vehicle leading-edge cooling, especially in the case of scramjet engine inlets.

Motivated by the results discussed by Modlin[7] on using liquid metal heat pipe, transpiration, and film cooling techniques to reduce hypersonic vehicle leading-edge surface temperatures, an investigation was conducted to study the feasibility of applying these same techniques to cool an aerodynamically heated scramjet engine inlet leading edge exposed to transient shock interference heating. The objective of this paper is to report the results of the investigation.

Model Development and Analysis

It has been demonstrated that the supersonic jet surface interaction (type IV shock interference pattern) yields the most severe heat transfer rates and, consequently, has been the focus of the majority of the research in this area.[6] Dechaumphai et al.[5] report on a study of the thermal-structural response of a 0.25-in. diam, internally cooled leading edge subjected to intense shock wave interference heating. The scenario analyzed represented the acceleration of a hypersonic vehicle through Mach 16. Initially, the assumed engine leading edge was exposed to a Mach 8 flow behind the Mach 16 shock. As the vehicle accelerated through Mach 16, the nose oblique shock swept across the engine leading-edge bow shock (at an assumed speed of 2 ips), intersected the bow shock, and produced transient Type IV shock interference heating. The envelope used in the study for peak heating values and distributions were idealizations of the experimental data given by Holden et al.[4] After the oblique shock swept by, the engine leading edge was assumed to be heated by the Mach 16 freestream flow.

In the present analysis, the shock interference conditions and data used in the Dechaumphai et al.[5] study were also applied. A 0.25-in. diam leading edge represented the scramjet engine inlet. Shock interference heating occurred on the surface as the vehicle accelerated through a Mach 16 freestream flow, exposing the engine leading edge to a Mach 8 flow (Fig. 2). For the assumed vehicle ascent flight trajectory of Fig. 3, this condition arose at approximately 900 s into the mission.

Colwell and Modlin[8] report on a study that was conducted to determine the feasibility of using liquid metal heat pipe cooling of hypersonic vehicle leading-edge surfaces supplemented by an internal active heat exchanger or surface mass transfer cooling. The leading-edge cooling model used was based upon a finite difference computational technique that accounted for 1) transient surface aerodynamic heating and surface cooling that varied with chordwise position; 2) radiation or convection heat exchange with the surroundings; 3) transient internal heat exchanger cooling; and 4) temperature-dependent thermal properties. In the study it was assumed that there was no shock/leading-edge/boundary-layer interaction nor shock/shock interference, the air in the vicinity of the leading edge was in local thermochemical equilibrium, and the leading-edge boundary layer was laminar. Surface mass transfer cooling was limited to gaseous coolant transpiration over the entire heat pipe cooled leading-edge surface, stagnation point coolant injection for film cooling and coolant temperatures of approximately 600 K. Results of the investigation included transient stagnation, chordwise and skin/capillary structure temperature distributions. Results also suggest that cooling a hypersonic aircraft leading edge using liquid metal heat pipes supplemented by internal heat exchangers, transpiration or film cooling was feasible.

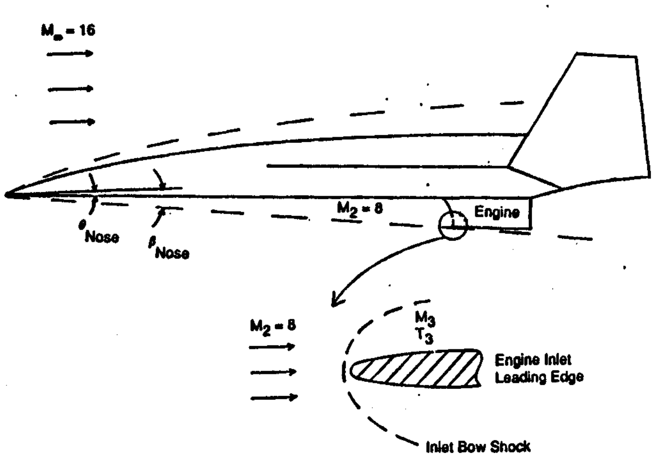

Fig. 2 Assumed hypersonic aerospace plane, configuration at mission time of 900 s.

