

MASS TRANSFER COOLING, A MEANS TO PROTECT HIGH SPEED AIRCRAFT

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IN HYPERSONIC flight, an aircraft is surrounded by a layer of hot air as shown in Fig. 1. This layer is generated either by adiabatic compression ahead of the vehicle or by internal friction along its sides. The temperature which the material of the skin can withstand is usually considerably lower than the temperature in this layer, and as a consequence, an intensive heat flux occurs into the skin of the aircraft. Part of this heat is removed from the surface by radiation. In many cases this removal constitutes, however, only a small fraction of the heat flux into the surface. The rest has to be absorbed by a material with adequate heat capacity—a heat sink—which has to be carried along in the aircraft because a utilization of the air outside the heated region as a coolant is in most cases too difficult or even impossible.

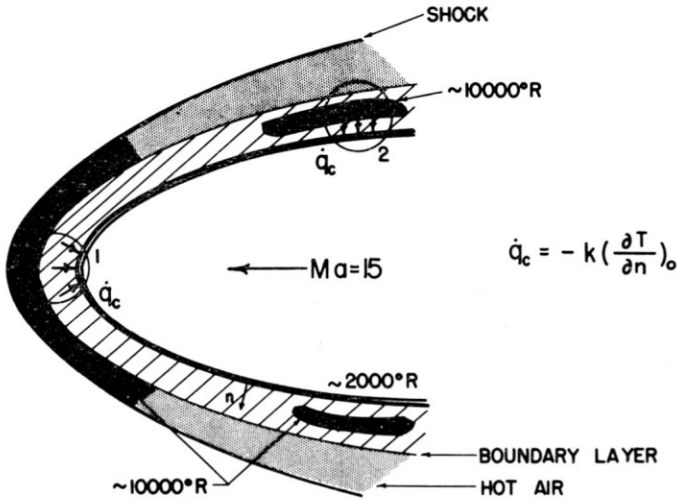


FIG. 1. Front part of a hypersonic vehicle with shock configuration and hot air regions.

The heat sink can be a solid material—possibly the skin itself—or a material which melts and evaporates or sublimates or chemically decomposes in the heat absorption process. A heat balance on the vehicle determines the required mass of the heat sink. Table 1 shows the equations expressing this energy balance for a vehicle re-entering from high altitude (h) through the atmosphere to the surface of the earth or for an aircraft flying with a

high velocity (V) in steady flight. The re-entering vehicle with a mass M has initially a kinetic energy $MV^2/2$ and a potential energy Mgh . Practically all of this energy is used up in the re-entry process and converted into heat. However, only a fraction of this heat enters the skin of the vehicle. This fraction is denoted by the coefficient α . The heat entering the skin is either radiated from the surface (Q_{rad}) or absorbed by the heat sink of mass M_s . The term in the square brackets gives the heat absorbed per unit mass of heat sink material where c denotes the specific heat, i_{sg} the heat absorbed in the sublimation process, T_i the initial temperature, and T_s the sublimation temperature. It is assumed that the vapor leaves the vehicle at the sublimation temperature. Similar expressions hold for a melting and evaporating or for a chemically decomposing heat sink. The coefficient β indicates that not all of the heat sink mass may be sublimed. The re-entry process is often so fast that the heat removal by radiation can be neglected for preliminary calculations. In this case the equation on the second line describes the energy balance. The ratio of the heat sink mass to the total mass of the vehicle can be calculated from this equation when the re-entry conditions, the heat sink material, and the coefficients α and β are known.

For steady flight, the heat transferred per unit surface area and time from the hot air layer into the skin of the aircraft is denoted by \dot{q}_c . The heat radiated per unit area and time is \dot{q}_{rad} . The difference between these values, multiplied by the flight duration time τ , gives again the heat which has to be absorbed by the mass m_s of heat sink material sublimed per unit surface area. The convective heat flux \dot{q}_c from the hot air surrounding the aircraft into the surface of the vehicle is described by the last equation in which k denotes the heat conductivity and $\partial T/\partial n$ is the temperature gradient in the air at the surface and in a direction normal to it.

