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FUNDAMENTAL RELATIONSHIPS FOR ABLATION AND HYPERTHERMAL HEAT TRANSFER

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FOREWORD

This report was prepared by the Structures Division of the Air Force Flight Dynamics Laboratory, Research and Technology Division, Air Force Systems Command at Wright-Patterson Air Force Base, Ohio. The work was conducted under Project No. 1368 "Structural Design Concepts", Task No. 136804 "Re-Entry and Hyperthermantic Structures". The manuscript of this report was released by the author November 1965 for publication as an RTD Technical Report.

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ABSTRACT

A qualitative review of fundamental relationships involved in ablation engineering and aerodynamic heating is presented. This document is written for rapid reading with the primary purpose of introducing the technical area to engineers, scientists, and management personnel associated with but not intimately working in hypervelocity thermal protection.

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ILLUSTRATIONS

SYMBOLS*

C_p	Specific heat of gas (Btu/lb°F) (joules/gm °C)
E	Non-sensible or latent energy required or released during aerodynamic gas thermal processes (Btu/lb) (joules/gm)
f	Fraction of pyrolysed material ejected as a gas (Dimensionless)
g	Gravitational acceleration constant (32.2 ft/sec^2) (N.A.)
H	Gas enthalpy (Btu/lb) (joules/gm)
h	Film coefficient for convective heat transfer based upon temperature difference (Btu/ft^2 - °F - sec) (watts/cm^2·°C)
J	Mechanical equivalent of heat (778 ft lb/Btu) (N.A.)
$\overline{KE}$	Kinetic energy of gas (Btu/lb) (joules/gm)
$\dot{M}$	Mass rate ejected into the boundary layer as a gas per unit surface area (lb/ft^2 sec) (gm/cm^2 sec)
$\dot{m}$	Pyrolysis rate per unit surface area (lb/ft^2 sec) (gm/cm^2 sec)
mol air	Molecular weight of air (Dimensionless)
mol gas	Molecular weight of gas injected into boundary layer (Dimensionless)
N	Empirical constant (Dimensionless)
q	General symbol for heat energy rate expressed per unit area of vehicle surface (Btu/ft^2-sec) (watts/cm^2)
$\dot{q}$	Aerodynamic convective heat flux to a unit area of vehicle surface (Btu/ft^2-sec) (watts/cm^2). This symbol falls within the definition of q (above) but is limited to a specific type of heat transfer that arrives at the vehicle exterior surface.
$\overset{*}{q}$	Effective heat of ablation per weight loss of ablator (Btu/lb) (joules/gm)
q_{eff}	Effective heat capacity per original ablator system weight (Btu/lb) (joules/gm)
R	Stagnation radius (ft) (cm)
r	Recovery factor (Dimensionless)

*Dimensions are given for both English and International unit systems. Where 'N.A.' is noted, the quantity would not be utilized in equations using International units.

SYMBOLS (Continued)

S	Surface distance (ft) (cm)
T	Temperature (^oR) (^oK)
V	Relative velocity of air (ft/sec) (cm/sec)
x	Distance dimension in air stream at vehicle wall (ft) (cm)
α	Empirical exponent (Dimensionless) or angle of attack (deg)
β	Transpiration factor (Dimensionless)
ϵ	Total hemispherical emittance (Dimensionless)
ρ	Density (lb/ft^3) (gm/cm^3)
σ	Stefan-Boltzmann constant (Btu/ft^2-sec-$^oR^4$) (watts/cm^2-$^oK^4$)

Subscripts

abs	Absorbed
aw	Adiabatic wall or recovery
BL	Blocked
BLOW	Blowing
c	Velocity required for a circular orbit at a specific altitude
COMB	Combustion
cw	Cold wall
e	Boundary layer edge
gc	Gas cap radiation
o	Sea level
ref	Reference
RET	Radiation equilibrium temperature
r	Radiated (generally away from a hot surface)
s	Air stagnation region or vehicle stagnation point
tc	Cool transpiring fluid entering permeable matrix

SYMBOLS (Continued)

tw	Hot transpiring gas at vehicle wall (exterior of permeable matrix)
w	Wall
∞	Free stream
1 and 2	Different situations

SECTION I

INTRODUCTION

In proportion to the complexity and diversity of aerospace technology, the task confronting research and systems engineering management is rapidly expanding. The extent of this task is emphasized where the manager with a particular background must direct a balanced program over a wide technical spectrum. Under this situation, and with time being characteristically limited, the program manager may seek council from specialists. However, the "expert" and possibly conflicting advice so obtained requires proper tempering or purification at the hands of the project manager. The success of this tempering process rests upon sound management judgement backed by adequate bases in each concerned technical area. Such bases should include an understanding of general governing equations, properties and qualitative process descriptions, inter-relationships between associated or alternative technical areas, general attributes and limitations of each technical area, and the overall state of technology. Some of these will vary with time and require continual review, other bases tend to be unchanging.

Hypervelocity vehicle thermal protection and, specifically, the ablation aspect thereof, form an area that is often foreign to the engineer or scientist working with "conventional" materials and structural configurations. The intent of this document is to qualitatively review the fundamental thermal, aerodynamic, and material relationships involved in ablator technology in order to readily provide a basis for its understanding. This presentation is primarily intended for use by supervisory, staff, and other personnel associated with but not intimately working in hyperthermal technology. However, it also introduces this broad and interesting area to junior engineers and scientists.

A great number of reports, papers, and memoranda have been generated on ablation and many are available for general reference. These vary from fundamental to highly complex treatments of the various aspects of mass transfer under hyperthermal conditions. Many contain detailed materials data without theoretical introduction, and many are directed toward readers who are highly familiar with the subject. Some concentrate upon limited facets of the overall ablation field. Some have become effectively outdated. In all, an ever increasing wealth of information is available for the detailed study and design application of ablation systems. Similarly, the technology of aerodynamic heat transfer is broad and growing. However, personnel first entering into or becoming associated with hypervelocity heating and thermal protection are often confronted with a seemingly monumental task of learning the fundamentals. This is especially true for technical people with non-aerodynamic, non-heat transfer backgrounds. Thus, the need for a very basic but well rounded summary of this broad technical area exists, even in light of the overall availability of information.

This document is compiled to cover aerodynamic heating, boundary layer mass transfer, general thermal protection, and the materials and structural aspects of ablation. Presenting no new information, per se, it is organized to concisely convey a large amount of information in as basic a manner as possible. Basic and familiar thermodynamic processes are detailed to aid in communication. Most formulations and qualitative statements used are fundamental and bear no particular relationship to specific sources. References are, therefore, not generally presented in the text. Quantitative data on material performance or environment are not presented. Thus, the document is not a source for design information but rather is an educational tool.

In writing this report, the author has utilized information obtained during recent years from many written sources as well as through personal contacts. Therefore, the report is a compilation of experience and not a systematic literature search. The author does not wish to take credit for the development of the technology described herein but rather is indebted to the general technical community.

For those readers wishing further general information or insight on the state of ablation technology, the following sources are recommended as concise, easily read documents. Reference 1 is a bibliography on ablation. Although published in 1964, it summarizes many reports and papers of current interest. References 2 and 3 are listed as sources for general information on material performance, thermal analysis, and test methods. Reference 4 details a thermal analysis in depth and is recommended as a ready source showing the overall structure of the typical ablator analysis. References 5 and 6 are recent papers presenting the current state of materials development. The latter paper concentrates on, and well describes, the currently popular elastomeric ablator class.

SECTION II

AERODYNAMIC HEATING

One of the most severe problems confronting aerospace system design is the control and dissipation of aerodynamic or engine induced heating to vehicle exteriors. Within the hyperthermal spectrum, vehicle configurations and missions of interest are numerous and diverse. These include high acceleration ground to air missile nose caps and control fins, ram inlet ducts, booster exhaust shields, and re-entry vehicle heat shields. They involve varying severities of heating, dynamic pressure, acceleration, shock, and wind shear. All of these are major parameters for the selection of thermal protection systems (TPS). In this document, the overall problem will be represented by re-entry heat shields and associated aerodynamic heating, boundary layer phenomena, and thermal protection requirements.

A vehicle in motion through a fluid will experience stagnation heating, downstream heating, heating due to super and hypersonic shock impingement, and heating by air leakage at structural gaps. With the partial exception of hot gas leakage into or through vehicle structures, most external aerodynamic heat inputs to a vehicle are related or can be considered similar in nature to the heating that occurs at the nose and other vehicle stagnation areas. Stagnation heating for the re-entry vehicle is produced by conversion of the kinetic energy of the moving vehicle to thermal energy in the enveloping gas as the body is decelerated; or as the moving gas is decelerated, as in a wind or hot gas tunnel. This heating is a function of fluid density, body velocity, temperature or enthalpy differences between wall and boundary layer gases, and size or sharpness of the moving body frontal region. Stagnation may be visualized as adiabatically bringing a moving gas to a complete stop at a surface perpendicular to the flow.

This process is described:

$$H_\infty + \overline{KE}_\infty = H_s + \overline{KE}_s = H_s \qquad (1)$$

for an inert (no ablation or other mass transfer) vehicle surface. Relative to kinetic energy, H_∞ for hyperthermal flight is very small; therefore, H_s can be expressed solely as a function of vehicle velocity:

$$H_s \cong \frac{V_\infty^2}{2gJ} \qquad (2)^*$$

The air stagnation temperature, T_s, associated with the increase in enthalpy depends upon the changes in gas composition, including the degree of dissociation and ionization and the state of gaseous equilibrium. The rise in internal energy of the air upon compression will produce sensible temperature rises until (for hypervelocity conditions) the temperatures of dissociation and ionization are reached for the various air constituents. These gaseous changes are latent in nature. In addition, very rapid deceleration of the air (relative to the vehicle) from free stream to zero velocity may instantaneously raise the gas temperatures above those required for molecular and atomic breakdown, thereby producing non-equilibrium states. These states rapidly transform to equilibrium states with the chemical energy required for

*Equations with this designation have constants applicable only to the English system of units.

dissociation and ionization coming from reductions in the gas thermal energy and, inherently, the gas temperatures. The period to reach equilibrium is termed "relaxation time". For decreasing flight severity and as the vehicle bluntness is increased, equilibrium stagnation conditions will tend to prevail. Conditions on a relatively sharp (high lift to drag) vehicle during re-entry might typically be non-equilibrium at the nose, becoming equilibrium on downstream surfaces where the enveloping gases have had time to "relax".

The basic relation for determining stagnation or "total" temperature is:

$$H_s - H_\infty = \sum E + \int_{T_\infty}^{T_s} C_p \, dT \cong \frac{V_\infty^2}{2 g J} \qquad (3)^*$$

with the air specific heat varying with pressure and temperature as well as the air constitutents and their chemical states. For continuum flow, under stagnation pressures typical of the high heating periods of re-entry, a theoretical T_s might not be reached due to the cooling effects of the vehicle surface (Figure 1, View A). However, the bulk stagnation gases are at T_s (and also H_s).

Heat transfer from the hot stagnation gases to the vehicle is proportional to the gas thermal gradient $\Delta T / \Delta x$ at the vehicle wall. This gradient actually is non-linear and quite complex. Its form or very existence depends upon the gas flow condition (laminar or turbulent) and the gas mean free path as usually expressed in flow type; for example, continuum, slip, or free molecule flow (in order of increasing mean free path). For ~~heat transfer analysis~~ the gradient form is usually ignored and the standard convection formula is used to calculate heat transfer:

$$\dot{q}_s = h \, (T_s - T_w) \cong h \left(\frac{H_s - H_w}{C_p} \right) \qquad (4)$$

where h is an overall convection coefficient, numerically different for laminar and turbulent flow, and air specific heat is assumed independent of temperature. Thus, the thermal driving potential is formed by the theoretical ideal temperature, T_s, of the bulk stagnation gases and by the gas temperature immediately adjacent to the vehicle wall, which is taken as being equal to the wall temperature, T_w. The numerous complex and often unknown heat transfer relationships and quantities, such as air constituent diffusion rates, chemical states and reactions, catalytic surface properties, and local temperatures and pressures, are thus embodied in the heat transfer coefficient, h, which would be empirically determined.

However, $\dot{q}_s$ is generally determined using enthalpy differences in empirically derived formulations such as that of Detra, Kemp, and Riddell (Reference 7):

$$\dot{q}_s = 17600 \left(\frac{\rho_\infty}{\rho_o R} \right)^{(.5)} \left(\frac{V_\infty}{V_c} \right)^{(3.15)} \left(\frac{H_s - H_w}{H_s - H_{cw}} \right) \qquad (5)^*$$

This direct use of energy terms without determining temperature gradients is very convenient, especially for hot gas or plasma tunnels where fluid enthalpy is equal to easily determined input power minus facility losses.

In addition to being directly proportional to the square root of the local air density, approximately proportional to the cube of the velocity, and inversely proportional to the square root of the frontal hemispherical radius (or an effective radius for non-hemispherical shapes) Equations 4 and 5 show that $\dot{q}_s$ is directly proportional to the difference between the air bulk stagnation energy, H_s, and that at the wall, H_w. Knowing $\dot{q}_s$, for a specific vehicle configuration at a specific velocity, altitude, and wall temperature, $T_{w,1}$, one can easily determine the heat flux, $\dot{q}_{s,2}$, for some other wall temperature, $T_{w,2}$:

$$\dot{q}_{s,2} = \dot{q}_{s,1} \left(\frac{H_s - H_{w,2}}{H_s - H_{w,1}} \right) \tag{6}$$

Equation 5 has set $H_{w,1}$ at a "cold" condition; that is, about 500°R or room temperature. The heat rate associated with H_{cw} is termed a "cold wall" heat flux, $\dot{q}_{cw}$.