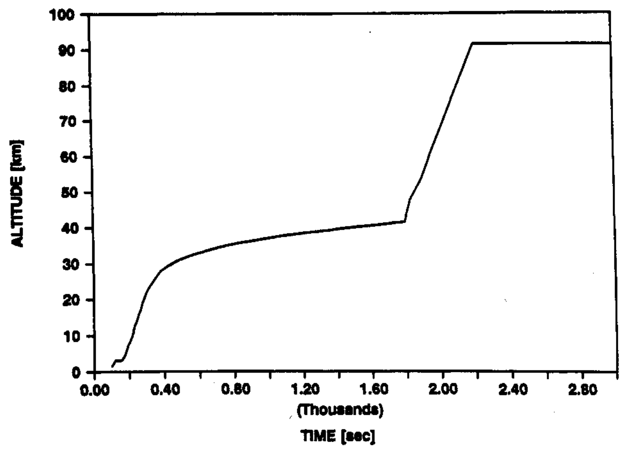

Fig. 3 Typical hypersonic aircraft flight trajectory.

To apply the hypersonic leading-edge cooling model of Colwell and Modlin[8] to the scenario developed for the present study, determination of the thermodynamic and flow properties of the air around the scramjet engine inlet leading edge, region 3 on Fig. 2 were necessary. To accomplish this, detailed information about the freestream flow conditions and those behind the nose bow shock, region 2, was first required. Because the region 2 air flow at this time was not greater than $M = 8$, standard compressible flow and oblique shock relations were used to calculate the required region 2 air data[9,10]:

$$\frac{T_2}{T_\infty} = \left(\frac{a_2}{a_\infty}\right)^2 = \left(\frac{M_\infty}{M_2}\right)^2 \left(\frac{u_2}{u_\infty}\right)^2 \tag{1}$$

$$\frac{u_2}{u_\infty} = \frac{(\gamma - 1)M_\infty^2 \sin^2\beta + 2}{(\gamma + 1)M_\infty^2 \sin^2\beta} \tag{2}$$

$$\frac{T_2}{T_\infty} = 1 + \frac{2(\gamma - 1)M_\infty^2 \sin^2\beta - 1}{(\gamma + 1)^2 M_\infty^2 \sin^2\beta} (\gamma M_\infty^2 \sin^2\beta + 1) \tag{3}$$

$$\theta \cong \frac{2\beta}{(\gamma + 1)} \tag{4}$$

Substituting Eq. (2) into Eq. (1) and equating this result with Eq. (3) allowed for the solution of β_{NOSE} at $M_\infty = 16$. Furthermore, because $M_\infty \beta \gg 1$, the hypersonic approximation for θ_{NOSE} was made using Eq. (4). Having a value for θ_{NOSE} allowed for the determination of the transient flowfield properties downstream of the nose shock using standard oblique shock relations for the mission time of 0–900 s. Beginning at 900 s, it was assumed that the oblique shock swept across the engine inlet leading-edge bow shock in less than 0.2 s, initiating the type IV shock interference behavior described by Dechaumphai et al.[5]

After approximately 900.2 s, the engine inlet leading edge was exposed to the transient freestream flight conditions for the remainder of the ascent mission. Flowfield conditions from 900 to 900.2 s were assumed to remain constant at 900 s values. The determination of condition 3 flow properties, used ultimately in the leading-edge cooling model, was accomplished

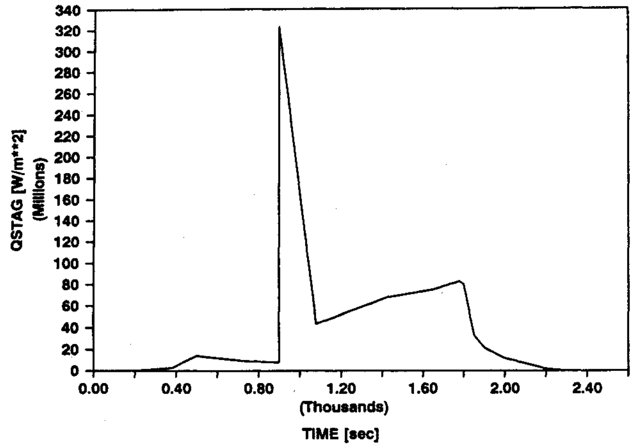

Fig. 4 Transient stagnation aerodynamic heat flux.