TABLE 1
HEAT ABSORPTION BY SINK

ENTRY:

$$\underbrace{\alpha M \left(\frac{V^2}{2} + gh \right)}_{\text{AIR TO SURFACE}} = \underbrace{\beta M_s [c(T_s - T_i) + i_{\text{sg}}]}_{\text{HEAT STORED IN SINK}} + Q_{\text{RAD}}$$

FOR $Q_{\text{RAD}} \approx 0$:

$$\alpha \left(\frac{V^2}{2} + gh \right) = \beta \frac{M_s}{M} [c(T_s - T_i) + i_{\text{sg}}]$$

STEADY FLIGHT:

$$(\dot{q}_c - \dot{q}_{\text{RAD}}) \tau = \beta m_s [c(T_s - T_i) + i_{\text{sg}}]$$

$$\dot{q}_c = -k \left(\frac{\partial T}{\partial n} \right)_s$$

If the material of the heat sink is transformed to vapor during the heat absorption process, then it has to be discharged from the aircraft. It has been found that by doing this in a proper way, the parameter α or the heat flux \dot{q}_c in Table 1 can be reduced considerably. For this purpose, a mass flow away from the skin of the vehicle has to be generated, and this cooling procedure is, therefore, called *mass transfer cooling*. Figure 2

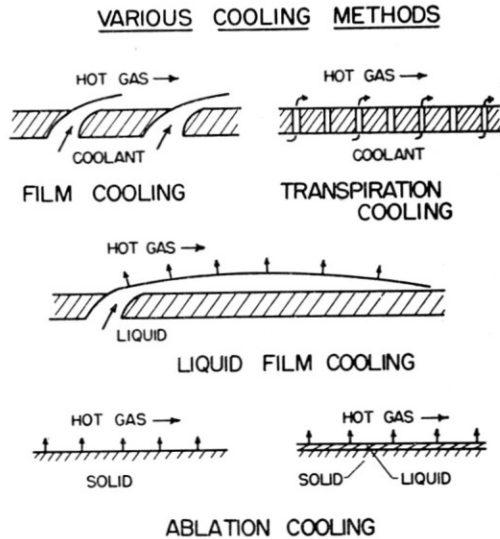


FIG. 2. Various mass transfer cooling methods.

sketches various ways in which this mass transfer cooling can be accomplished. In the film cooling process, the coolant gas or vapor is discharged through a sequence of slots in a downstream direction. More efficient is the transpiration cooling process in which the coolant gas is blown through the passages of a porous surface. In the liquid film cooling process, the coolant is discharged through slots as a liquid in such a way that it covers the outside surface of the skin as a continuous film. Evaporation into the hot gas stream takes place from the surface of this liquid film. The heat sink material can also be applied to the outside of the skin. Sublimation or decomposition starts as soon as the surface reaches a certain temperature. The material may also undergo at first a melting process, and evaporation occurs then at the liquid-gas interface. The last two processes are referred to as ablation cooling.

The heat flow from the hot air layer into the skin is reduced in such a mass transfer cooling process. In the following discussion it will at first be reviewed what reduction of the heat flow is expected from the results of analytical investigations and of wind tunnel experiments. Two limiting conditions will be considered, namely, plane stagnation point flow and flow over a flat plate with constant pressure along its surface. Locations where these conditions are approximated are circled in Fig. 1. It is

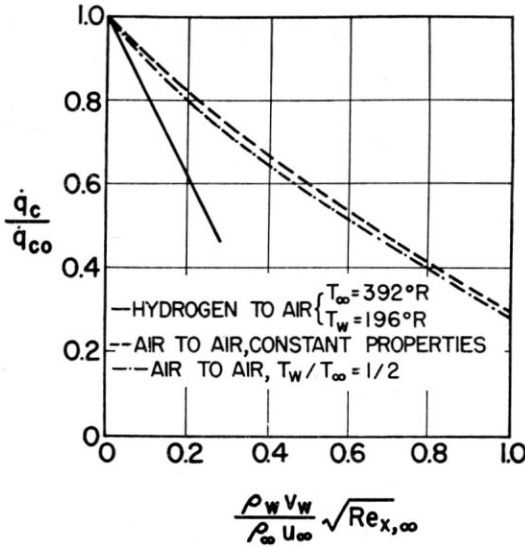


FIG. 3. Reduction of conductive heat flux from laminar boundary layer to surface in region near plane stagnation point by coolant mass release at the surface.

\dot{q}_c heat flow with coolant release

\dot{q}_{c0} heat flow for same flight condition and surface temperature but without mass release

ρ_w coolant density at wall surface

v_w coolant velocity at surface (normal to it)