Noting that H_s is independent of the wall temperature and that T_w is highly sensitive to aerodynamic, radiative, and internal heat transfer; the use of a "cold wall" heating rate allows one to describe or rate the magnitude of aerodynamic heating to a specific vehicle at a given flight condition without knowing T_w. This "cold wall" rate is artificial except where T_w is actually cold; for example, a water cooled calorimeter.

Since the actual or "hot wall" heating rate is a function of T_w or H_w (Equations 4 and 5), and T_w depends upon both the aerodynamic heating and internal vehicle heat transfer, an iterative procedure balancing aerodynamic heating with internal heating and storage is used to determine the overall heat transfer rate.

Two additional facts on the "cold wall" condition are helpful to a fundamental understanding of its use. First, T_{cw} is generally the temperature of the calorimeters used in hot gas test facilities. It may vary at each test period (approximately room temperature for water cooled calorimeters) and should be specified for a complete or exact statement of $\dot{q}_{cw}$. For re-entry conditions T_{cw} is usually very much smaller than T_s and:

$$\dot{q}_{cw} \cong h\, T_s \tag{7}$$

Therefore, $\dot{q}_{cw}$ is a useful quantity without referring to a specific T_{cw}. Equation 6 is often approximated as:

$$\dot{q}_{s,2} = \dot{q}_{cw} \left(\frac{H_s - H_{w,2}}{H_s} \right) \tag{8}$$

The above relationships have been discussed with respect to a stagnation point; that is, the forward most point of three dimensional body. Clearly, Equation 5

becomes invalid for large, flat stagnation points or areas where R becomes infinite. Similarly, the range of R will have a minimum value, and ranges will exist for V_∞ . Heat transfer formulas for a stagnation point (Equation 5) can be modified to cover stagnation lines (as on a hemicylindrical leading edge) by changing (decreasing) the numerical constant and including geometric factors for sweep angle.

For a re-entry vehicle frontal surface, such as the hemispherical-like nose of Figure 1, the stagnation region, per se, is but a small portion of the overall frontal region. To be strictly accurate, it would be the infinitesimal region where the vehicle wall is perpendicular to the vehicle or free stream velocity. Moving circumferentially away from this region, an air velocity basically parallel to the vehicle wall will form, and with it an associated boundary layer. This relative air velocity increases with distance from the stagnation region (in typical airfoil fashion) ultimately surpassing Mach 1 for supersonic or hypersonic flow.

As shown in Figure 2, aerodynamic convective heating becomes less with peripheral distance from the stagnation point, assuming no change from laminar to turbulent flow. However, the heat transfer over much of the spherical surface can be approximated as that occuring at the stagnation point for many basic studies. This is especially true of small radii blunted frontal surfaces where the studies are used to calculate total thermal inputs. Such studies will produce conservative results since more heat flux will be calculated than actually occurs. For the preliminary design of large (for example, greater than about four inch radius) spherical components where actual local heating should be considered, convective heat flux to a point at angle θ from the stagnation point, and in the plane which includes the relative wind vector, can be approximated as $\dot{q}_s \cos \theta$. Figure 2 shows this approximation to be conservative. However, it is only applicable within ninety degrees of the stagnation point and is further limited to fluxes to the circular profile. The latter limit is shown by the angle of attack (α) lines of Figure 2. A representative temperature profile for the total area surrounding the stagnation point on a spherical surface is shown in Figure 3.

Surface heating to cylindrical components, such as leading edges, where the cylinder axis is perpendicular to the relative wind vector, varies along the circular dimension in a manner similar to that shown in Figure 2.

Aft of the stagnation regions, the bulk boundary layer air at a given surface position is moving (relative to the vehicle) at velocity V_e at the boundary layer outer edge. Neglecting interactions or shock impingements from the nose or other leading regions, the basic thermodynamic relation can be expressed as:

$$H_e = H_\infty + \overline{KE}_\infty - \overline{KE}_e = H_s - \frac{V_e^2}{2gJ} \qquad (9)^*$$

The temperature associated with H_e is determined by the same method type used to compute the stagnation temperature (Equation 3) or simply:

$$H_e - H_\infty = \int_{T_\infty}^{T_e} C_p \, dT \qquad (10)$$

for ideal flow with no dissociation. However, within the boundary layer viscous effects slow the gases to zero velocity at the vehicle surface and thereby increase the temperature of the gas as it approaches the wall (Figure 4). If the vehicle

were completely insulated from the boundary layer, that is adiabatic, the maximum boundary layer gas temperature would occur adjacent to the wall. This "adiabatic wall" or "recovery" temperature is denoted T_{aw}. The magnitude of $(T_{aw} - T_e)$ primarily depends upon the ratio of the molecular diffusivities of momentum to heat as expressed by the local Prandtl number. It is, therefore, a rather complex function of thermal and flow conditions.

Although the idealized vehicle wall may be considered adiabatic, the overall ideal thermal process at the wall is not adiabatic. This is due to heat transfer in the boundary layer away from the adiabatic wall by the forcing action of the idealized gas thermal gradient between T_e and T_{aw} (Figure 4).

The gas immediately adjacent to vehicle surfaces lying parallel to the air flow is relatively stagnant, but the thermodynamic process associated with these regions differs from that occurring at a true stagnation (frontal) point, where relative deceleration is not due to viscous drag and the whole region is considered adiabatic. Thus, comparing the adiabatic wall non-stagnation area condition with the stagnation point: T_{aw} is less than T_s. This is described by the following relation:

$$r = \frac{T_{aw} - T_e}{T_s - T_e} \leq 1 \tag{11}$$

T_{aw} and the recovery factor, r, are functions of numerous variables; for example, body location and geometry, Reynolds Number, and local attitude. They generally are empirically determined. At a stagnation point or stagnation line of zero sweep r equals unity since T_{aw} is T_s. Associated with T_{aw} is the adiabatic wall enthalpy, H_{aw}. For ideal gas approximations, H_{aw} is a direct function of T_{aw}, the air specific heat, and other energy terms (Equation 3).

This discussion on an adiabatic wall has portrayed the thermal condition of the air adjacent to vehicle surfaces which do not modify the boundary layer either by mass transfer or by thermal means. Considering the latter, the temperature of the vehicle wall, T_w, will affect the immediately adjacent air. This is described in Figures 1 and 4 where a negative gradient $\Delta T/\Delta x$ is formed at the vehicle wall, the shape of which is a particular function of the aerodynamic condition.

Heat transfer in the non-stagnation situation is expressed in terms of an effective thermal gradient based upon an empirical coefficient:

$$\dot{q}_w = h\,(T_{aw} - T_w) \tag{12}$$

or by equivalent expressions using enthalpy differences.

As previously discussed, aerodynamic heat is primarily transferred by conduction/convection between the hot boundary layer gases and the cooler body surface. However, in extreme cases of very high velocity and/or low altitude, the stagnation boundary layer gases become hot to the extent that their radiant energy becomes appreciable. This heat, termed gas cap radiation (q_{gc}), may typically surpass 10 percent of the convective aerodynamic heat rate for short periods during a superorbital entry. It is generally insignificant in lifting body manned re-entry from earth orbit. It should be noted that radiant energy produced from non-equilibrium gas states may

surpass an order of magnitude greater than that at the equilibrium state. These relations and their practical effects are complex and often unknown due to a lack of flight data.

An additional temperature term, reference temperature, will be briefly discussed. This temperature is a weighted mean of three distinct boundary layer temperatures and is used to compute constants for use in aerodynamic heat transfer relations. Thus, it is a practical temperature for use in determining gas properties in a boundary layer where complex thermal gradients exist. Its derivation and effectiveness are based upon and verified by experiment. The relation derived by Eckert (Reference 8) is:

$$T_{ref} = 0.22\ T_{aw} + 0.28\ T_e + 0.5\ T_w \tag{13}$$

or

$$T_{ref} = T_e + 0.5\ (T_w - T_e) + 0.22\ (T_{aw} - T_e) \tag{14}$$

Using the simplified relationship for enthalpy, $H = T \cdot C_p$, these equations can be modified to give:

$$H_{ref} = 0.22\ H_{aw} + 0.28\ H_e + 0.5\ H_w \tag{15}$$

To conclude this section on aerodynamic heating, a general engineering method for determining vehicle heat loads will be discussed. From theoretical and wind tunnel data, profiles of heating rates to the total vehicle exterior are determined for particular speeds and vehicle attitudes. The heat rate to each specific surface location can then be expressed as a function (fraction) of stagnation point heating. For hypervelocity conditions these functions are approximately independent of both Mach number and altitude. Thus, for a particular vehicle configuration, convective aerodynamic heat transfer to a particular area is a relatively straight forward empirical function of the angle of attack, body location, and stagnation heating rate. Computer programs written to determine (in part) stagnation point heating as a function of vehicle frontal profile, velocity, and altitude can be organized to input information on angle of attack and non-stagnation heating rate functions (as functions of the angle of attack); thus allowing for a total determination of vehicle heat inputs.

SECTION III

THERMAL PROTECTION

Having described the processes of aerodynamic heat transfer to a vehicle, the conservation of this energy will now be discussed. Heat arriving at the vehicle surface must be balanced with subsequent heat transfer or storage. This heat is partially radiated back to space, absorbed by surface materials, or absorbed into solid or fluid heat sinks (Figure 5).

At thermal equilibrium all heat transfer at a surface will be balanced. For example, take the case of a perfectly insulated thin shield (Figure 6a). As aerodynamic heating increases, the surface will absorb heat and rise in temperature; the resulting greater temperature will radiate more heat back to space (proportional to the emissivity and fourth power of the temperature) and thus balance the energy system. This process can continue until the material melts, sublimes, and so forth. Heat radiated outward from the hot surface is often and erroneously termed "heat re-radiated". The prefix "re" implies that the heat originally arrives by radiation. This is not the case since convection is the predominant transfer mode.

For a given heat input to the perfectly insulated wall, that wall temperature which radiates back into space all incoming heat at thermal equilibrium (no heat storage or temperature change) and with no internal heat transfer is called the radiation equilibrium temperature (RET) for the surface. This temperature is a function of surface emissivity:

$$q_r = \sigma \epsilon T_{RET}^4 = q_w \tag{16}$$

T_{RET} is the highest temperature the surface can experience under the given heat input, q_w.

Actual aerodynamic surface structure involves inward heat transfer and transient storage into material mass. This heat transfer, from the hot exterior to cool substructure, will predominately be by conduction, but with internal radiation and convection often being important. For example, a ceramic or metallic heat shield system, efficiently insulated, will only allow a small quantity of heat to penetrate to the interior (typically less than 1 Btu/ft^2 sec). At the other extreme, high energy absorption systems using liquid metals or fuel regenerative cooling (the latter not generally applicable during re-entry) may absorb the major part of incoming aerodynamic heat by forced convection. Gross internal radiation from a thin heat shield may be of the same magnitude as that radiated back to space from the shield exterior. However, on a net basis, inward radiation heat transfer will generally be substantially lowered by the reflective and emissive effects of the enclosed internal spaces, especially if high internal temperatures are present. Internal radiation from windward to leeward surfaces of wing-like components may provide important cooling of the hot (windward) structure. In addition, due to the joint gaps between structural or insulative components and the porous nature of many refractory or insulative materials, radiation is inherently involved in much heat transfer that may appear to be conductive.

It is stressed that convective heat arriving at a surface is first absorbed at the exterior and subsequently controlled by outward radiation, internal heat transfer, and internal storage. The last being accomplished by a sensible rise in material temperature and/or latent phenomena; for example, sublimation or melting. In actual application, weight efficiency considerations which might promote a predominately heat sink or absorption approach for rapid flights, such as ballistic re-entry, might not allow this method for long time re-entry, where major use of radiation may produce a more efficient system. Figure 6 presents typical thermal protection concepts, demonstrating control of heat arriving at the vehicle surface.

The aerospace TPS designer has three tools with which to work. As previously discussed, the first two, absorption and radiation, provide the means to dissipate heat arriving at the vehicle. The third is his ability to reduce the aerodynamic heat before it reaches the vehicle surface. An important characteristic of re-entry and other hypervelocity bodies is that they play major roles in creating or modifying their own environment, particularly the thermal. As previously described, heat transferred to a hot surface is dependent on the wall temperature as it affects the thermal gradient or forcing potential between boundary layer and surface. Thus, heat input can be reduced by allowing a hot vehicle exterior; this is in addition to the greater radiation accompanying the hotter surface. The effects of the body on forming, modifying, or insulating itself from the boundary layer are also of great importance to high energy absorbing TPS. Insulation can be accomplished by placing a thin liquid or gas film on the vehicle surface through discrete openings (film cooling) and, in effect, adding a cool cover to absorb heat before it reaches the vehicle. Of more application importance and utility is the process of gaseous injection into the boundary layer. In proportion to the rate of mass transferred, these ejected gases will tend to thicken the boundary layer and reduce the thermal gradient adjacent to the vehicle surface. A mass flow limit is reached where the gradient vanishes and the boundary layer will separate from the surface. In addition, for the typical case where ejected gases are of substantially lower temperature (T_w) than the boundary layer, these gases will absorb heat within their own mass. Effectively, these processes lower the thermal driving potential producing convective aerodynamic heating. Increases in boundary layer turbulence may tend to reduce this beneficial "heat blockage". Gaseous injection can be from a degrading and/or porous vehicle surface or from a dissipating surface film (Figure 7), the latter being aided by the degree of surface turbulence.

The extent of reduction in aerodynamic heating by gas injection into the boundary layer is expressed by the heat blockage term, q_{BL}. The net convective heat reaching the vehicle surface is $\dot{q}_w - q_{BL}$. Thus, the energy balance at a vehicle wall with simple mass injection can be expressed:

$$\dot{q}_w - q_{BL} + q_{gc} - q_r = q_{abs} \tag{17}$$

where q_{abs} is the energy absorbed and/or transferred internal to the exterior surface. For a fluid transpiring through an inert, non-degrading permeable matrix at equilibrium (that is, no heat storage within the solid matrix) and all heat being absorbed by the fluid with no other internal heat transfer:

$$q_{abs} = \dot{M}\,(H_{tw} - H_{tc}) \tag{18}$$

where subscripts tw and tc define the transpirant at the hot matrix exterior and cool inner side, respectively. This absorbed heat can be quite large depending upon the transpiration system design.

