von KARMAN INSTITUTE

FOR FLUID DYNAMICS

PREPRINT 1979-5

KINETIC HEATING OF HIGH SPEED MISSILES

B.E. RICHARDS

JANUARY 1979

RHODE SAINT GENESE BELGIUM

VON KARMAN INSTITUTE FOR FLUID DYNAMICS
AEROSPACE DEPARTMENT
CHAUSSEE DE WATERLOO, 72
B - 1640 RHODE SAINT GENESE, BELGIUM

PREPRINT 1979-5

KINETIC HEATING OF HIGH SPEED MISSILES

B.E. RICHARDS

JANUARY 1979

PAPER PRESENTED AT AGARD LECTURE SERIES ON
"MISSILE AERODYNAMICS" IN ANKARA, ROME AND
RHODE SAINT GENESE (BELGIUM), MARCH 1979

ART 7901 BER/NT

TABLE OF CONTENTS

KINETIC HEATING OF HIGH SPEED MISSILES

by

Bryan E. RICHARDS

Professor in Department of Aeronautics/Aerospace
von Karman Institute for Fluid Dynamics
Chaussée de Waterloo, 72
B-1640 Rhode Saint Genèse, Belgium

SUMMARY

This paper provides a review of kinetic heating as related to high speed tactical missiles. The aspects considered are : the design problem; areas in which kinetic heating is important; the prediction of heating in attached and separated flows; measurements on configurations in flight and in wind tunnels; ground simulation of flows over models; and the measurement of kinetic heating. The experience is generally obtained from studies in related areas such as high speed aircraft, reentry vehicles, etc. rather than on tactical missiles since little information appears to be available on this application due to it being a fairly recently confronted problem area. Because of the wide coverage of the field, a generous list of references is provided to guide the reader to papers giving more refined details.

LIST OF SYMBOLS

c_p	specific heat at constant pressure
g	gravity constant
Gr	Grashof number
h	heat transfer coefficient, enthalpy
k	thermal conductivity
m	pressure gradient parameter
n	power related to pressure interaction theory
Nu	Nusselt number
p_{t_2}	pitot pressure
Pr	Prandtl number
Pr_t	turbulent Prandtl number
q	heat transfer rate
r	recovery factor
R_N	nose radius
Re	Reynolds number
s	distance along wall
St	Stanton number
t	time
T	temperature
u	streamwise velocity
x	distance in streamwise direction
y	distance at right angles to wall
z	distance in a spanwise direction

α	thermal diffusivity
β	coefficient of volumetric expansion
γ	intermittency distribution
ε	emissivity, eddy viscosity
κ	turbulent conductivity
μ	viscosity
ν	kinematic viscosity
ρ	density
σ	Boltzmann constant
θ	momentum thickness

Subscripts

e	edge of boundary layer
g	gas
n	exponent
o	stagnation point
r	recovery, radiation
tr	beginning of transition
tend	end of transition
w	wall

Superscript

fluctuation quantity

1. INTRODUCTION

Kinetic heating of flight vehicles can be considered a relatively well studied problem at present. Experience already has been obtained with reentry ballistic missiles, manned space vehicles and high speed military and research vehicles. Heating due to attached flows over simple shapes can be reasonably well predicted, and it is well known that the most critical regions are at stagnation regions, on thin structures, such as wings and fins, and in the regions of reattaching shear layers caused by flow interactions. There have been, however, few publications on aerodynamic heating associated with tactical missiles since generally flight speeds and mission times have been small enough that no special design is required to combat it. For missiles with a speed range above Mach 2, when aerodynamic heating becomes significant in structural design ($T_0 \sim 260°C$ at sea level, see Fig. 1), some consideration should be given to verify whether production costs of thermally protected vehicles can be reduced and/or failures minimized by appropriate modification. It is thus to hypersonic flow studies that one must look for information useful to missile kinetic heating.

Compared with manned vehicles, which require to have a very low failure rate and a cabin environment in which humans can exist, the design of a missile to counter kinetic heating does not necessarily have to be so rigorous. However, it is desired to use inexpensive materials for the external skin whenever possible and delicate electronics, guidance systems or payload (for example most bombs and fuses have as their explosive charge some form of TNT, which melts at 80°C) may be themselves affected by heat. Furthermore, the wider manoeuvre margins allowed by the absence of a pilot is likely to increase the likelihood of incurring highly localized heating rates due to shock-surface interactions.