T_w wall surface temperature

ρ_∞ air density outside boundary layer

u_∞ air velocity outside boundary layer

T_∞ air temperature outside boundary layer

$Re_{x, \infty} = \frac{\rho_\infty u_\infty x}{\mu_\infty}$ Reynolds number

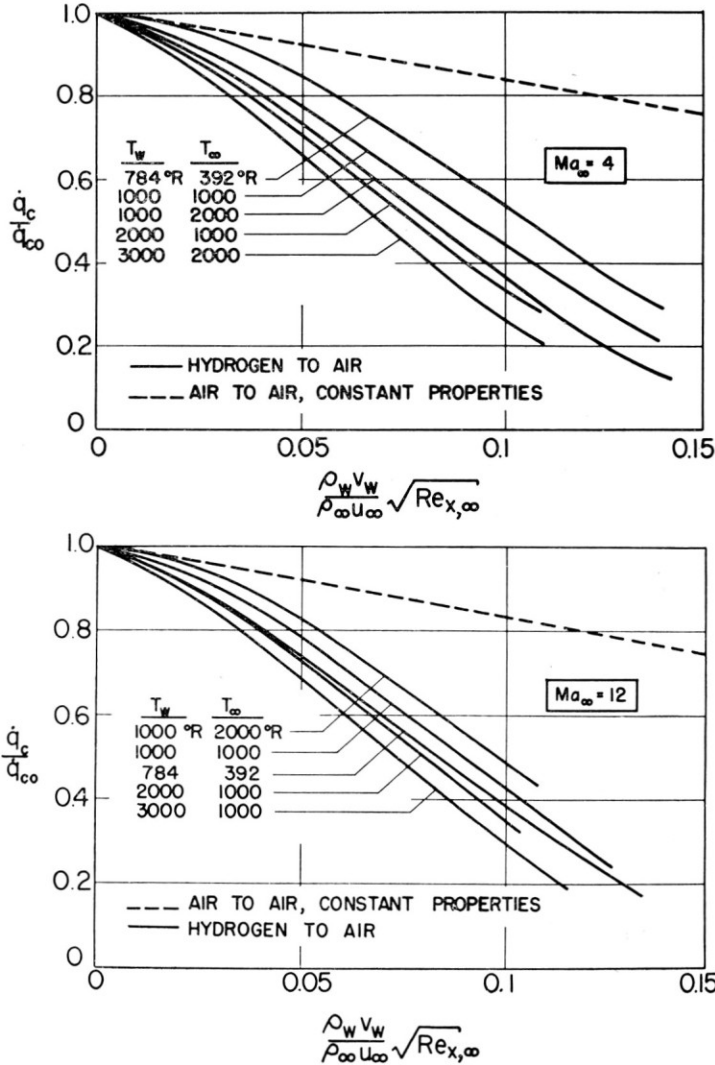
x distance from stagnation point

μ_∞ air viscosity outside boundary layer

expected that the flow will generally be laminar near a stagnation point, whereas it may be laminar or turbulent at location 2 in Fig. 1 which is approximated by a flat plate condition.

Figure 3 indicates the reduction in the convective heat flow \dot{q}_c which can be realized in the neighborhood of a plane stagnation point. The ordinate in this figure is the ratio of the actual heat flux \dot{q}_c into the surface to the heat flux \dot{q}_{c0} which is obtained under the same flow and temperature boundary conditions but without mass transfer. The abscissa gives the ratio of the mass velocity $\rho_w v_w$ with which the coolant leaves the skin surface to the mass velocity $\rho_\infty u_\infty$ outside of the boundary layer. This

ratio is additionally multiplied by the square root of the Reynolds number $Re_{x,\infty}$ based on the distance x of the location under consideration from the stagnation point, on the mass velocity $\rho_\infty u_\infty$ and on the viscosity of air at this location just outside the boundary layer. The dashed and dotted curve is for air or for a coolant gas with properties close to air⁽¹⁾. The dashed curve assumes the properties of the air and of the coolant to be constant and identical⁽²⁾. It holds, therefore, with good approximation



FIGS. 4 and 5. Reduction in conductive heat flow from laminar boundary layer to surface of flat plate by coolant mass release at surface.

Ma_∞ flight Mach number

Rest of nomenclature same as in Fig. 3.

for small temperature and concentration differences throughout the boundary layer. The full line is the result of a boundary layer analysis for hydrogen as coolant⁽³⁾. Figure 3 and the following Figs. 4 and 5 have been obtained by a solution of the system of boundary layer equations which is presented in the appendix to this paper. It can be observed in Fig. 3 that generally a considerable reduction in the heat flow q_c is connected with a mass release at the surface. The reduction is largest for hydrogen as coolant.