For an ablative surface the basic relations must be modified to account for char formation, receding exterior, mass loss ($\dot{m}$) coming from a receding zone or zones within the matrix, and so forth. The fraction, f, is used to distinguish mass transferred into the boundary layer from that forming a char:

$$f\dot{m} = \dot{M} \tag{19}$$

Heat blocked is expressed by the following empirical equation form:

$$q_{BL} = \dot{M}\beta\,\Delta H \tag{20}$$

In this relation ΔH is an enthalpy difference which can take one of the following forms: $(H_s - H_w)$, $(H_e - H_w)$, or $(H_{aw} - H_w)$, depending upon vehicle location and best data correlation. Note that these enthalpies would be per unit weight of air at the respective location temperatures.

The transpiration factor, β, is a function of the flow conditions (laminar or turbulent), the transpiring gas thermodynamic and transport properties, and the transpiring gas molecular weight. Numerous investigators have utilized the following as an expression for this factor across a boundary layer:

$$\beta = N\left(\frac{\text{mol air}}{\text{mol gas}}\right)^{\alpha} \tag{21}$$

This expression is used in Equation 20 with $H = (H_e - H_w)$. It is most applicable where the transpiring products have molecular weights close to that of air. Experiments have determined ranges of values for the "curve fitting" exponent, α, and factor, N. For laminar flow, α is often reported as 0.25 and N between 0.6 and 0.7. Heat blockage for turbulent flow is less well understood. Compared to laminar flow values α is generally larger (0.3 to 0.4,); whereas, N is less (for example, to 0.2). It should be noted that these quantities are given in this report solely to depict general number magnitudes, not as design factors. However, using these numbers for laminar flow ablator mass transfer, where free stream and ejected gas molecular weights are approximately equal, β is often approximated as the constant N; for example, 0.6. The above relations are not generally valid for large heat blockage; for example, where $q_{BL} > 0.7\ q_w$.

Some empirical formulations of β may have the gas molecular weight expressed to a power different from that for the undissociated air molecular weight. One laminar flow expression which modifies the general expression of Equation 21 for light gas transpiration is presented as another example of an empirical expression form for β :

$$\beta = \frac{1.6}{(\text{mol gas})^{0.6}} = (0.67)\,\frac{(\text{mol air})^{0.25}}{(\text{mol gas})^{0.6}} \tag{22}$$

This empirical formula varies from Equation 21 only in the denominator. Also, it is based upon $(H_{aw} - H_w)$, not $(H_e - H_w)$. These differences tend to balance each other in the overall relationship of Equation 20.

The form of expressions for β and use of these approximations depend primarily upon the flow type, the percent of incoming heat blocked, the ratio of air to gas molecular weights, pressure of velocity profiles of the boundary layer at specific vehicle locations, number and types of injected gases, and the ratio of wall temperature to boundary layer edge or adiabatic wall temperatures. As shown above, the form of β will depend upon its associated ΔH. Thus a smaller β would be associated with $(H_{aw} - H_w)$ than would be used for correlation with $(H_e - H_w)$ for the same heating condition. In order to more fully appreciate the total concept of heat blockage and its controlling factors, one might visualize two processes occurring. The first would be a thickening of the boundary due to the presence of incoming mass. The other process is that of heat being absorbed by the ejected gases in the boundary layer. The latter is incorporated in Equation 20 by the term H_w.

Referring to Equation 17, two additional comments are applicable. First, the difference, $\dot{q}_w$ minus q_{BL}, is often lumped and referred to as a "blowing" heat input ($\dot{q}_{BLOW}$); that is, the net aerodynamic convective heat reaching the vehicle wall. Second, where ejected gases appreciably burn, an additional term, q_{COMB}, is added to the equation's left hand side to handle the convective and radiant heat transfer of this energy. [Boundary layer burning is characteristic of ejected hydrogen, carbon, and hydrocarbon ablator products.] The presence of this combustion heat is physically related or incorporated into the actual boundary layer thermal profile. Its transfer is generally treated as separate from the "regular" aerodynamic heating ($\dot{q}_{BLOW}$). Equation 17 can be rewritten as:

$$\dot{q}_{BLOW} + q_{gc} - q_r + q_{COMB} - q_{abs} = 0 \tag{23}$$

Combustion heating is, naturally, dependent upon the oxygen in the boundary layer, or lack thereof, in some planetary atmospheres. Equations governing q_{COMB} will be either rate or diffusion limited. Thus, where oxygen is plentiful combustion energy will be proportional to the rate of "unrestricted" burning. With low oxygen concentrations the speed that this oxygen can reach or diffuse to the burning material will govern the rate of combustion.

Referring to Equation 21, the most efficient injectants would be low molecular weight gases. The lower the gas molecular weight, the lower will be the net incoming heat transfer, skin friction, effective recovery factor (r), and the ejection rate required for boundary layer separation. These lower weight gases will diffuse more readily and thicken the boundary layer. It should be noted that hydrogen, providing the lowest molecular weight, is a major product in the pyrolyses of organic ablators. In addition to excellent heat blockage it has an extremely high specific heat for sensible energy absorption. As previously stated, the average molecular weight of typical organic ablation products is not too far different than that of air.

Two TPS methods that block incoming aerodynamic heat by gaseous injection are active transpiration and ablation. The former utilizes an inert matrix through which a stored fluid (either gas or liquid) is actively pumped. The latter method generates its own gas primarily by pyrolysis. As will be shown, most ablation processes involve forms of transpiration flow. It should be noted that passive transpiration systems with fixed (non-degrading) exterior surfaces could be formed using a back-face ablator under forced contact with an inert outer porous material (Figure 7).

To clarify and provide definition to an area of much confusion, a matter of terminology will be discussed. A passive system is defined as being self-initiating and self-regulating; for example, solid heat sink, radiative shield, and (by this definition) most ablators. Active denotes a requirement for external control, either manual or automatic. Some define passive as being inert and active as a system where material mass is transferred. Ablators would be classified as active by the latter definition. However, this presentation will consider ablation as passive since it is self-initiating and self-controlling. This definition is based on the mode of TPS control.

Being passive, relatively easily fabricated, and economical, ablator heat shields are applicable to numerous vehicles and environments. These range from design requirements where many potential TPS concepts could theoretically be used, such as, moderate lift entry from orbit, to severe thermal conditions where boundary layer injection or other mass transfer must be used to maintain proper protection of vehicle structure; for example, ballistic and most superorbital lift entries. Only ablation is currently practical for use in many of these severe cases. For moderate thermal conditions over long flight times, use of metallic or ceramic heat shields generally appear most efficient. However, the continuing and rapid development and refinement of ablators from dense, heavy materials to lightweight, insulative composites has increased their utility for long flight time missions at low to moderate heating; thereby placing this class of materials in a competitive position with theoretically efficient, non-degrading heat shields for many applications. In addition, critical problems posed by oxidation protective coatings, maintenance of material properties at high temperature, thermal shock sensitivity, material expansion matching, fabrication control, and heat input limitations are reduced or eliminated in ablating systems, as opposed to metal or ceramic structures. Needless to say, any ultimate selection of vehicular thermal protection types for specific application must rest upon sound, open minded judgment and detailed trade study between applicable concepts.

SECTION IV

ABLATION DESCRIPTION

A basic picture of the aerodynamic heating problem and common techniques utilized for thermal protection has been presented. This provides a frame of reference showing where the phenomena of ablation fits into aerospace technology and why it is required for many system applications. The ablator, per se, will now be described, using charring ablator composites as models. These are formed from organic materials composed of natural and synthetic polymers such as plastics and elastomers. These organic composites, as will be discussed, often include inorganic constituents.

Without restricting ourselves solely to any specific class of ablators, such as organics, one could paraphrase a popular conception of an ablator as an exterior material or shield that is sacrificed or burned in order to soak up aerodynamic heat and so protect substructure, personnel, and equipment. A more meaningful definition would point out that (generally) combined physical, chemical, and mechanical degradation occurs by thermal, aerodynamic, and mechanical actions. In addition, partial blockage of incoming aerodynamic heat is accomplished in the boundary layer, as previously described. This involves boundary layer thickening, further endothermic material degradation, and temperature increases in the expelled products; thereby effectively absorbing and rejecting aerodynamic heat before it can reach the vehicle.

It should be pointed out that any solid or even liquid could be made to ablate, considering the overall meaning of the word. One can trace the early progress of ablator development starting in the 1950's from initial metallic heat sink concepts, to the melting of the heat sink (for example, copper), to high melting and vaporization point glassy materials (for example, quartz), to subliming compounds, and to the many types of charring ablators.

These charring ablators form a varied and versatile class embodying many of the processes or phenomena of earlier (and simpler) concepts. Thus, the charring class of ablators is the product of an extensive chain of technological development. This progress is continuing both within the charring class and with other basic material forms. For example, the ablation of ceramic, carbon, and graphite solid forms are considered as substitutes for the conventional (polymeric) charring ablator for some very high heating, high aerodynamic shear stress applications. However, carbon and graphite solids are actually a type of charring ablator; that is, they are effectively chars to begin with. Ablator descriptions are as complicated as the many different mission environments, materials selections, fabrication processes, and structural compositions that can be used.

Figure 8 shows typical phenomena occurring in or with ablators. These can range from simple physical changes (such as might occur if the ablator were ice) to complex pyrolysis and physical degradation. Pyrolysis is defined as chemical decomposition by the action of heat. Although physical changes with their resulting latent and sensible heat reactions are important to the heat sink ability of most ablators, the major heat absorption within the organic ablator is generally by pyrolysis.

A description of a typical polymeric ablator composite must address itself to two subjects; that is, ablator composition, and the thermal reactions that occur during ablator degradation. These shall be covered by first portraying a simplified pyrolysis history of a charring organic resin and then describing ablator constituents

their functions, and the structure of the typical ablator composite. The latter discussion will show how the thermodynamic and mechanical properties of an ablator can be tailored by proper material application. Let it be stressed that not all organic resins leave chars; some, like Teflon, completely gasify upon pyrolysis and are commonly termed "sublimers".

SECTION V

ORGANIC PYROLYSIS

Referring to Figure 9 one can trace the thermal history of a virgin organic ablator as it undergoes pyrolysis and physical degradation. At the onset of aerodynamic heating the material surface temperature rises as energy is absorbed. Heat is also conducted inward; the system thus behaving like an inert heat sink. Rapidly, as indicated by steep transient thermal gradients, the surface reaches a temperature where resin pyrolysis occurs. It should be noted that compared to ceramics and metallics, ablators have low conductivities; thus promoting large thermal gradients. With the onset of pyrolysis, the ablator surface decomposes, evolving gases and (for charring ablators) leaving a porous residue. Two primary zones are formed above the virgin plastic; the coke-like char and a pyrolysis zone. Under certain circumstances, a third major zone, called a melt zone, may form. This is usually associated with silica reinforced composites under moderate heating, where heat is inadequate to vaporize the silica, and molten glass flows on the surface. Our immediate thermal model, pure resin, would not have this glassy zone.

The char is relatively inert, compared to the pyrolysis zone. However, reactions do occur in this carbonaceous zone; for example, oxidation or combustion of the carbon, further chemical degradation and enthalpy change as temperatures rise in the transpiring gases that evolve from the pyrolysis zone, and, if the exterior gets hot enough, the carbon or graphite may start to sublime. In essence this charred zone performs three major functions: it provides a high emissivity, high temperature exterior for heat ejection by radiation; it acts as an insulative barrier between the hot gases and unpyrolysed plastic; and it provides a relatively inert porous matrix through which gaseous products transpire. These products tend to cool and protect the char from oxidation attack. As the char thickens, it performs its thermal functions more effectively. However, material degradation, thermal shock, and mechanical or gas flow removal of the char complicates the process. Consider two extremes of char formation. In one, a relatively weak, thin char forms. It is evenly and predictably degraded, but poorly insulates. At the other extreme, the ablator forms a thick, fairly strong, and highly insulative char. However, being basically brittle and subject to thermal shock and other failure mechanisms, this thick char is prone to breaking in large chunks at unpredictable times or rates, thereby forming rugged contours with localized heating and perhaps catastrophic failure. Ablator research and design objectives must resolve between these extremes.

The receding pyrolysis zone is basically at one temperature; that is, the pyrolysis temperature. This zone may be of infinitesimal thickness (a line zone) or of finite thickness. Generally, in the low heat flux ablators used for lifting re-entry, the zone would have finite thickness and a thermal gradient would be present within the zone. High heat flux ablators and sublimers might have an isothermal line zone. For a given material the zone temperature strongly depends upon the rate of pyrolysis which, in turn, increases as the heat flux increases. The zone temperature usually varies in proportion to the rate of pyrolysis. Most chemical reactions occur within the pyrolysis zone.

Behind the pyrolysis zone lines the virgin plastic, bonding agents, and backface structure. Being a good insulator, the virgin plastic has accenuated exponential or parabolic-like thermal gradients (Figure 9); these have maximums at the pyrolysis temperature. For organic resins (phenolics, epoxies, silicones) subject to typical manned re-entry heat fluxes this maximum temperature is below 1000°F. It follows that the internal ablator surface (the bond line) would be substantially

below 1000°F until just prior to complete ablator removal (were the system designed for this removal) under moderately high heating. Under lower heating where the overall thermal gradient between the pyrolysis zone and bond line is not so steep, high bond temperatures may develop with appreciable ablator thicknesses remaining. This case would require added ablator thickness to insulate the bond line.