In most cases the mission of a missile includes a short dash to its objective at a low drag incidence, with finally rapid manoeuvres at large incidences and control angles. The result of this is that steady state heat transfer and relatively isothermal wall conditions are unlikely to exist at the critical regions mentioned earlier. Hence one general problem that should be solved is a transient heat transfer situation with anarbitrary wall temperature variation. Another critical situation is that missiles carried by high speed aircraft to their launch point may be subjected to kinetic heating for extended periods such that the skin and perhaps most of the internal parts may achieve temperatures close to the flight recovery temperature. Indeed, the performance of the aircraft could be severely limited by temperature limits of the store (Ref. 1), as illustrated in figure 2. Active cooling systems are unlikely to be used because of their complexity and cost. Protection,if needed, will come from the use of heat sink cooling, use of appropriately high temperature resistant materials, low thermoconductivity materials such as silicon rubber and from the use of low temperature ablators.

A review of this kind is likely to appeal to four types of individuals: the missile designer, the missile user, the researcher and the educator. These individuals may require different information. The designer is likely to require information at all levels of sophistication. In the preliminary design phases only time and money for simple numerical predictions will be available. At later stages of design as more capital is risked by poor design then more sophisticated numerical calculations or wind tunnel tests are likely to be used. An organization specializing in missile design over an extended time is likely to expend much effort in adapting up-to-date numerical techniques and modern instrumentation and facility techniques to produce the most optimum design. The missile user has usually an interest in less sophisticated prediction methods to enable him to specify the vehicle he wants, and to assess the product in its design, trial and commissioning phases. The researcher to be effective and useful has to understand current problems in the area in order to keep ahead of the requirement of the designer or to be able to adapt his researches to the application. The educator is interested in providing a suitable training to the future workers in the field in fundamentals and experience. The article is written as far as possible keeping all these requirements in mind, but it must be emphasized that the subject is a broad one, and hence references for more deeper research are quoted generously in order to keep the review down to a reasonable size.

In the following chapter of this review, the elements of a design calculation are outlined. Chapter 3 outlines flow situations in which kinetic heating is most likely to be a problem. Chapter 4 deals with a review of prediction methods for attached flow situations. Heat transfer involved with flow interactions are discussed in Chapter 5. Some applications to configurations are presented in Chapter 6. Finally, some remarks on wind tunnel facilities for simulating aerodynamic heating are given in Chapter 7.

2. GENERAL DESIGN PROBLEM

A general but realistic design problem that may be considered is the heat transfer to a skin of finite thickness by forced convection, conduction through the skin and radiation from its surface to the exterior gas and the interior components as well as internal free convection. The problem is illustrated in figure 3. A few remarks on ablation protection will be made at the end of this chapter. The equations can be generalized as follows (Ref. 2).

Forced convection

This is created by the viscous effects within the boundary layer through which the fluid velocity decreases from the free stream velocity at the outer edge to zero adjacent to the wall. With this zero velocity condition, heat can be transferred between the fluid and the body only by conduction. This may be represented by Fourier's heat conduction law :

$$q = \left(k_g \frac{dT}{dy}\right)_{y=0} \tag{1}$$

In practice, especially for turbulent boundary layers, to calculate the temperature gradient near the wall is complicated and hence the convection process is usually represented by :

$$q = h(T_r - T_w) \tag{2}$$

where h is a heat transfer coefficient (units watts/m^2K) for q in watts/m^2 and temperature in K) to be calculated by appropriate solution of the boundary layer equations and T_w is the wall temperature. T_r, the recovery temperature (sometimes known as the adiabatic wall temperature) is calculated from :

$$T_r = T_\infty + r \frac{u_\infty^2}{2c_p} \tag{3}$$

r is a recovery factor usually given a value of $Pr^{0.5}$ (where Pr is the Prandtl number = $\mu c_p/k$ and which has a value between 0.68-0.72 depending on temperature) for laminar flows and 0.89 or $(Pr)^{1/3}$ for turbulent flows.