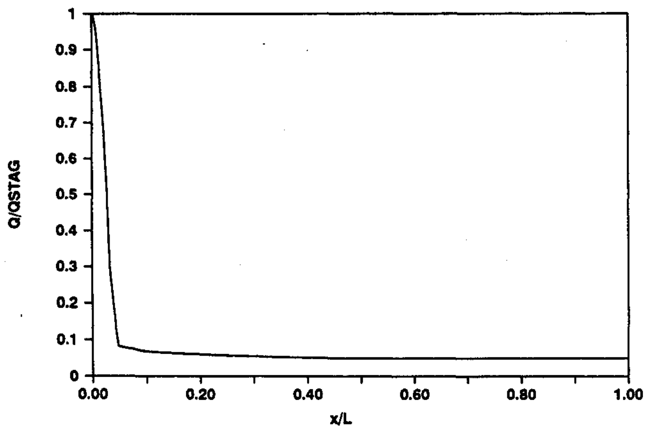

Fig. 5 Scramjet leading edge chordwise surface heat flux distribution.

using the same techniques and assumptions employed by Modlin[7] and described by Colwell and Modlin.[8]

Using engine inlet leading-edge flowfield property data, the transient leading-edge aerodynamic heat flux data were determined.[11] Figure 4 shows the transient engine inlet leading-edge stagnation heat flux data used in this study. Figure 5 shows the assumed leading-edge chordwise surface heat flux distribution. From 900 to 900.2 s, the shock interference data of Dechaumphai et al.[5] has been incorporated. However, as with the leading-edge analysis of Colwell and Modlin,[8] this analysis assumed the maximum heating occurred at the leading-edge stagnation point. This assumption was reasonable, except during the shock interference condition. Holden et al.[4] acknowledges that attempting to define a peak heating load for a type IV interaction is difficult, but report that the general trend is for peak loads to occur when supersonic jet impingment is approximately 20 deg below the stagnation point, regardless of Mach number. Nevertheless, it was felt that due to the very short time duration of the shock impingment effect, this shortcoming did not significantly affect the present results. The current analysis was also based upon the use of a 0.4029-m-long, lithium-filled heat pipe incorporated into the engine inlet leading edge. The heat pipe colombium alloy shell thickness was 0.0005 m and the molybdenum wick was 0.000762-m-thick. Figure 6 shows a schematic of a heat pipe cooled leading edge.

Results

Results of applying the leading-edge cooling model of Colwell and Modlin[8] to the scramjet inlet leading-edge case are shown in Figs. 7–10. Figures 7 and 8 show predicted transient surface stagnation temperatures. Figure 7 shows a comparison of the heat pipe only cooled case to that of the heat pipe with air transpiration cooling. Figure 8 similarly shows a comparison of the heat pipe only to the combined heat pipe/film cooling case. Figure 9 shows the leading edge stagnation point normal direction skin/capillary structure temperature at a mission time of 1800 s. Figure 10 shows the chordwise direction surface temperature gradient at 1800 s. Unlike the analysis of Colwell and Modlin,[8] no internal heat exchanger was considered in the present analysis.

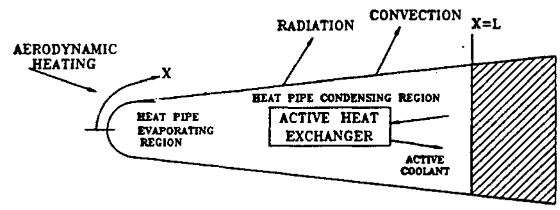

Fig. 6 Schematic of a heat pipe cooled hypersonic aircraft leading edge.

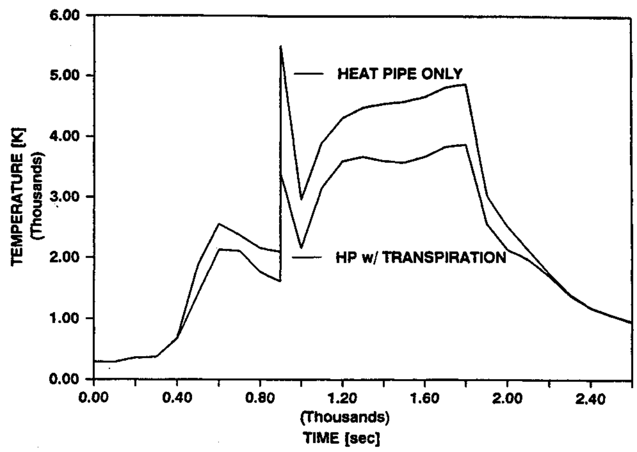

Fig. 7 Transient surface stagnation temperatures using transpiration cooling.