It should be noted, however, that inclusion of high pyrolysis temperature reinforcements and fillers may increase the overall or effective ablator pyrolysis temperature above that of the basic resin. Also, pyrolysis temperatures generally increase as aerodynamic heating becomes excessive. Thus, bond temperatures well above 1000°F are possible. Control flaps are an example of this condition.

SECTION VI

ABLATOR COMPOSITION

The typical "plastic" ablator is a composite of numerous organic and inorganic materials. These are used to form the three major ablator constituents; that is, resin, reinforcement, and filler. Constituent types, material ratios, orientation of the reinforcing agent, and materials processing and fabrication are major parameters affecting such ablator behavior characteristics as heat absorption capability, virgin plastic strength, char strength, conductivity , and efficiency. Environmental parameters which pertain to the aerodynamic gases and affect material behavior include gas enthalpy and composition or reactivity, mechanical forces of dynamic pressure and shear, and type of aerodynamic flow (laminar or turbulent).

A large variety of thermoplastic and thermosetting resins and elastomers may be used singularly or in tailored combinations to form the resin systems; for example, phenolics, epoxies, silicones, and polyesters. Each resin or combination has its own thermochemical and mechanical characteristics.

Thermal degradation of the resin is a highly complex function of environment and resin temperature. A typical thermogram (Figure 10) shows pyrolysis products as a function of temperature. Generally, higher reaction (pyrolysis) temperatures and pressures yield lower molecular weight gases. As previously stated, lower molecular products indicate more complete degradation and heat absorption, and more effective boundary layer blockage. This diagram gives an indication of the amount of residue or products not directly becoming gaseous. A non-charring resin which completely decomposes to gases, such as Teflon, may appear quite efficient from the standpoint of mass transfer cooling and heat blockage. However, materials leaving residue, though less efficient in gas ejection, may become more efficient when all thermal aspects are considered. Thus, the formation of a thick char provides an insulation barrier between the hot aerodynamic gases and the virgin plastic. This beneficial layer reduces heat input and mass loss at the pyrolysis zone. Due to the high thermal gradient, it can produce an extreme outer surface temperature with inherent out-radiation at high emissivity. Typical charring ablators may leave char residue up to 50 percent of virgin plastic weight. This char has a potential temperature limit well above 5000°F, 6600°F being the sublimation temperature of graphite. However, for typical lift re-entries oxidation and mechanical removal will generally degrade the char prior to its reaching temperatures above 6000°F.

Reinforcements are usually added to the resin for strengthening both the virgin material and the char system. A wide variety of materials, forms, and orientations are used; for example, chopped fibers, mats, tape wrapped, or honeycombs of carbon, glass, nylon, or asbestos. The form or orientation of these materials greatly determines overall system conductivity, erosion resistance, and strength. For example, a system with glass reinforcement fibers oriented perpendicular to the external surface would be more conductive inward, due to direct heat transfer along the high conductivity fibers, and would tend to be less prone to erosion or delamination in flight than a system with fibers parallel to the surface. With the latter system, direct paths for heat transfer are greatly lessened, but delamination of the hot surface and surface spalling due to internal pressure from trapped gases are strong possibilities. Trades between these two extremes often dictate that oriented reinforcement be laid at oblique, downstream angles to the local flow. This is typical of tape wrapped shields for missile nose cones and manned re-entry vehicles. Random fiber orientations can also be used. These give more or less

intermediate material properties compared to isotropic orientation extremes. Combinations of reinforcements may be used; for example, honeycomb reinforced porous ablators with fibers in the resin.

Open-face glass or quartz fiber honeycomb impregnated with reinforced porous plastic currently receives much attention in manned re-entry due to both structural and thermal requirements. Cold soak conditions in space (to -290°F) pose a most severe structural test of an ablator. At these temperatures resins become brittle. Being subjected to shrinkage and differential thermal expansion, structurally strong composites often fail by internal cracking, delamination, or peeling from the substructure. However, by using a relatively weak (low modulus) resin system, one can avoid this type of failure. Weakness can be provided by chemical treatment or porosity. The porous material systems also provide good thermal efficiency and insulation of the back-face structure. Methods must be employed to maintain material structural integrity, attachment, and char retention. The open-face honeycomb and modifications thereof may be used to fulfill these requirements. Stress problems in the resulting composites are thus reduced or eliminated due to flexibility. Recent advances with elastomeric ablators have produced porous materials not always requiring honeycomb-like reinforcements for moderate heating application. In addition, some of these new materials remain viscoelastic to temperatures approaching -200°F, thus being highly compatible with space soak thermal conditions.

Fillers form the third major constituent in an ablator composite. These are particles or fibrous inclusions which are added to the resin for numerous reasons. They may increase the absorption of heat, form large volumes of gases, lower material strength and conductivity, increase surface emissivity, and so forth. Phenolic or silica microballons are examples of a filler used to lower the composite conductivity by providing voids. These are often used in the honeycomb reinforced ablators previously described. Foam techniques are used as alternates to balloon fillers.

The ablator composite may be constructed in layers with thin overcoats for boost heating and with micrometeorite shielding materials. For the prime thermal protection task, multilayer composites may be formed with the inner layer(s) serving an insulative or low ablation temperature function. These composites can be constructed in a honeycomb or modified honeycomb matrix, the latter being a matrix where the cell walls are modified from hexagonal to a variety of unique shapes. This type of matrix can be used to give better flexibility for surface contouring the ablator around the vehicle profile, improve directional char retention, or to lower ablator strength and modulus. Filament or tape wound composites may be constructed in multi- as well as single layers. Use of thick, elastomeric bonds between the ablator and back-face structure effectively adds another ablator layer to provide flexible support and attachment stress reduction.

Numerous plastic forming processes are applicable to ablator fabrication. Typical techniques include molding and casting of shaped parts for subsequent attachment to a structure. Hand lay-up and resin application of tape wrapped ablators directly on flight structure, using the back-face structure as a mandrel, is typical for bodies of revolution (for example, nose caps). Machines are used for some tape wrap operations and for filament wound composites. Spraying, painting, and bonding lay-up of sheet cloth or reinforced plastic are utilized. Open-face honeycomb may be filled by hand guns, by hand troweling and packing tools, or by molding integrally with the resin. Machine contouring in conjunction with many of the above operations produces finished shapes.

SECTION VII

APPLICATION

Material selection for and the design of ablative composites is a direct function of mission/environment characteristics, vehicle configuration, and vehicle peripheral location. Associated with these during re-entry are heating rates and types, total heat inputs, static and dynamic pressures, shear stresses, gas enthalpies, and so forth. Boost and space conditions of temperature, pressure, micrometeoric and rain particle impingement, and radiation are general factors for overall ablation system design. When considering hypervelocity vehicles (low altitude missiles, ballistic warheads, drag re-entry capsules, and lift re-entry vehicles) one can classify applicable ablation materials into two general groups. The first includes those for use under extreme heating and high gas forces. The second group is for use under low to moderate heating and erosive actions.

The former class is required for stagnation, forward lifting, and control surface regions, where surface recession resistance and large heat absorption capabilities are required under severe heating. These ablators are generally dense, hard, strong, and of relatively high modulus of elasticity. Thermal conductivity is also relatively high. Reinforced phenolics and graphitics are prime examples of this class; polyaromatic resins are recent candidates. High silica content solids (generally reinforced), ablative ceramics, and internally ablative refractories are also included in this class of ablators. Charring ablators (organics) of this group typically utilize an inorganic reinforcement, such as silica, glass, carbon, or graphite fibers and woven forms.

Even the most attractive ablators for stagnation areas will experience some recession under extreme heating. However, except for small radii caps and edges, recession of leading surfaces is typically small with respect to the original dimensions and is concentrated towards the stagnation point or line. For example, at a lifting body nose the ablator external radius of curvature will not appreciably change and the vehicle cross section (viewed head-on) will remain approximately constant. Therefore, vehicle stagnation heating and body drag (both pressure and friction) will not vary appreciably.

On the other hand, small control vanes for hypervelocity missiles may experience degradation of a large percentage of effective area with accompanying increased heating and possible malfunction. Sharp forward regions of high lift/drag vehicles may also have their leading radii reduced with an appreciable increase in heating. However, the overall projected frontal cross section and drag characteristics would generally remain fairly constant.

The second ablator grouping mentioned above has prime application to non-stagnation or 'body' areas, such as: the leeward surfaces of conical re-entry capsules. wing-like appendages, and lifting body shapes; and much of the areas of orbital re-entry lifting-body windward surfaces. Large radii forward surfaces of high drag re-entry capsules may also utilize these materials. Part time ablative overlay of radiative structure is typically performed with these materials. Materials range from porous phenolics and epoxies, subliming salt compositions, and impregnated carbon cloth, to elastomeric composites generally formed from silicones. Elastomeric material systems are currently highly promising for use as body ablators. Typical composite types include spray-on, cast, and honeycomb reinforced forms with or without fiber reinforcement. These materials may be self-foaming or contain porous fillers to provide insulation.

'Body' ablators are generally constructed to be efficient insulators as well as efficient absorbers and boundary layer heat blockers. They generally are designed to minimize surface recession under low to moderate heating. As an example, some silicone compositions produce no surface recession at heating rates below 120 Btu/ft^2 -sec within typical mission heating periods (30 minutes). In fact, their surfaces may actually swell or 'bloat' as temperatures are increased.

Compared to dense, stagnation area ablators, porous 'body' area ablators are weaker but more insulative. Therefore, they are more efficient for shielding interior structure from heat conducted from the hot pyrolysis region. However, they are not designed to withstand severe heating and severe gas impingement and are, therefore, limited in application.

SECTION VIII

EFFECTIVENESS

In order to select and judge the performance of an ablator composite or system, one should be directed to a specific mission/vehicle application. For the particular thermal and erosive environment, determination of ablator thicknesses and weights can be performed by theoretical or theoretical/empirical techniques. The former theoretically calculates detailed thermochemical/physical reactions and material degredation, and requires knowledge of char properties and gas kinetic data. The later technique utilizes test data obtained under simulated flight conditions (e.g., gas pressures, enthalpies, and constituents) to determine ablator mass degredation. Both methods use the thermal properties of the virgin ablator to calculate backface temperature rises and insulation requirements. Electronic data processing equipment for calculation efficiency is widely employed. Thus, the all-theoretical technique requires knowledge of highly complex and often unknown chemical reactions which, in turn, depend upon complex thermal and fluid environments. The semi-empirical technique circumvents this problem by tabularizing the net material performance data, obtained by test, as a function of environmental parameters without actually requiring a thorough understanding of the ablation process. Test data are usually obtained from plasma facilities where good simulation can be obtained.

Tests conducted to determine ablator properties are generally performed under a constant input environment. This allows accuracy and simple standards of comparison in measuring heating parameters and degradation rates for various materials. Data collected is in a form that is readily applicable to analyses; for example, rate of mass loss per given rate of heating, surface and pyrolysis temperatures versus respective heat input rates.

For example, a typical test run might begin by setting a plasma heating apparatus at a specific environmental level of gas enthalpy, velocity, and pressure. Heating rates would be measured next to a calorimeter of known temperature. Finally, a series of ablator coupons would be subjected to the same facility environment over a measured time duration. Oxy-acetylene burners may be used as heating apparatus for these effectiveness measurements, where only heat input simulation is desired.

The test run provides a known standard of heating to a cold calorimeter (cold wall heating rate) and the gas velocities, enthalpies, and pressure producing the rate. Measurements of each ablator tested would include mass of material consumed, char thickness, erosion distances, internal temperature distributions, and so forth, for each measurement producing information required in analytical procedures.

Although each test series would be performed using constant energy inputs to the facility, the actual heat fluxes arriving at the hot ablator surface would not be directly known since these (hot wall) heating rates are dependent on the surface temperature. Subsequent to measuring, calculating and/or estimating the surface temperature the actual or hot wall heating could be determined or approximated. It should be noted that the surface temperature is extremely difficult to measure due to complex emissivities and luminescence of hot gases.

In essence, therefore, the ablator is subjected to test conditions which can be fairly accurately measured and reproduced (enthalpy, pressure, and velocity), thereby providing a standard for material comparison and further calculation. It should be stressed that this presentation is very basic and simplified and should not be considered as representative of all ablator testing. For example, some testing must actually determine hot wall heating for basic research in ablator reactions; other testing is conducted in static environments. To repeat, the above test description is representative of a basic and easily performed empirical method to compare various ablators for effectiveness without fully understanding or knowing the thermochemical and mechanical phenomena occurring during degradation. Verification of the effectiveness of this method for analyzing ablators comes when ground test data and analytical methods adequately predict flight conditions.

Early ablator application was primarily to ballistic vehicles where very high rate, short duration heating was experienced. Ablator performance effectiveness for these applications is primarily a function of absorption, radiation, and heat blockage. It is measured by a term called the "effective heat of ablation". This quantity gives an indication of heat shielded per amount of material sacrificed. Generally, it is empirically determined as previously described. "Effective heat of ablation", often denoted as $\overset{*}{q}$, is the quotient of heat transfer that would occur to a non-ablating body at the ablator's surface temperature divided by the mass ablated, Btu/lb being the dimensional units generally used. Figure 11 defines $\overset{*}{q}$ formulations.

The "effective heat of ablation" does not account for internal conduction to the vehicle structure nor the added amount of ablator that would be required to insulate the inner vehicle structure in order to keep it below maximum temperature allowables. These considerations are important to many ablation systems currently being developed. However, the usefulness of $\overset{*}{q}$ for ballistic-like flights is not greatly affected by these deficiencies of internal heat transfer due to the shortness of flight duration. Simply, the weight of consumed ablator is much greater than the excess required for insulation. $\overset{*}{q}$ provides a good indication of ablation weight required as a function of heat input and can be used in transient analyses to determine weight loss rates and total amounts of ablation material consumed.