Radiation from wall

The surface will be losing heat by radiation to its surroundings at a rate given by the Stefan-Boltzmann law :

$$q_r = \sigma \, \varepsilon \, T_w^4 \tag{4}$$

where ε is the surface emissivity factor (tabulated for various materials in Ref. 2) and σ has a value of 5.67×10^{-8} W/m^2K^4 to give q in W/m^2.

Radiation to wall

The surface will be receiving back heat by radiation from the surrounding air of the amount :

$$q_r = \sigma \, \varepsilon \, \varepsilon_g \, T^4 \qquad (5) \qquad (5)$$

where ε_g is the gas emissivity and T is the temperature of the radiating gas. This source of heat transfer is usually small in comparison with the forced convection heat input.

Conduction through the skin

Here one must consider the transient condition that the heat transfer is time variant, and hence the energy storage in the material must be considered in addition to heat conduction. The general equation is known as the Fourier-Poisson heat conduction equation which in rectilinear coordinates is given by :

$$\frac{\partial T}{\partial t} = \alpha \left(\frac{\partial^2 T}{\partial x^2} + \frac{\partial^2 T}{\partial y^2} + \frac{\partial^2 T}{\partial z^2}\right) \tag{6}$$

where $\alpha = k/\rho c_p$ is the thermal diffusivity of the material. This is a parabolic partial equation which requires an initial condition of the temperature distribution at time t=0 and such boundary conditions as the heat transfer rate (or surface temperature) variation with time at the two bounding surfaces over the calculation period.

The solution is considerably simplified if equilibrium conditions hence steady or quasi-steady state conditions are achieved when the Fourier heat conduction equation is applicable, i.e. :

$$q = \frac{k(T_1 - T_2)}{y} \tag{7}$$

where T_1 and T_2 are the temperatures at the two bounding surfaces. A quasi-steady state situation occurs when the time for a significant change in average wall temperature is very large when compared to the characteristic time ℓ^2/α, where ℓ is the mean thickness of the material. Reference 3 provides some standard solutions to equation 6.

Radiation to interior

The equations for this contribution are similar to equations 4 and 5.

Natural convection in interior

The fluid motion is caused by density differences. The heat transfer rate is usually determined from correlations involving the equation :

$$Nu = K\,(Gr \cdot Pr)^n \qquad (8)$$

where N is the Nusselt number $= \frac{hx}{k}$;

Gr is the Grashof number $= \frac{\Delta T x^3 \rho^2 g \beta}{u^2}$

where β is a coefficient of volumetric expansion and ΔT is a temperature difference.

All these equations should be solved simultaneously to arrive at the time history of skin or interior temperature during a mission. If quasi-steady state conditions are achieved, considerable simplification is achieved because of the change in conduction solution from equation 6 to equation 7. In many cases some of the contributions to heat transfer will be negligible and can be safely dropped from the calculation. The solution of the general heat transfer problem for typical aircraft or missile structures usually involves a finite element approach (Refs. 4, 5) involving the solution of a large number of simultaneous equations to be solved on a digital computer or analogue simulator. An approximate approach has been applied in reference 6. The transient conduction problem is out of the scope of this lecture, but the latter three references provide a preliminary guide to the subject. More sophisticated techniques and computer codes may arise from the present great interest in heat transfer in space vehicles.

This section serves to illustrate that the forced convection source of heating (in this application known as aerodynamic heating or kinetic heating) provides a primary role in determining a missile's capability of withstanding thermal effects. The normal input into a design program is the heat transfer coefficient, h, which is often given in its non-dimensional forms :

$$Nu = \frac{hx}{k} \qquad (9)$$

or

$$St = \frac{h}{\rho u c_p} \qquad (10)$$

where x is a distance from the stagnation point, k is the thermal conductivity of the gas, u the local gas velocity and c_p the specific heat of the gas. St is a statement of the proportion of energy/unit area of flow transferred to unit area of surface. Nusselt number is related to Stanton number through the Reynolds number (ratio of inertia forces to viscous forces) and the Prandtl number (ratio of "momentum diffusivity" or kinematic viscosity to thermal diffusivity, ν/α or $\mu c_p/k$) by

$$Nu = St \cdot Re \cdot Pr \qquad (11)$$

Such values can be predicted by various methods of differing complexity ranging from empirical relationships to full solution of the time-dependent three dimensional Navier-Stokes equations to be described in the following chapters.