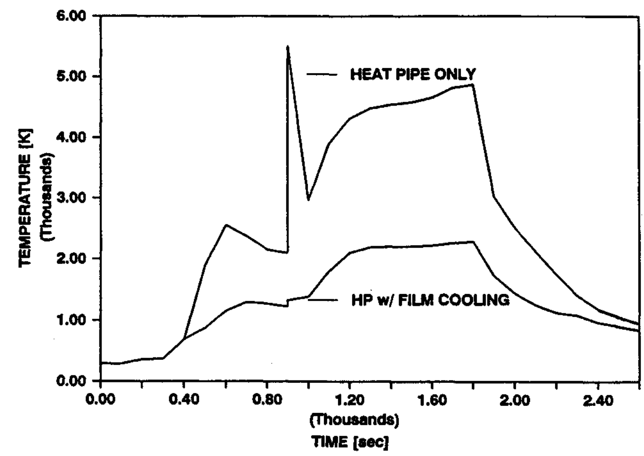

Fig. 8 Transient surface stagnation temperatures using film cooling.

The dimensionless transpiring air injection time dependency used in this study was maximized while still ensuring model applicability, as discussed by Colwell and Modlin,[8] and is shown in Fig. 11. Chordwise surface distance dependency was selected to match surface aerodynamic heat flux distance-dependency and transient air injection along the engine inlet stagnation line used for film cooling is shown in Fig. 12.

Applying a criteria of 1500–1800 K maximum surface temperature, as discussed by McComb et al.,[12] to computed results in the present study yields a number of observations. First, heat pipe cooling alone is not sufficient during a mission time interval of approximately 450–2250 s. The temperature spike shortly after 900 s on Figs. 7 and 8 corresponds to the shock interaction effect discussed previously and represents

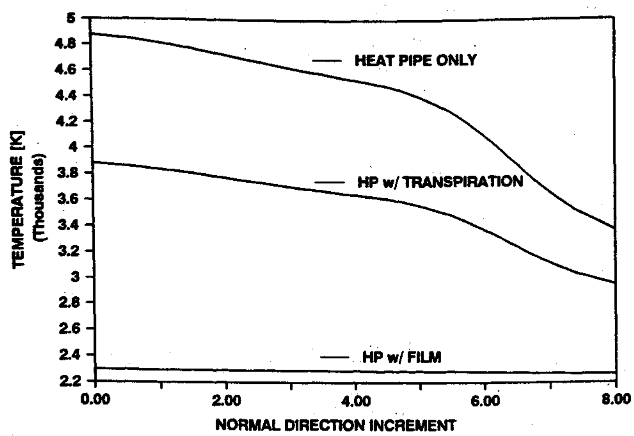

Fig. 9 Normal direction stagnation point temperature gradient at 1800 s.

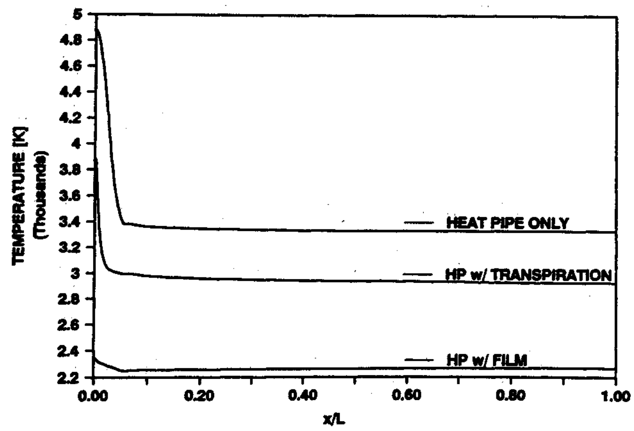

Fig. 10 Chordwise direction surface temperature gradient at 1800 s.