For manned lifting re-entry and other long time hypervelocity flights (30-60 minutes), the governing design parameter often is the allowable temperature maximum for the internal structure. The design engineer must, therefore, select a "good" ablator from a heat dissipation standpoint and then determine the ablator system thickness required to shield the back-face structure for each particular mission. This determination is accomplished by assuming an ablator thickness, analytically subjecting the system to the design trajectory heating environment, and calculating the back-face temperatures produced. These trial-and-error (computer) runs are performed until the minimum weight (and thickness) system is determined. This system should have a back-face temperature just below the design allowable maximum for attachment bonds or the back-face structural materials.

The ablator system for these applications may have varying properties in the thickness direction. Thus, the exterior materials would tend to be good heat absorbers and dissipators, whereas interior layers would be highly insulative. These inner materials or layers probably could be designed solely as structural insulators and not be required for actual ablation, except where mission perturbations might excessively use up the ablator and the inner layers, acting as "factor of safety" material, would actually be ablated to complete the mission.

A parameter applicable to these long time missions must include insulation and non-used ablator weights, not just the amount of material ablated as in the case of $\dot{q}^*$. The term "effective heat capacity" was coined to measure the amount of energy (Btu) into an ablator surface divided by the total initial ablator system weight. The effective heat capacity is empirically determined using techniques similar to those in the method for determining $\dot{q}^*$. However, this parameter is time dependent and cannot be directly employed as a general analysis factor. Thus, it is mainly used to rank ablator effectiveness for a given thermal subjection.

It should be noted that the numerator for both ablator effectiveness parameters can use different heat quantities. Observing the heat transfer at an ablator, or any other surface, it can readily be seen that the net heat being absorbed into the ablator is the difference between that arriving at the surface and that being radiated outward. Using this absorbed energy in the ablator effectiveness parameters may radically decrease the parameter's numerical value. This would not be representative of the ablator's total performance since both absorption and radiation are important ablator functions. However, by eliminating radiation from the parameter; and using only the heat absorbed, one can express the absorption or (neglecting internal heat transfer) thermochemical plus thermophysical effectiveness of the ablator. This specific aspect is important for selecting material constituents and analyzing the ablator operation. Either way, the heat blockage effectiveness is included in both parameters.

SECTION IX

ATTACHMENT AND VEHICULAR INTEGRATION

The attachment of the ablator to its back-face structure (that structure which supports the ablator) requires consideration of both loading and thermal requirements. Briefly, the attachment of an ablator to its supporting back-face can be by adhesive (or possibly cohesive) and/or mechanical methods, the former being more universal. Attachment strength, effects of ablator/bond combinations on substructure, and manufacturing requirements and methods all are important factors in the selection and design of attachment methods. The back-face structure may take many forms: a metallic or ceramic heat shield protected part time by the ablator; a portion of a vehicle or component prime load carrying exterior structure such as a superalloy or refractory metal wing utilizing the exterior as a partial load carrier; or a non-structural stiffened panel. The last form would be typically utilized with insulation over conventional fuselage structure of aluminum or titanium (Figure 12) as determined by overall vehicle weight trades. Considering the moderate temperatures required of manned and equipment carrying vehicle compartment walls, and the major problems and costs of internal thermal control were the wall structure operated hot, fuselage structural design (under aerodynamic heating) will tend toward as low an internal wall temperature as possible. A typical design goal for a manned re-entry cabin wall is for temperatures under 200°F. Thus, the potential use of the non-structural heat shield, shown in Figure 12, for non-stagnation areas is great.

The ablation system must be properly designed so that the bond region temperature does not exceed allowable bond material strength or decomposition limits. The latter occurs at temperatures well above those for the former but may be important in localized areas. Bond strength requirements during ablator re-entry are generally not critical due to low loading and low ablator modulus of elasticity. However, as previously stated, at moderate temperatures and below, many ablative plastics have relatively high modulii of elasticity and are prone to internal and bond separation failures under differential thermal expansion. These failures may also occur during fabrication cool down from elevated bond or plastic cure temperatures, or from sterilization thermal soaks. Where the ablator composite or material cannot be designed or compounded with a small modulii of elasticity, engineering design of the high modulii systems can attempt to match ablator expansion coefficients with those of the substructure, design the substructure so that it deforms with ablator, provide extremely strong bonding agents, or provide flexible bonds of appreciable thickness. The last alternative allows bond stresses to distribute over large areas, thus reducing peak localized stresses at the ablator edges. It is rather applicable where cast or otherwise preformed ablators are attached to substructure. RTV (room temperature vulcanized) silicone rubbers are prime candidates for flexible bonds.

The use of mechanical methods for ablator attachment is generally limited to design problems where adhesive attachment alone may not suffice. Mechanical fasteners might be used where the bond or attachment region operates at excessive temperature and would probably be used in conjunction with adhesive bonds. Their use as supplements to adhesive bonds takes added importance when one considers that theoretical adhesive characteristics may not be effective due to manufacturing limitations. Mechanical attachment can take such general forms as threaded fasteners, deformable clips where the plastic in various states of cure is forced over straight metallic clips causing crimping, and non-deformable clips over which the ablator would be molded.

The surface effects of the ablator and its attachment on substructure must be considered, especially for dual protection systems where the ablator is used as an overcoat on radiative heat shields for part time protection against the initial re-entry heating pulse(s). Effects on the oxidation protection function of refractory metal coatings must be determined for each ablator/bond considered for application. The multiple use of refurbished refractory metal panels using ablative overlays will require successful resolution of the problem of removing the loose, powdery, exterior layers which form on the coating during re-entry exposure in order to obtain proper ablator adhesion without ruining the coating. Similarly, chemical and physical effects must be considered when attaching an ablator to ceramic shields, particularly if light weight porous ceramics are used.

The above discussions have been oriented towards the attachment of a single ablator unit to a rigid panel or back-face. Joining of individual ablator pieces and attachment to non-rigid back-faces are two additional technical areas that require resolution. The latter area includes attachment over expansion joints in back-face structures, warpage or bowing of the back-face from gas pressure differentials of component flexing, and the general warpage of ablator and back-face under differential thermal expansion of the dissimilar materials. This expansion is most important during thermal drops to cold space temperatures; that is, ablator shrinkage causing inward (concave) bowing. Being a function of the relative stiffnesses of both ablator and back-face, general panel bowing is proportional to the ablator thickness and modulus of elasticity (assuming ultimate ablator tensile strength is not surpassed). In addition, differences between ablator and back-face expansivity, thermal gradients, net changes in material temperatures, and panel sizes determine the degree of bowing. It should be noted that not all space soak conditions lead to vehicle exterior temperatures that surpass the ablator's ductile/brittle transition, nor do these temperatures always reach the region where ablator strengths and modulii become strong relative to back-face structure. Ablator thicknesses for body locations are often not sufficient to cause bowing, especially when considering that the vehicle surface will generally have convex curvature to begin with.

Due to both concave ablator/panel warpage and differential expansion within the ablator, ablator joints on the vehicle periphery will tend to enlarge during cold space soak and to diminish in size upon re-entering heating. Engineering design must properly size and/or fill the gaps with flexible material. Joints must also be correctly oriented relative to the aerodynamic flow to reduce erosive action and local hot spots.

Optimization of the overall thermal protection system for a hypervelocity vehicle requires evaluation of the many types of structural cooling, insulation, and combinations thereof. Integration and trade-offs of thermal protection and structural concepts with vehicle configurations and trajectories must be accomplished to provide overall mission/vehicle system efficiency. Such trades will often lead to vehicle systems utilizing total mission ablative plus radiative or other types of thermal protection over areas where each is respectively more efficient. Thus, designs involving ablative shields adjacent to refractory metal heat shields will be found on advanced vehicle concepts. Figure 13 shows a possible layout for a moderate L/D vehicle which utilizes various thermal protection concepts for overall efficiency.

The practical integration of ablative with non-ablative thermal protection systems along the periphery of a vehicle exterior requires consideration of numerous technical factors. In addition to general structural, construction, and ablator attachment problems, factors of foremost importance include: the maintenance of surface continuity at ablator/non-ablator joints from aerodynamic and aerodynamic

heating standpoints; internal heat transfer from hot gas leakage through joint gaps or by conduction and radiation from hot structure to the relatively cool ablator backside; and local and overall effects of ablation products on downstream exteriors. The last factor is important from both chemical and erosive aspects. The chemical aspect might involve reaction or mass transfer between glassy ablator products deposited on refractory metal coatings (silica base); whereas erosive action would be produced by char particle impingement. Such impingement on deflected control surfaces adds to the severity of the general problem.

It should be noted that the optimum thermal protection arrangement for the re-entry portion of a manned vehicle mission may be inadequate for the various abort trajectories that must also be considered. Not only do heating rates and net heat inputs differ between normal and abort trajectories, but also the relative thermal profiles in areas of peak heating will vary as functions of vehicle attitude. The abort design condition may, for example, dictate that a vehicle area, nominally suitable for a radiative heat shield, be designed for ablatives due to excessive short-time heating. These ablative heat shields could take the form of overlays over radiative structure or efficiency trades might dictate that total mission ablation shields be used.

SECTION X

CONCLUSIONS

In accordance with the intent expressed in the Introduction, this document has presented important general properties of the charring class of ablators and has related ablation to aerodynamic heating and vehicle thermal protection. It should be recognized that simplified technical relations and descriptions must always be treated with discretion in conjunction with sound appreciation of the complexity of actual physical phenomena. However, simplified relations, and a thorough understanding thereof, form a valuable tool for technology management. Basic, qualitative technical knowledge is not usually out-dated. Therefore, this document, in future years, should continue to function as an elementary basis for the understanding of ablation thermal protection, even in light of current rapid advancements and the future potential growth of ablator technology.

SECTION XI

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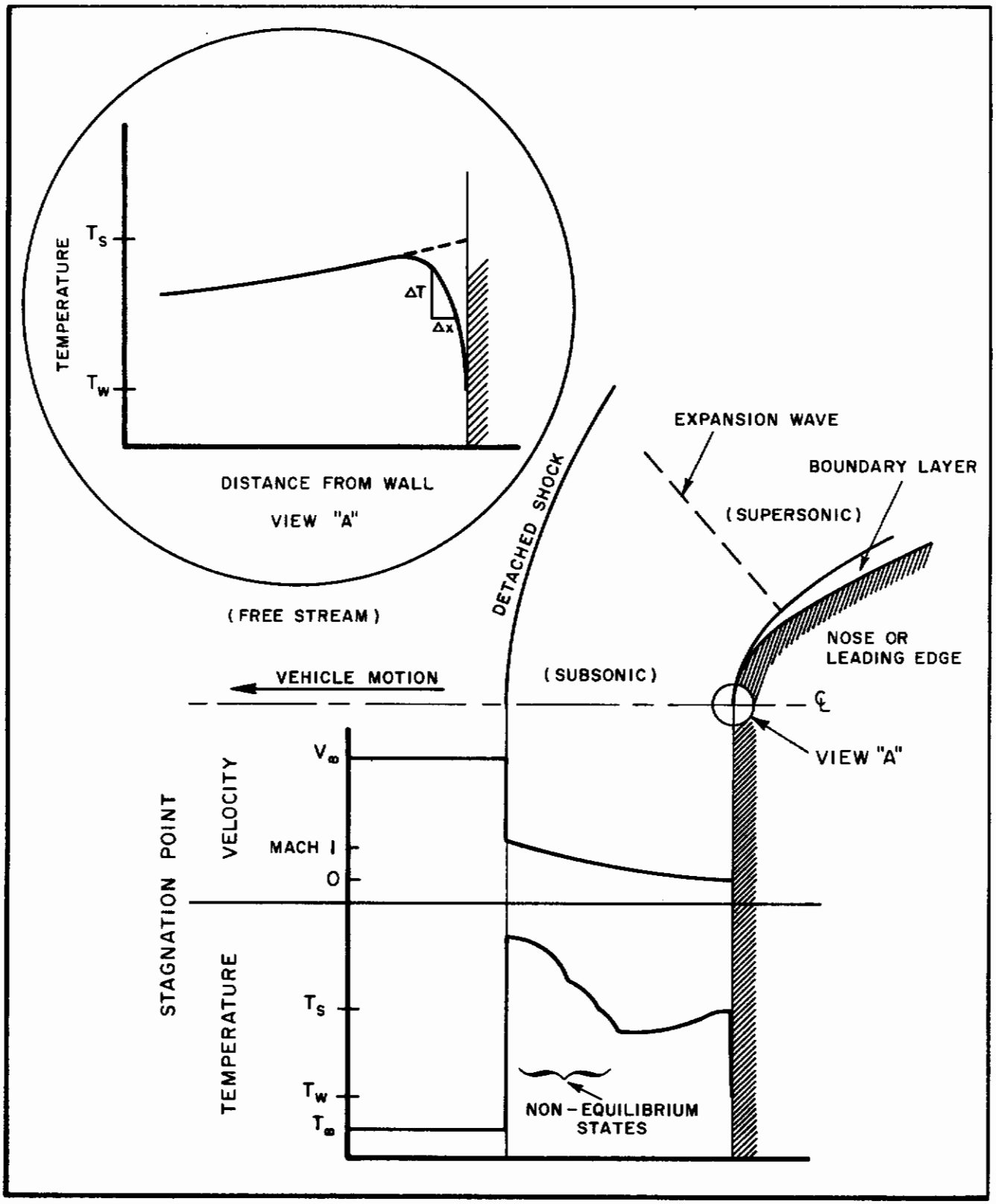
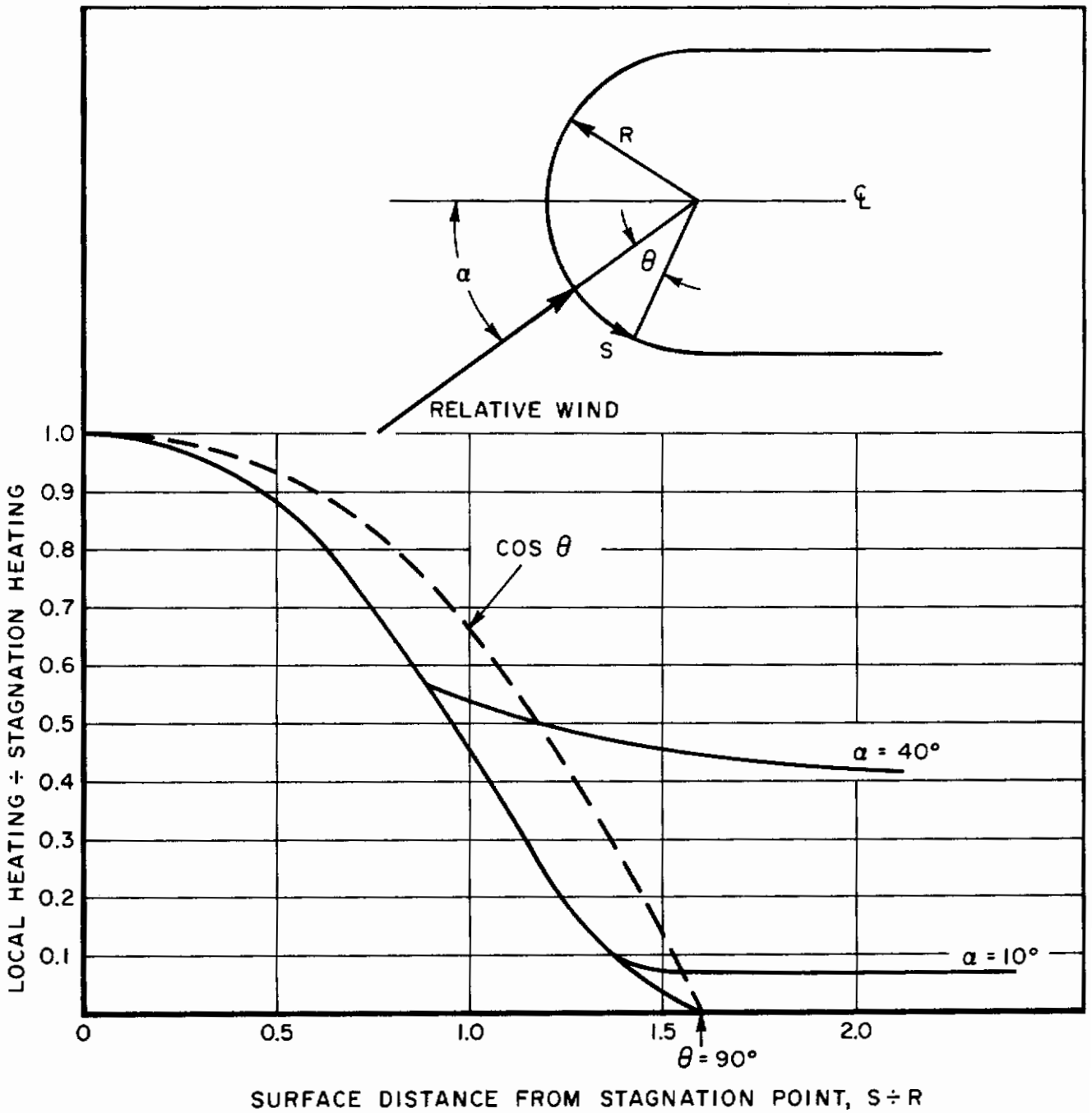