Ablation protection will not be dealt with in this notes. Its use can be expected only for hypersonic speeds. For example, the X-15 (Ref. 7) flew at speeds up to Mach 6 without such protection. For an extension of its speed range to Mach 8, ablation materials to protect its nose and wing leading edges were examined (Ref. 8). A description of the development of the thermal protection system for the Apollo spacecraft is discussed in reference 9 and for the Space Shuttle in reference 10. An up-to-date review of ablation in cluding 119 references is given in reference 11.

3. FLOW SITUATIONS CONTROLLING HEAT TRANSFER ON A MISSILE

A series of descriptive examples will be given in this chapter to provide the background for the review and to aid the location of possible areas of high heating.

3.1 Axisymmetric body at zero incidence

A typical body is given in figure 4. The vehicle should have a blunt nose if the vehicle speed is high enough such that kinetic heating is important. As will be illustrated later, stagnation heating is inversely proportional to the square root of nose radius, and furthermore it is desired to conduct heat away from this critical region. Along the cone surfaces an axisymmetric laminar boundary layer will develop. When a particular value of Reynolds number is achieved, the flow will undergo a transition to a turbulent boundary layer, when heat transfer rates will be typically tripled above the laminar values which would have existed at that point without transition. The transition region will be quite extensive, sometimes of the order of the length of the laminar region, and the heat transfer rate will rise monotonically from the laminar to the turbulent level.

The expansion around the corner will cause a gradual drop in heat transfer rate to its cylinder value. The flare on the body could cause a sufficiently high induced pressure gradient to separate the boundary layer which can initially lower or raise the heat transfer rate whether the boundary layer is laminar or turbulent, respectively. If separation exists on the flare, there will exist locally an intense increase in heat transfer rate just downstream of flow reattachment to a value many times that of the undisturbed flow and sometimes even greater than the stagnation point value. The flow will expand and develop in a similar way as past the shoulder, and finally expand around the base. Some interaction with the engine exhaust plume may cause locally high heat transfer rates on the base.

3.2 Body at angle of attack

In this case, the boundary layer is three dimensional providing more complication than axisymmetric flows. A stagnation line will occur along the windward generator and there exists a cross flow at the body side surfaces with an angle greater than the incidence of the body to the stream. A characteristic of the three dimensional boundary layer is that the direction of the shear stress at the surface is different from that of the flow at the edge of the boundary layer, with the result that the profile is skewed. Depending on the degree of skewness determines whether the profile in the outer flow streamwise direction can be considered quasi-two dimensional in the boundary layer solutions. With the change to an adverse pressure gradient, as the flow passes around the body, the flow is likely to separate, flow aligning shocks are formed and the flow reattaches giving a vortex flow behaviour. Lines of high heat transfer rate are formed in the region of flow reattachment lines. The level of heat transfer rate here does not usually exceed typical stagnation point values.

3.3 Component interactions (illustrated in figure 5)

Aerodynamic, flap and jet control and other protuberances will all have the effect of causing separation, hence high energy shear layer reattachment, the cause of localized regions of high heat transfer rates. Such flows may be subject to spanwise non-uniformities of the oncoming stream as well as finite span effects.

3.4 Wing-body interactions (figures 5, 6)

At zero incidences, corner boundary layer flows will cause minor localized high heat transfer rates near the corner. Much more severe problems occur with wings or tails at incidence in the presence of a body.

On the windward surface of such protuberances, a three dimensional separated region occurs and a region of very high heat transfer rate is found on the body near the wing body junction. On the leeward surface of the wing, if it is reasonably highly swept, leading edge vortex flow will occur with the reattaching shear layer likely to cause high heat transfer rates on the body along a line centered near the wing root at its leading edge. The surfaces within gaps between all moving control surfaces also are likely to encounter high heating rates.

3.5 Shock interactions (illustrated in figure 7)

Shocks caused by the nose on aerofoil leading edges or by accompanying bodies such as boosters engines or stores on a body, wing or pylon are also the cause of severe heating problems. If the shock is directed towards a stagnation region, many types of interaction can occur, some of which cause very high local heat transfer rates.