Figures 4 and 5 present the results of a corresponding analysis for a flat plate^(2,4). They hold for Mach numbers 4 and 12 respectively. The wall temperature T_w and the temperature T_∞ outside the boundary layer for the various curves are indicated in the figure. Calculations⁽⁵⁾ performed with helium as coolant indicate a reduction in the heat flux which lies between the curves for hydrogen and for air as coolant. It can again be observed that, even for small mass transfer rates, considerable reductions

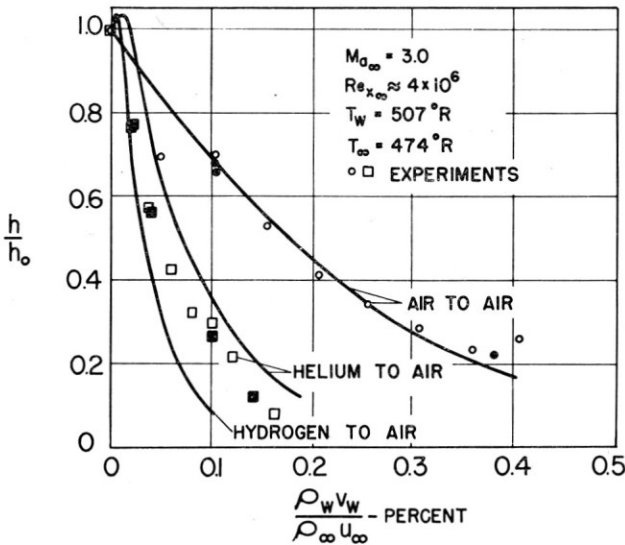


FIG. 6. Reduction in film heat transfer coefficient from turbulent boundary layer to surface of flat plate by coolant mass release at surface.

$$h = \frac{\dot{q}_c}{T_r - T_w} \text{ heat transfer coefficient with coolant release}$$

$$h_0 = \frac{\dot{q}_{c0}}{T_{r0} - T_w} \text{ heat transfer coefficient without coolant release for same flight condition and surface temperature}$$

T_r, T_{r0} recovery temperature (adiabatic wall temperature) with and without coolant release

Rest of nomenclature same as in preceding figures.

(Experiments were performed on a cone and full symbols distinguish runs on different days.)

in the convective heat flux can be obtained, especially with a light-weight gas as coolant.

The results of wind tunnel experiments⁽⁶⁾ on a flat plate with a turbulent boundary layer and with air and helium as coolants are indicated as circles and squares respectively in Fig. 6. In these experiments, the plate consisted of a porous material, and the cooling gas was blown through its passages. The ordinate in this figure indicates the ratio of the heat transfer coefficients with and without mass release at the surface. This ratio differs in most cases little from the ratio of heat fluxes which was used in the preceding figures. The lines present the results of an analysis by Rubesin⁽⁷⁾ which agrees very well with the experiments for air-to-air and fairly well with the experiments for helium-to-air injection.



FIG. 7. Copper cone exposed to an arc-heated gas jet. Maximum temperature in jet $15,000^{\circ}$ K, nozzle diameter 1.5 in.

(Courtesy Chicago Midway Laboratories, Chicago, U.S.A.)

In summary, it can be concluded from theory as well as from wind tunnel experiments that mass transfer cooling leads to a considerable reduction in the conductive heat flow to the skin of a vehicle, especially with a light-weight gas. In this way, the amount of heat which has to be absorbed by the heat sink, and therefore, the mass of this sink can be effectively reduced. Real conditions on a re-entering object differ by various factors from those assumed in the calculations, the results of which have been presented, as well as from the conditions under which the wind tunnel tests have been performed. The extremely high temperature in the air surrounding the object leads to dissociation and possibly to ionization of the air, and the coolant gas released from the skin may react chemically with the air. Various calculations have been performed and are reported in the literature⁽⁸⁾ which study these effects separately. The general conclusion can be drawn from their results that these effects do not change the heat flux into the surface essentially for conditions as indicated in Fig. 1. This is also true for the situation that a liquid film covers the surface⁽⁹⁾. The results shown in the preceding figures refer also to a very uniform mass flow of the coolant away from a smooth surface. In reality the coolant flow may not be so uniform, and the surface may be rough or wrinkled, especially when it is a liquid surface. The influence of these effects has to be checked by experiments, and the question arises as to what kind of tool can be used to simulate as far as possible the conditions existing on the high speed aircraft. Gas jets heated to temperatures of order 10,000° to 20,000°C by an electric arc discharge have been found useful in this connection. Figure 7 shows such a gas jet and a copper cone with blunt nose exposed to it. This photograph has been obtained in an experimental facility at Chicago Midway Laboratories. Experiments on ablation cooling performed with this jet will be shown in a film. A surprising feature observed in this film is the good performance of a nylon cone as compared with that of cones made of metals when they are exposed to the hot gas stream. This fact has to be interpreted as an indication of the effectiveness of the mass transfer cooling process occurring on the surface of the nylon cone.