Figure 1. Stagnation Region Profiles



NOTE: THIS GRAPH SHOWS A DISTRIBUTION TYPICAL OF BOTH LEADING EDGES AND NOSE CAPS. IT IS NOT DRAWN TO SCALE.

Figure 2. Convective Heating About a Stagnation Point

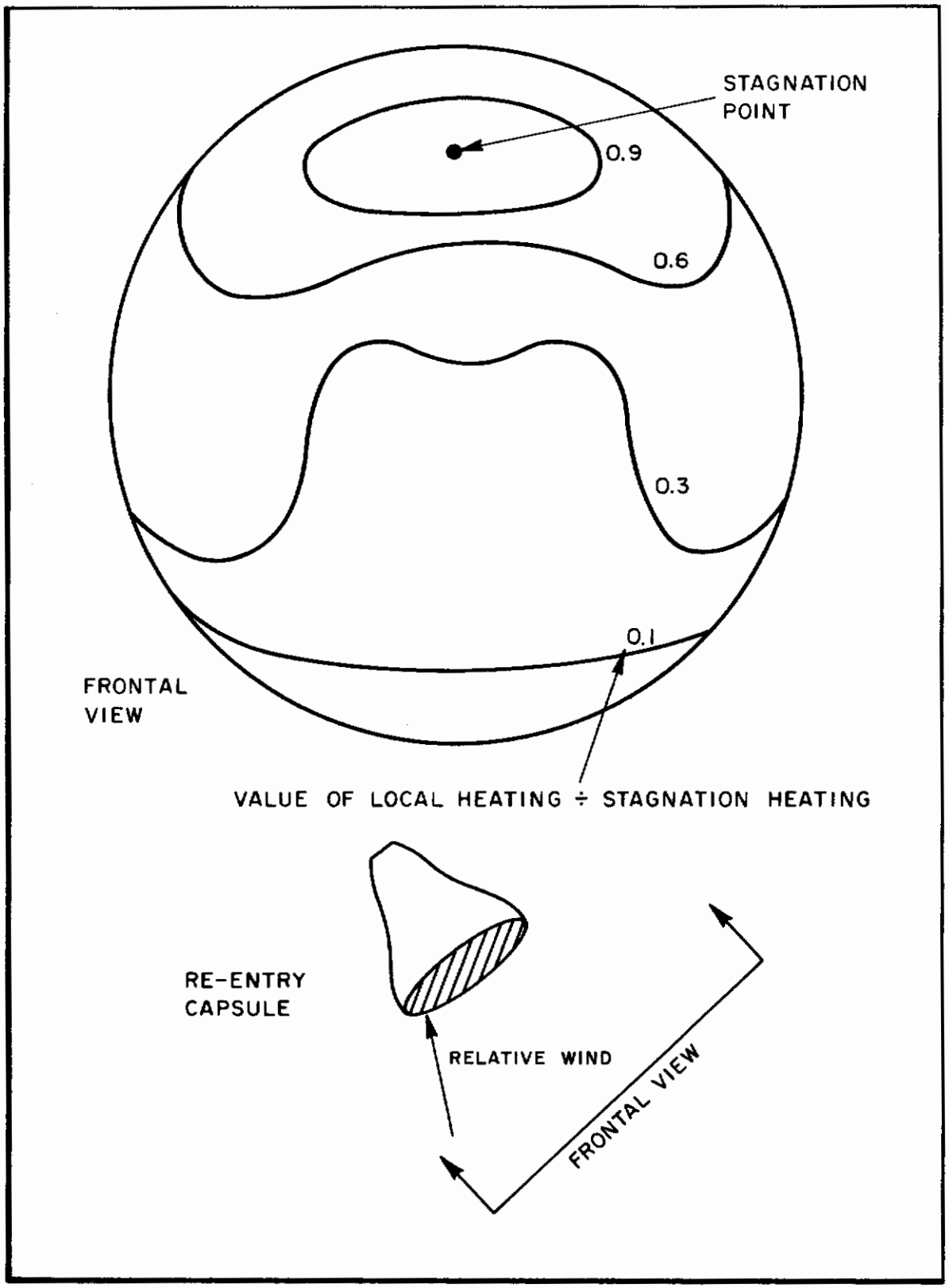


Figure 3. Thermal Contours on a Blunt Body at an Angle of Attack

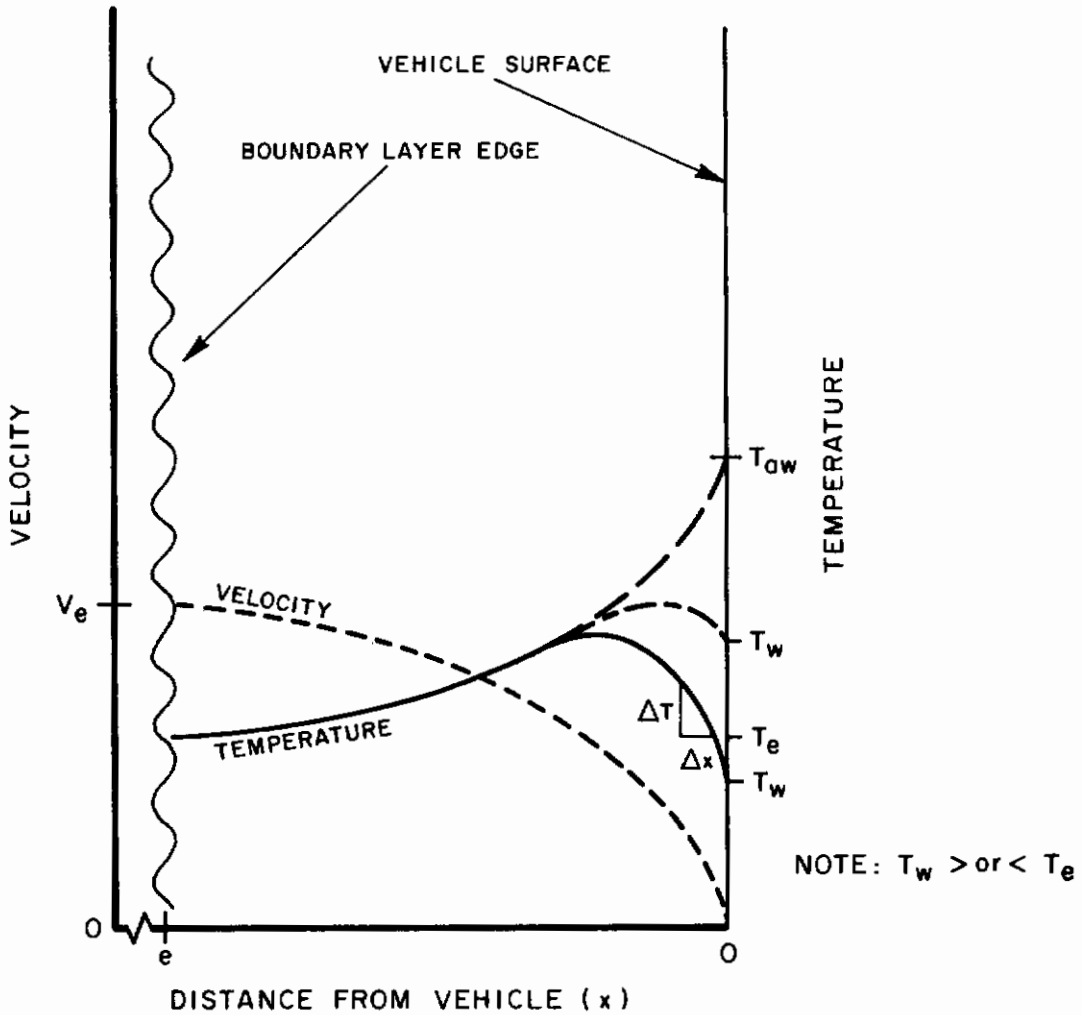


Figure 4. Thermal and Velocity Profiles in Boundary Layer

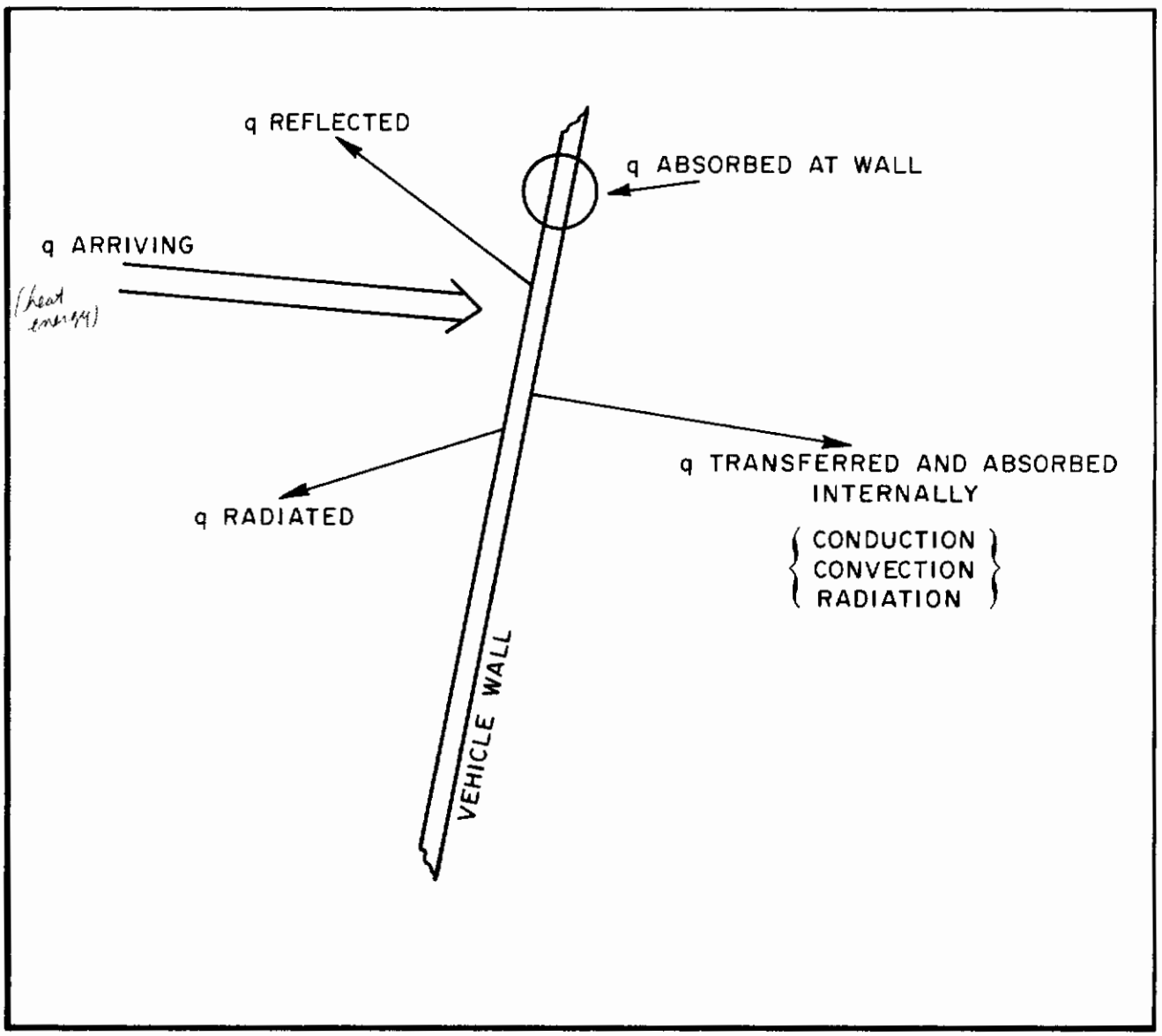


Figure 5. Energy Transfer at Inert Surface

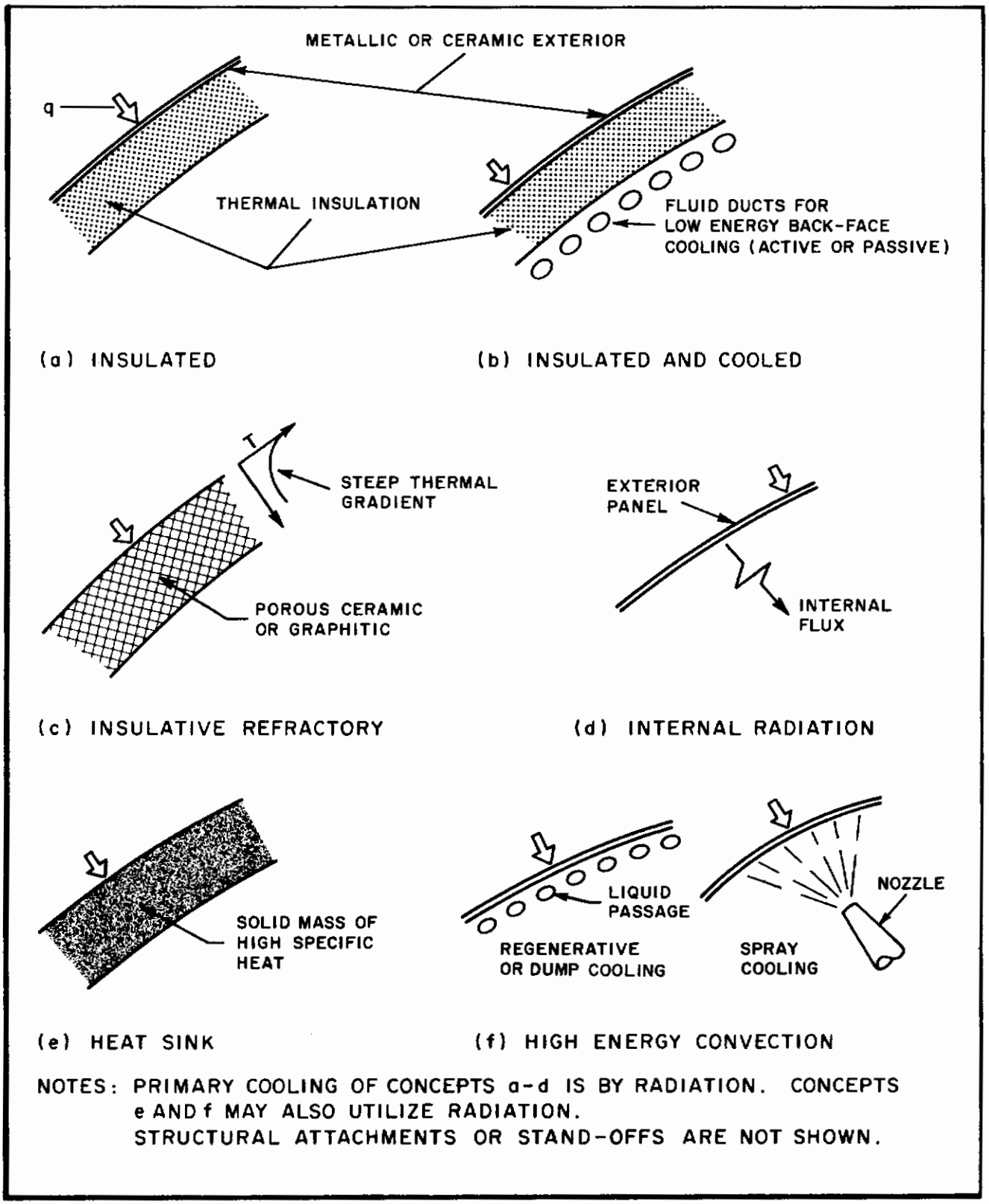