4. ATTACHED FLOW HEAT TRANSFER

In this section we assume that the flow can be dealt with by using the thin boundary layer approach with boundary conditions defined by inviscid flow solutions. Again, these latter solutions can be obtained at various degrees of complication and have been dealt with in other parts of the Lecture Series. Flows over simple configurations can be solved using the gasdynamic isentropic relations, flow across normal or oblique shocks, Prandtl-Meyer expansion, Newtonian and modified Newtonian theories, tangent cone and tangent wedge methods (see for example Refs. 12, 13). More complicated solutions of the inviscid supersonic flow relations involve the method of characteristics (Ref. 13),panel methods (Ref. 14) and the most up-to-date are the modern successice line over-relaxation techniques with fast solvers (NASA Ames Illiac Group) etc. Turning to the viscous part of the solution which involves the solution of the compressible laminar and turbulent boundary layer equations (in their differential or more simple integral forms), a good overall review (emphasizing turbulent flows,however) appears in a book by Cebeci and Smith (Ref. 15) of the McDonnell Douglas Aircraft Company suitable for application to aircraft and missiles. The full equations are presented and discussed in this book. Although the subject of heat transport is introduced, however, the contents of this book little discusses applications in heat transfer.

The solution of the boundary layer equations are usually approached in many ways involving different degrees of complication : firstly, "simple" methods involve correlation methods, integral methods and transformations to the inviscid equations, secondly, numerical solutions of the partial differential equations. Suitable available solutions will be briefly reviewed in the following subsections by using the state of the boundary layer as the primary classification. The simple correlation methods mentioned are presented in references 2 and 16. Many such simple methods are being used in the design of the space shuttle orbiter showing that this approach should not be scorned (Ref. 17).

4.1 Stagnation regions

A summary of early work on stagnation point heating on spherical and cylindrical shapes was made by Fay, Riddell and Kemp (Ref. 18). The most widely used predictions are those developed from the expressions of Fay and Riddell (Ref. 19) for these cases. Without flow dissociation, the following formula is used

$$q_0 = 0.763\ Pr^{-0.6}\ (\rho_e u_e)^{0.5}\ \left(\frac{\rho_w \mu_w}{\rho_e \mu_e}\right)^{0.1} \left(\frac{du_e}{ds}\right)_0^{1/2} [h_e - h_w] \qquad (12)$$

where ρ, u, h, μ are density, velocity, enthalpy and viscosity and subscripts w, e, o mean at the wall, at the outer edge of the boundary layer and at the stagnation point and s is the distance from the stagnation point. Using modified Newtonian theory, the velocity gradient at the stagnation point of a sphere or cylinder with radius, R_N, is

$$\left(\frac{du_e}{ds}\right)_0 = \frac{1}{R_N}\sqrt{\frac{2p_{t2}}{\rho_e}} \qquad (13)$$

where R_N is the sphere radius and p_{t2} is the stagnation pressure.

The heat transfer on the stagnation lines of yawed cylinders (representing leading edges of swept wings or bodies at high angles of attack) is discussed in reference 20. At large free stream Mach numbers, the effect of yaw is essentially to correct the unyawed value by $(\cos\Lambda)^{1.1}$ where Λ is the sweep angle. At high enough Reynolds numbers and angles of sweep from 40° to 60°, it was found that the boundary layer on a swept cylinder was completely turbulent even at the stagnation line.

4.2 Laminar boundary layer

A first order approximation concerns considering a surface as part of a flat plate. Eckert (Ref. 21) amongst others has shown that for a wide range of Mach numbers and temperatures, a close approximation is obtained for the skin friction coefficient on a flat plate in laminar (and turbulent) flow, by using the incompressible value evaluated at a temperature according to an intermediate (or reference) enthalpy (or temperature for a perfect gas). Application of a Reynolds analogy factor, relating the Stanton number to the skin friction is then used. A value of $Pr^{-0.5}$, equal to 1.18 for Pr = 0.72 as ascribed to Colburn is usually used for this factor.