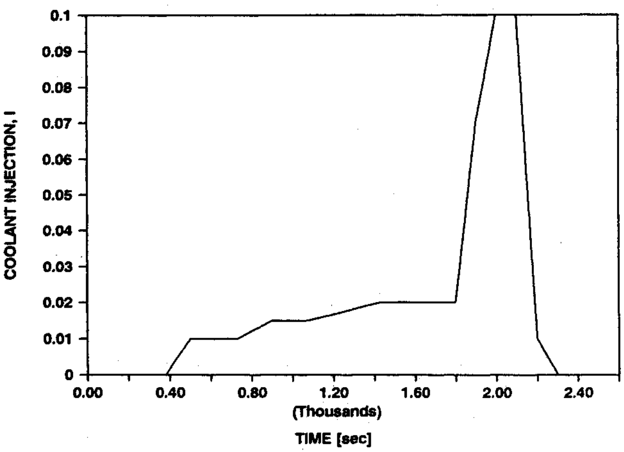

Fig. 11 Dimensionless transpiration coolant injection profile.

the maximum heat pipe only cooled surface temperature. Second, incorporating air transpiration cooling with the heat pipe does not sufficiently lower the maximum temperature (Fig. 7). However, the supplemental air transpiration surface cooling does seem to shift the maximum mission surface temperature away from the shock interaction time and to a later mission time (approximately 1800 s). This same shift is also indicated with the film cooling case (Fig. 8). Third, because the maximum leading-edge surface temperature resulting from heat pipe cooling alone is due to the shock interaction effect, the supplemental transpiration or film cooling seems to reduce the thermal effect of shock surface interaction and tends to shift the maximum surface temperature to a later mission time, well beyond the time when shock interaction heating occurs.

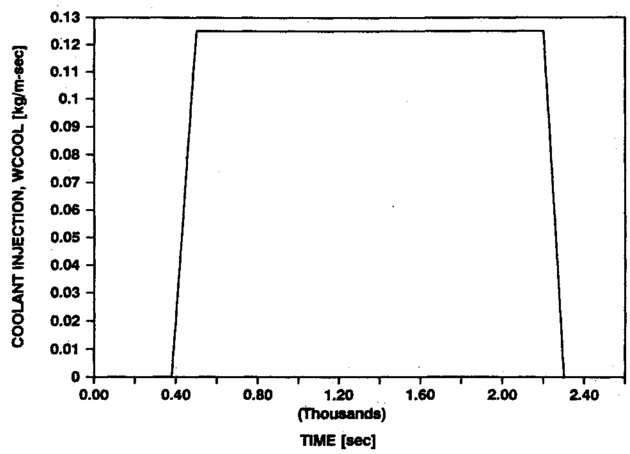

Fig. 12 Film coolant injection profile.

Thus, based upon the assumed conditions of this study, these results suggest that heat pipe cooling supplemented by surface transpiration or film cooling could be used to mitigate the otherwise severe type IV shock interaction surface heating effects. They indicate that this cooling technique might be an attractive alternative to those discussed by Holden et al.[4] and Glass et al.[13]

Additionally, making use of conclusions drawn by Modlin[7] and Colwell and Modlin[8] the use of some internal type of active heat exchange or the use of a surface coolant with a lower molecular weight and higher specific heat than air (such as helium) could improve the overall leading-edge cooling effectiveness. In the second case, potential effect on engine combustion efficiency, must be considered as a result of the presence of coolant gas in the combustion air. The use of hydrogen as a surface coolant was not considered due to its low combustion temperature (approximately 1100 K) relative to the high allowable maximum leading-edge surface temperatures.[14]

It must be emphasized that the analysis presented herein, as in the analysis reported by Colwell and Modlin,[8] assumed that the engine inlet leading edge had an attached laminar boundary layer throughout the entire mission. Although the transpiration and film coolant mass addition rates used in this study were selected using the best available criteria in the literature to ensure boundary-layer stability,[7] the severe adverse pressure gradient imposed on a laminar boundary layer by shock interaction effect could cause local separation. Unfortunately, there is a lack of information in the literature regarding this aspect of the shock interaction. A turbulent boundary layer has a greater capability of withstanding adverse pressure gradients than a laminar one and this ability increases with Mach number. Ledford and Stollery[15] report on a test program intended to evaluate the effect of shock interaction on a turbulent, film cooled, hypersonic, flat-plate boundary layer. Their results indicated that turbulent film cooling offered little thermal protection to a flat plate.

Concluding Remarks

This paper has reported the results of applying a finite difference based computational technique to the problem of predicting the transient thermal behavior of a hypersonic aerospace vehicle scramjet engine inlet leading edge exposed to a typical ascent flight aerodynamic heating condition, including type IV shock interference surface heating. It has been demonstrated that liquid metal heat pipe cooling supplemented with surface transpiration or film cooling may be used to reduce substantially the very high surface and structural temperature gradients that are caused by this type of heating. The mathematical models developed are very general in that they can accommodate different geometrics, vehicles, missions, and structural materials.

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