Figure 6. Heat Shield Concepts - Non Heat Blocking

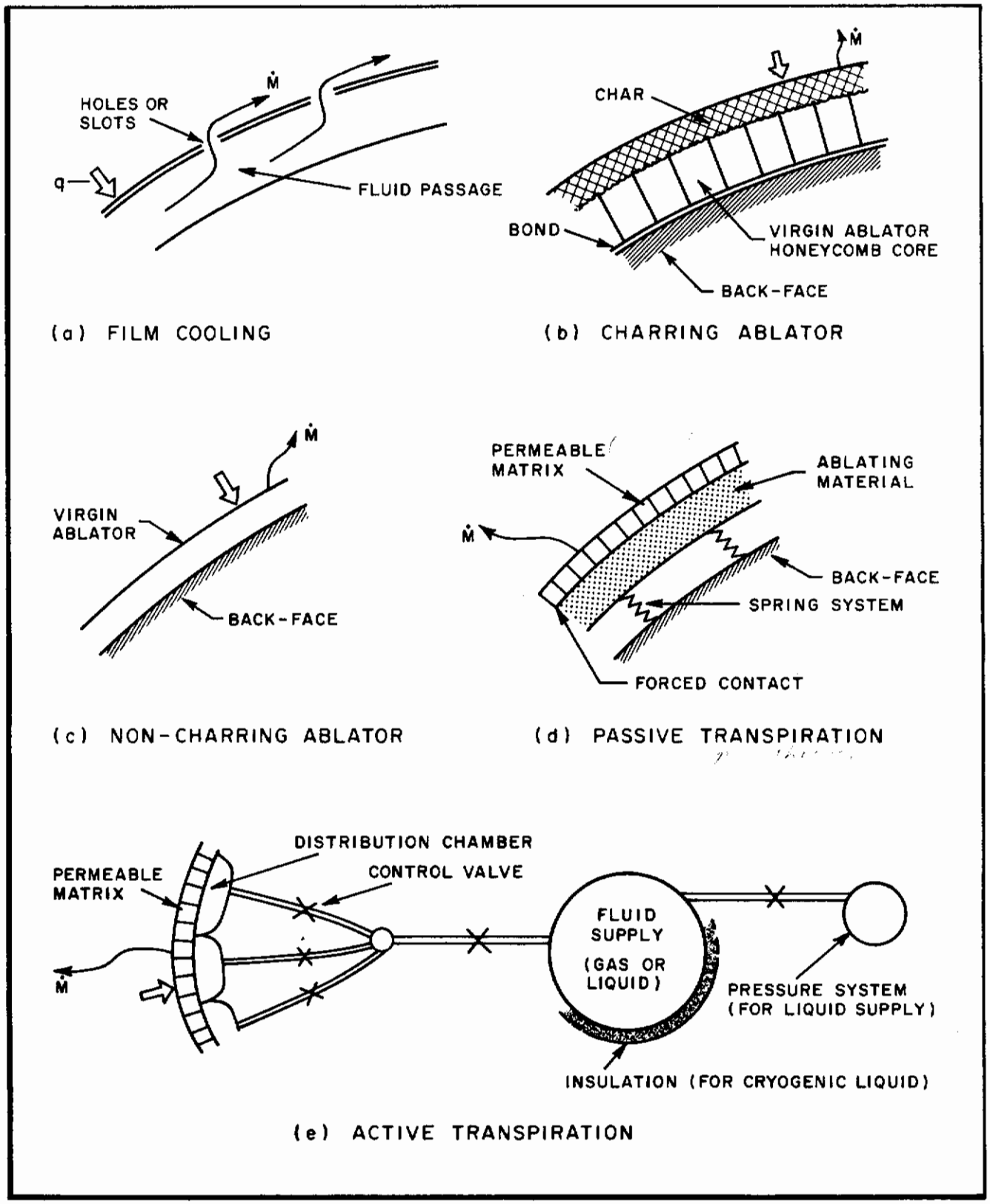


Figure 7. Heat Shield Concepts - Boundary Layer Injection

Complex Hydrocarbons ⟶

Less Complex Hydrocarbons (gas or non-gas)

Carbon Residue

Hydrogen
Nitrogen
Oxygen
Other Elements
} In ionic, atomic, and molecular (elemental or compound) states.

Inorganic Solids (fillers and reinforcement) ⟶ Gases or Liquids

SiO_2 (crystalline or vitreous) ⟶ Molten Glass & Vapor

Oxidation ⟶ CO_n, H_2O, N_xO_y, etc. (n,x, and y define molecular forms)