APPENDIX

The following system of inter-related partial differential equations describes the velocity, concentration and temperature fields in a two-dimensional laminar boundary layer flow of a two-component gas mixture.

$$\begin{aligned}
 \text{continuity} \quad & \frac{\partial}{\partial x}(\rho u) + \frac{\partial}{\partial y}(\rho v) = 0 \\
 \text{momentum} \quad & \rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} = -\frac{\partial p}{\partial x} + \frac{\partial}{\partial y} \left(\mu \frac{\partial u}{\partial y} \right) \\
 \text{diffusion} \quad & \rho u \frac{\partial w}{\partial x} + \rho v \frac{\partial w}{\partial y} = \frac{\partial}{\partial y} \left(\rho D_{12} \frac{\partial w}{\partial y} \right) \\
 \text{energy} \quad & \rho c_p u \frac{\partial T}{\partial x} + \rho c_p v \frac{\partial T}{\partial y} = \frac{\partial}{\partial y} \left(k \frac{\partial T}{\partial y} \right) + \mu \left(\frac{\partial u}{\partial y} \right)^2 + u \frac{\partial p}{\partial x} + \\
 & + \rho D_{12} (c_{p1} - c_{p2}) \frac{\partial T}{\partial y} \frac{\partial w}{\partial y}
 \end{aligned}$$

The following symbols are used:

x, y	co-ordinate parallel and normal to wall surface
u, v	velocity component in x and y direction
p	pressure at x (independent of y)
w	mass fraction of coolant gas
T	temperature
$\rho, \mu, D_{12}, c_p, k$	density, viscosity, diffusion coefficient, specific heat at constant pressure, and heat conductivity of two-component mixture
c_{p1}, c_{p2}	specific heat at constant pressure of coolant gas and air.

In the above equations, thermal and pressure diffusion are neglected. Dissociation and chemical reactions are assumed absent. For the plane stagnation flow the second right-hand term in the energy equation describing viscous dissipation is assumed negligible. For flat plate flow the pressure gradient in the momentum equation is zero. The properties in all equations are assumed as functions of temperature and mass fraction. Best available information on this dependence was used in obtaining solutions.

The boundary conditions are

$$\text{at wall surface } (y = 0): u = 0, w = w_w, T = T_w$$

$$\text{at the outer edge of boundary layer } (y = \infty): u = u_\infty, w = 0, T = T_\infty.$$

The flow of air through the wall surface is postulated to be zero and the

mass fraction as well as the temperature to be uniform along the wall surface. This condition leads to a similarity within the boundary layer analogous to the one utilized by Blasius and permits a transformation of the partial differential equations into total ones. The resulting system of interrelated total differential equations was solved by iteration on a Remington Rand 1103 electronic computer.

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DISCUSSION

N. P. RUDEN*: Ich habe nicht deutlich erkennen können, ob die Injektionskühlung bei laminarer Grenzschicht besser (oder schlechter) ist als bei turbulenter Grenzschicht. Meine erste Frage ist daher folgende: Besteht im Hinblick auf den Wirkungsgrad der Injektionskühlung ein Unterschied zwischen laminarer und turbulenter Grenzschichtströmung und welche von beiden liefert das bessere Resultat?

Wenn es aber einen Unterschied dieser Art gibt, dann stellt sich sofort auch die Frage nach der Lage des Umschlagpunkts. Man kann wohl vermuten, dass die Instabilität der laminaren Grenzschicht durch die Injektion vergrößert wird. Für den Fall, dass die Lage des Umschlagpunkts mit der Stärke der Injektion merklich variiert, müsste die Bilanz der Injektionskühlung auch diesem Umstand Rechnung tragen. Ich wäre Ihnen dankbar, wenn Sie hierzu etwas sagen könnten.

E. R. G. ECKERT: The reduction in any heat flux which can be obtained by mass transfer cooling is significantly larger for a laminar than for a turbulent boundary layer when the comparison is made for a fixed ratio of the coolant mass velocity to the mass velocity in the stream outside the boundary layer.

Injection of a gas from the wall into a boundary layer has the effect of increasing the boundary layer thickness and of changing the shape of the velocity profile. Both factors tend to make the boundary layer with mass transfer cooling less stable. Experiments at the Rosemount Aeronautical Laboratories of the University of Minnesota have been conducted by Bernard M. Leadon on a cone in supersonic flow of Mach numbers 3 and 5. They indicate that the point of transition from laminar to turbulent flow moves with increasing mass injection into the boundary layer in an upstream direction. However, it was found that the amount of this shift is comparatively small, so that the transition Reynold's number does by no means decrease by an order of magnitude.

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