Heat transfer over more complicated axisymmetric shapes may be dealt with the similarity analysis of Lees (Ref. 22). More sophistication and probably adequate accuracy may be obtained by the relatively easy application of the Cohen and Reshotko (Ref. 23) method based on exact solutions for specified pressure gradients and heat transfer.

An efficient numerical scheme for both two dimensional and axisymmetric flows is presented by Cebeci and Smith (Ref. 15) based on the Keller-Box computational technique. In their book, however, they provide little evidence as to its accuracy in predicting heat transfer.

4.3 Transition

Transition location usually provides the largest uncertainty in thermal protection design systems (see for example Ref. 10 concerning the Space Shuttle design). This may not be so critically important in high supersonic speed missiles since the chief problem areas are likely to be on stagnation regions and regions of impinging shear layers. There have been numerous correlations of transition location.

For a first order application, it can generally be assumed that transition will commence in the range of Reynolds numbers based on distance from the leading edge, Re_x, of around 5×10^5. A more convenient parameter to use is one based on a local boundary layer parameter such as Re_θ. Use of such a parameter also reflects the reluctance of a boundary layer to undergo transition in a favourable pressure gradient. A value of $Re_{\theta,tr}$ = 360 is a universally used value (corresponding to an Re_x of 3×10^5 on a flat plate, with zero pressure gradient).

References 15, 24, 25 give formulae which relate the transition momentum thickness Reynolds number, $Re_{\theta,tr}$, with the transition x-Reynolds number and which claim to take into account pressure gradient. For transition on airfoils, Crabtree (Ref. 26) correlated data with $Re_{\theta,tr}$ against a pressure gradient parameter $m = -\frac{\theta^2}{\nu}\frac{du}{dx}$ and found that appropriate data collapsed on this line. These methods relate, however, to incompressible flow. Hopkins et al. (Ref. 27) prepared some useful charts for estimating boundary layer transition in supersonic flows. This report reflects the number of parameters important in describing compressible boundary layer transition. Benek and High (Ref. 28) have developed a transition point prediction technique for compressible flows incorporating free stream disturbances and weak pressure gradients which has met with reasonable success for simple configurations. All these reports provide simple criteria for a very complicated and truly unsolved problem area. The most recent review of the subject by Morkovin (Ref. 29) accentuates this point.

Transition does not occur instantaneously but usually over a considerable length, which for high speed flows can be of the same order as that over which the laminar boundary layer develops. Dhawan and Narashima (Ref. 30) demonstrate the particular behaviour of the mean velocity profiles within a transitional boundary layer and have provided an empirical correlation concerning the length of transition. Masaki and Yakura (Ref. 31) have shown that the length of transition over a large Mach number range of conditions can be reasonably approximated to the length of laminar flow. Chen and Tyson (Ref. 32) have developed a correlation for transition occurring on blunt bodies for both compressible and incompressible flow.

The heat transfer distribution in a transition region is often assumed to follow a linear variation joining the laminar value at the beginning of transition to the turbulent value at the end of transition. Slightly more sophistication is achieved by incorporating the physics of turbulent spot development in a simple fashion by "sharing" the laminar and heat transfer rates using an intermittency distribution similar to that of reference 30, i.e. :

$$\gamma(x) = 1 - \exp\left[-0.412\{(x-x_{tr})/\Delta x\}^2\right] \tag{14}$$

where $\Delta x = (x_{tend}-x_{tr})/2.96$, x_{tr} and x_{tend} are the position of the beginning and end of transition.
For precise prediction of transition heating, note should be made of the so called "precursor effects" at high Mach numbers (Ref. 33).

Some more sophisticated methods have been used to calculate the boundary layer through transition. Harris (Ref. 34) developed a method for high Mach numbers cases. This requires, however, empirical information concerning transition location, transition length and an intermittency distribution to modify the eddy viscosity distribution through the transitional boundary layer. A similar approach has been developed by Adams (Ref. 35). The Cebeci-Smith (Ref. 15) method is in a more practical state for predicting missile flows, but requires similar empirical techniques.

Shamroth and McDonald (Ref. 36), however, have used a different approach in developing a practical method for computing the development of the boundary layer flow during transition which also takes into account boundary conditions such as free stream turbulence and wall roughness. The method also copes with relaminarization.