Carbon Solid ⟶ Carbon Gas

Endothermic Reactions

Exothermic Reactions

Latent Phase Changes

Sensible Temperature Changes

Internal Heat Storage

Internal Conduction

Internal Radiation

Internal Transpiration

Outward Radiation

Heat Blockage

Figure 8. Typical Ablator Reactions and Thermal Processes

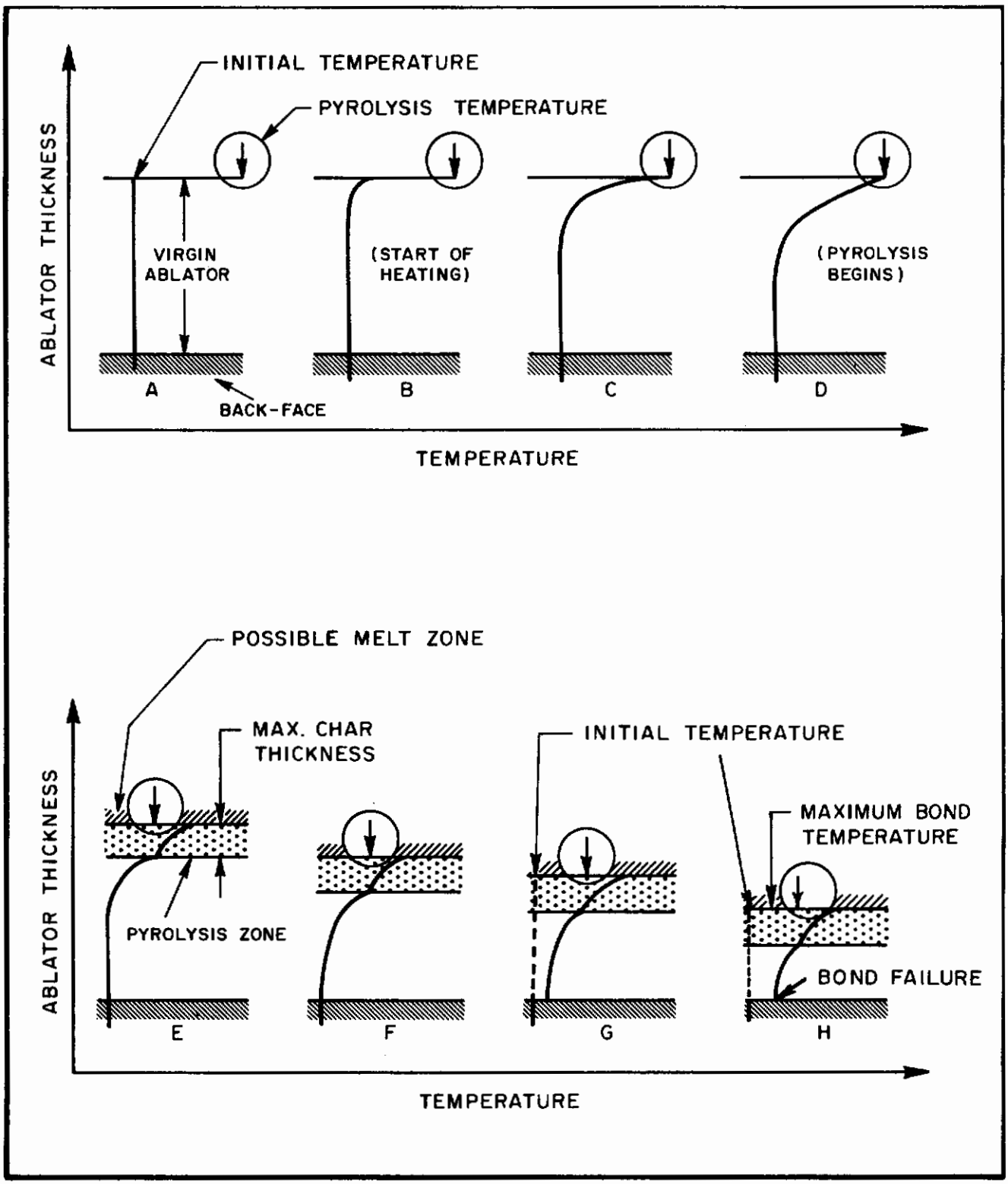


Figure 9. Ablator Thermal History - Line Pyrolysis

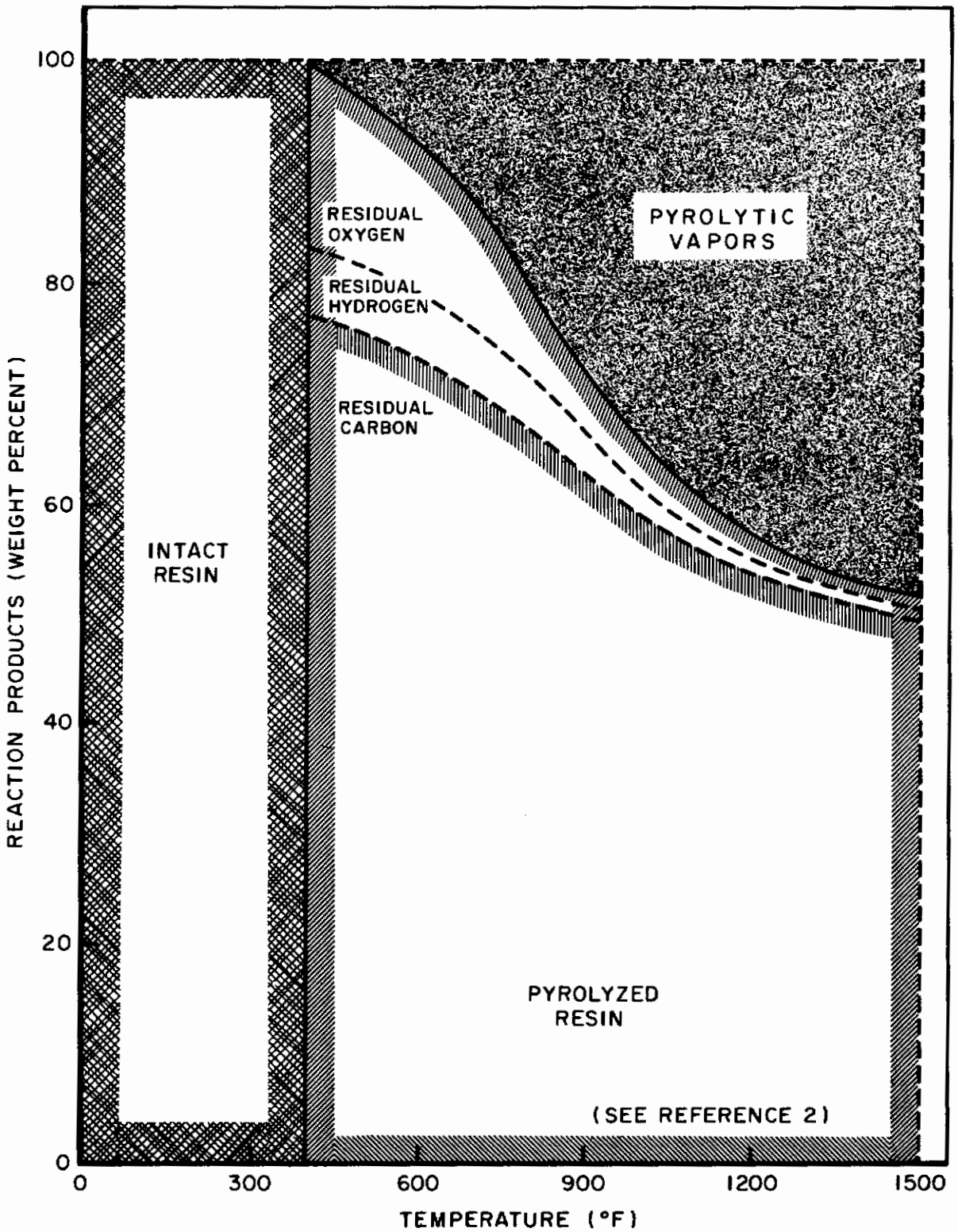
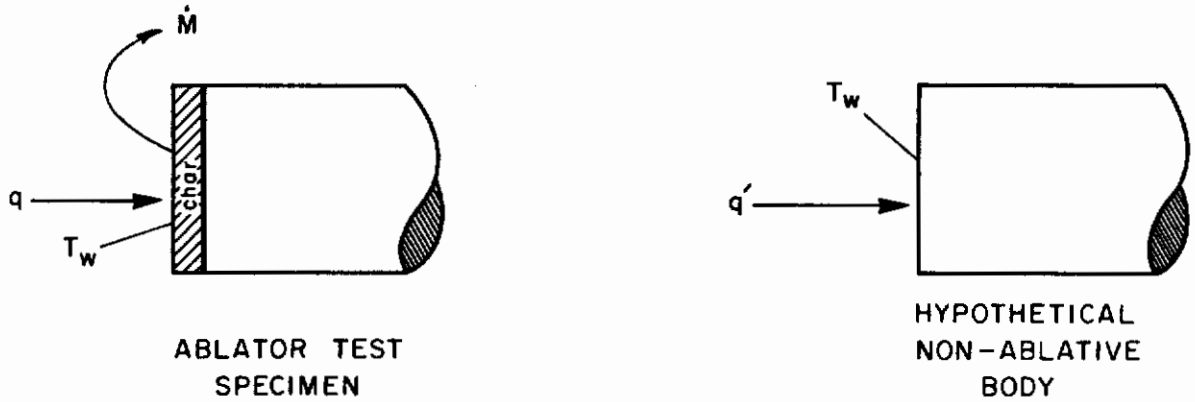


Figure 10. Typical Thermogram



$q \equiv$ Energy rate into ablator during test; may be considered as unknown, not necessary for calculation (Btu/sec).

$T_w \equiv$ Ablator surface temperature, measured by pyrometer or thermocouple (^oF).

$\dot{M} \equiv$ Mass loss rate from ablator; measured as difference in specimen weight per test duration (lb/sec).

$q' \equiv$ Steady heat rate to fictional non-ablative body operating at T_w; usually calculated from calorimetry (cold wall) data (Btu/sec). Neglecting hot gas or plasma radiation, q' can be and is expressed in two ways (often without specifying which):

$$q' = \dot{q}$$

or

$$q' = \dot{q} - q_r = \dot{q} - \sigma \epsilon_{Ablator} T_w^4$$

$\overset{*}{q} \equiv q' \div \dot{M}$, (Btu/lb)

$q_{eff} \equiv q' \div$ (total initial ablation system weight), (Btu/lb)

Figure 11. Ablator Effectiveness Parameters

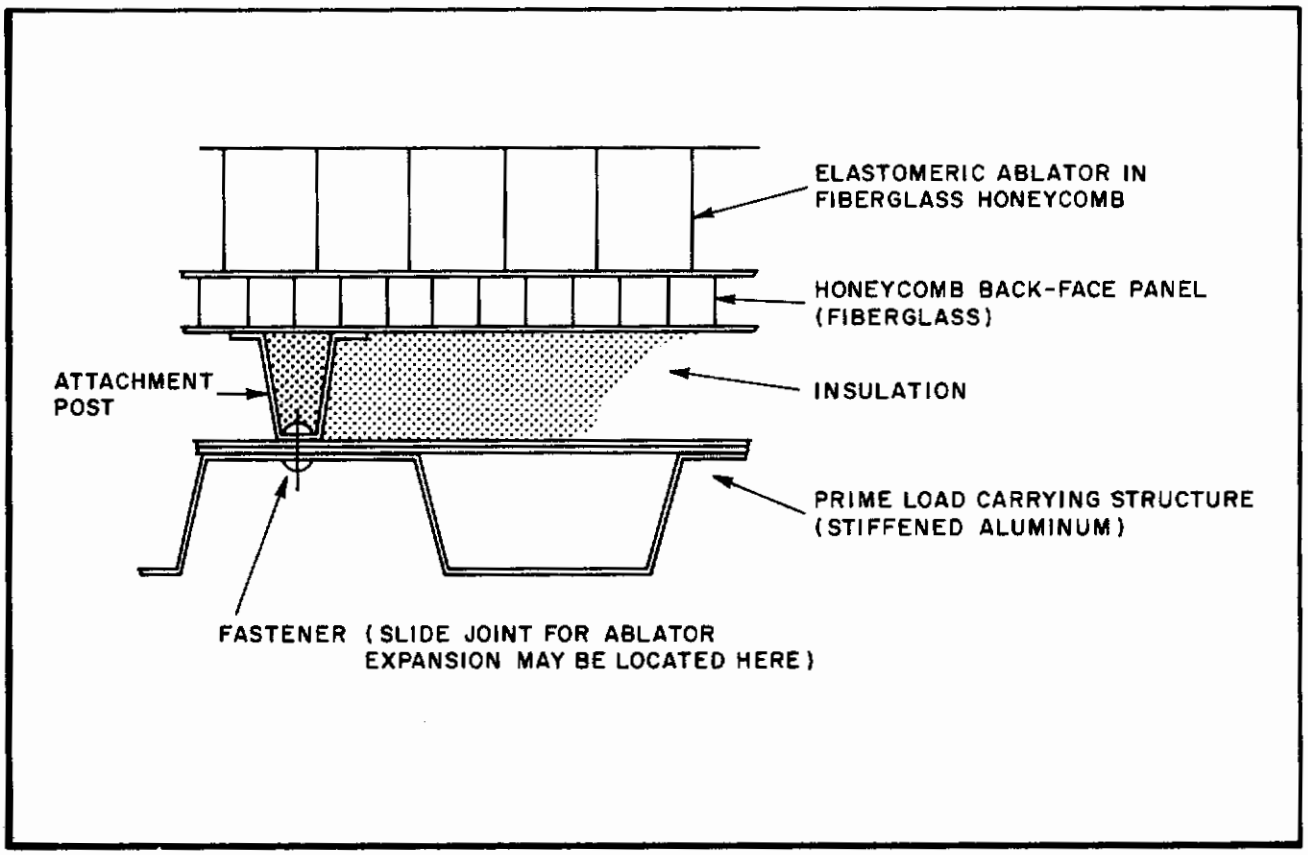


Figure 12. Typical Ablator Heat Shield

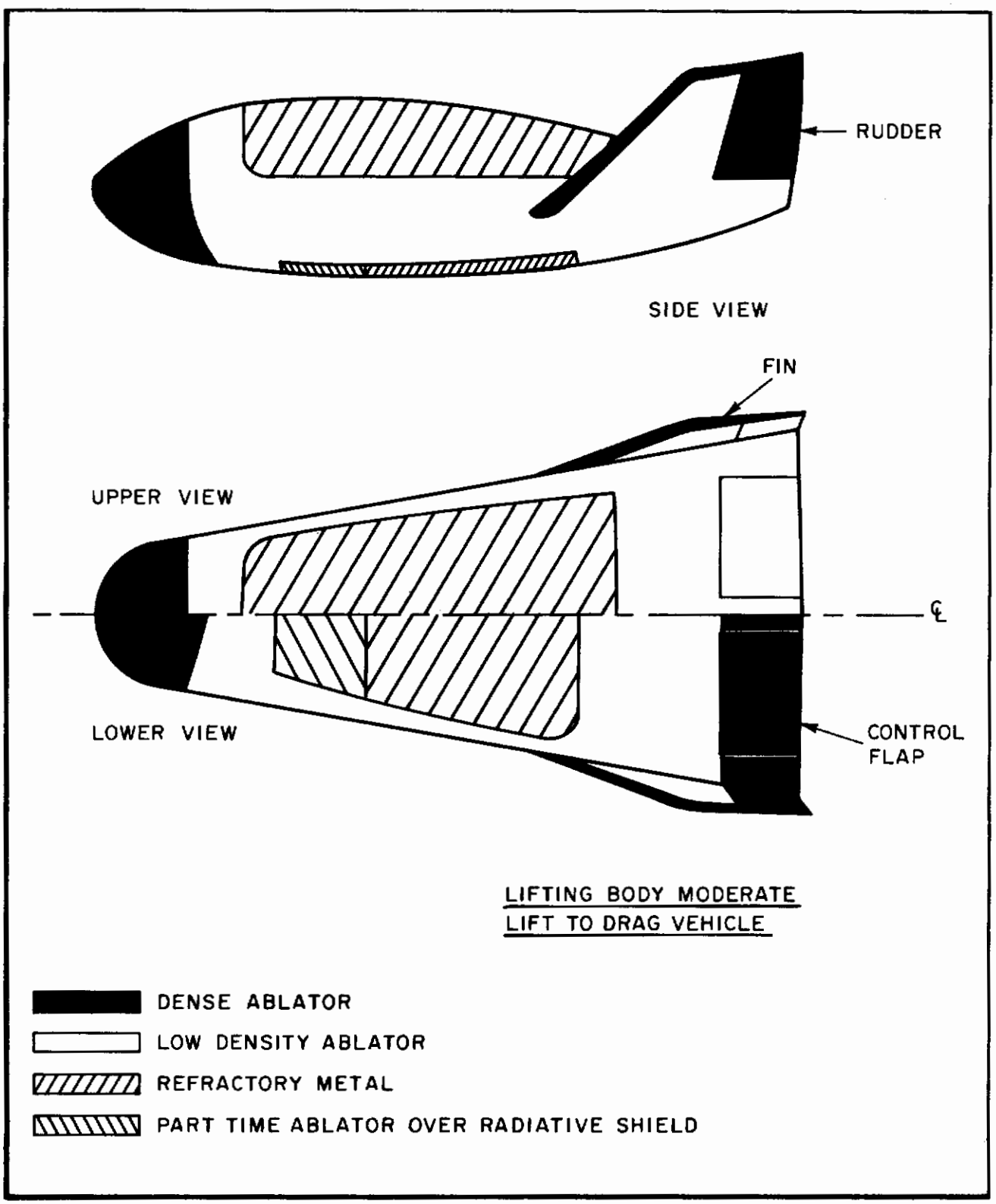


Figure 13. Possible Ablative and Radiative Heat Shield Locations

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13. ABSTRACT

A qualitative review of fundamental relationships involved in ablation engineering and aerodynamic heating is presented. This document is written for rapid reading with the primary purpose of introducing the technical area to engineers, scientists, and management personnel associated with but not intimately working in hypervelocity thermal protection.

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14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
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