4.4 Turbulent boundary layer

As in the case of laminar boundary layers, a first order approximation of calculating the heat transfer rate on a surface is in assuming that it is a flat plate with locally a zero pressure gradient. An appropriate reference enthalpy method to obtain skin friction is that of Eckert (Ref. 21) and Sommer and Short (Ref. 37). Spalding and Chi (Ref. 38) have presented a summary of early correlation methods for prediction of skin friction for compressible flow up to 1964 and developed an appropriate semi-empirical method. A more recent method is that of White and Cristophe (Ref. 39).

Care must be taken to apply an appropriate value of the Reynolds analogy factor to be used to assess heat transfer information (through the Stanton number) from the predicted skin friction coefficients. Hopkins and Inouye (Ref. 40), in providing the latest of many survey papers on correlation methods for predicting compressible boundary layers have indicated that the Van Driest method (Ref. 41) with a Reynolds analogy factor of 1.0, Spalding and Chi (Ref. 38) and Eckert (Ref. 20) with a von Karman Reynolds analogy factor (see Ref. 42) give the best predictions of heat transfer rate. The virtual origin of the turbulent boundary layer used to define the flow Reynolds number is normally taken near the end of transition.

Some scope for improvement over a wider range of wall to recovery temperature ratios may arise from application of Bradshaw's "Van Driest III" method (Ref. 43) of predicting skin friction which is based on a more generalized velocity profile law. However, at present no application of this to heat transfer prediction has yet been perceived in the literature. A value of Reynolds analogy factor near 1.1 appears to provide a suitable mean of values. Two examples of comparisons of Eckert's (Ref. 21) method with data from flight are shown in figures 8, 9 taken from the Bushnell et al article (Ref. 44). All of these preceding methods are developed for zero pressure gradient, but usually appear to give reasonable predictions of flows with modest pressure gradients.

Integral methods solve the von Karman integral momentum equation along with various auxiliary equations. These methods provide reasonably accurate predictions for a wide range of flow conditions except under severe pressure gradients, non equilibrium boundary layer conditions and to flows where the empirical correlations used within do not apply. The methods provide fast and inexpensive calculations, but have their limitations. A recent review of these is given in reference 44.

Significant advances are being made in solving the full boundary layer equations and are being incorporated in design codes (Refs. 44, 45, 46). Once this sophistication is considered for missile of aircraft applications, the full three dimensional equations should be taken into account. An optimistic review of current research programs given in references 44 and 47 indicate that with increasing capabilities of digital computer systems and the maturing of three dimensional inviscid flow field codes, general purpose three dimensional boundary layer codes (i.e., attached flow cases) will become available in the next two or three year period. Designers should keep themselves up-to-date on this progress.

An example of such full solutions is the mean field closure methods. There are usually four closure assumptions which must be specified before a solution can be obtained. Three of them are those familiar in incompressible solutions to define the Reynolds stresses through the eddy viscosity or mixing length, i.e., the wall damping region, law of the wall region and wake or outer region. The fourth involves dealing with the Reynolds heating term $\overline{v'h'}$, which is generally handled through the use of a "turbulent conductivity" :

$$\overline{v'h'} = \kappa \frac{\partial \bar{h}}{\partial y} \tag{15}$$

which then allows the definition of a "turbulent Prandtl number" :

$$Pr_t \equiv \frac{\varepsilon}{\kappa} \quad \text{or} \quad \kappa = \frac{\varepsilon}{Pr_t} \tag{16}$$

Once Pr_t is known, the model for Reynolds stress can be used to determine κ. There are several codes available that are applicable to the kinetic heating problem and which are well documented. Examples are the method of Bradshaw, Ferris and Atwell (updated and put in Fortran IV language in Ref. 48), Stan5 (explained in detail in Ref. 49) based on the Spalding and Patankar numerical scheme.

5. VISCOUS INTERACTIONS

The most severe heating problems of high Mach number flight are associated with viscous-inviscid interactions. Complicated geometries expected in praxtical configurations can cause regions of high compression causing the flow to separate. High values of heating are then obtained near the flow reattachment positions. Such "hot-spots" can lead to heat transfer rates sometimes considerably higher than the stagnation region values. Normally, for practical vehicles at cruise or design flight conditions, the interactions will occur in regions of turbulent flow. An excellent survey on such interactions is given by Korkegi (Ref. 50).

Prediction of even the simplest of these flows is extremely difficult. The type of flow field is categorized according to the order of magnitude of a characteristic Reynolds number. If the Reynolds number is large, the flow is made up of regions which, to a great extent, can be described by Euler's equations for inviscid flow, or by Prandtl's equations for boundary layer flow. However, the full Navier-Stokes equations usually are needed for the description of the flow in local zones where the above approximations fail; the existence of such zones tends to be the rule rather than the exception in practical missile flow fields. These regions in which the Navier-Stokes equations are needed, although small,often have an important influence over the complete flow and they are an essential feature of the flow field for the determination of the overall solution. Some classical examples are flow in the region of a wing trailing edge, shock-boundary layer interactions, boundary layer flow past a corner, base flow, separation over a compression corner, etc.

Large Reynolds number flows are likely to be turbulent and this is a feature of practical missile flows. The Navier-Stokes equations remain valid for turbulent flows, but the numerical prediction of such flows with existing methods and computers cannot deal with the small scale three dimensional turbulent fluctuations. The result is that the time averaged Navier-Stokes equations are employed together with additional relations or differential equations for the various correlation terms in order to close the system of averaged equations. These additional equations, which constitute a turbulence model, are necessarily highly empirical. Hence, the usefulness of a calculation involving turbulence is linked strongly to the validity of turbulence model used.

Solutions are being accomplished for simple flows at present, but full solutions of flow over configurations at realistic Reynolds numbers at a design level are not envisaged for at least 10 years (Ref. 47). The situation at present is that much reliance on prediction of kinetic heating in regions of flow interaction is based on empirical analysis of experimental information. Such methods are likely to persist for the far future, not only to provide information to designers but also to computer program developers. It is for this reason that later on in this review, importance is given to reviewing existing and future wind tunnel projects.

Each of the types of interaction mentioned in chapter 3 are discussed in the following subsections.

5.1 Compression corners and shock interactions

Extensive measurements of supersonic laminar, transitional and turbulent boundary layer separation over various configurations including a compression corner were made by Chapman, Kuehn and Larsen (Ref. 51). They clearly showed the strong influence of transition and the much larger pressure gradients associated with turbulent than with laminar separation. A review of heat transfer in separated and reattached flows was presented by Fletcher, Briggs and Page (Ref. 52) in 1970. A more recent and rather complete review on the overall problems of two dimensional laminar and turbulent shock wave-boundary layer interactions in high speed flows has been made in 1975 by Hankey and Holden (Ref. 53). For a designer, the most important information of interest is the shock strength (impinging or induced by a wedge) required to achieve incipient separation, the length of the separated region and angle of the boundary shear layer for a particular shock strength and the heating level within the separated region and at or near the reattachment point for particular flow conditions. Correlations of some sort are available for each of these parameters, but however, only for simplified geometrical situations such as two dimensional flat plate flow or axisymmetric flow and generally only for fully laminar and fully turbulent conditions. Appropriate correlative graphs for most of the above parameters are presented in Hankey and Holden (Ref. 53).

One of the most important pieces of information concerns the maximum heating level in the separated region, generally occurring just downstream of flow reattachment. For both laminar and turbulent flows, experimentalists have related this parameter to the pressure distribution through a power law relation such as

$$\frac{q_{max}}{q_o} = \left(\frac{p_{max}}{p_o}\right)^n \qquad (17)$$

where subscript o denotes undisturbed values. The mean value of n is 0.7 for laminar flows and 0.85 for turbulent flows. The peak pressure is an easier parameter to assess than heat transfer rate (Refs. 54, 55, 56).

For more complete solutions one must look to the development of numerical schemes for solving two dimensional or axisymmetric boundary layers. Some success has been achieved by the traditional technique of dealing with the viscous and inviscid flow fields separately and matching them appropriately, but as mentioned earlier some attention has to be given to the "elliptic" characteristics of the equations in the reverse flow separated regions within the "parabolic" boundary layer region. Hankey and Holden (Ref. 53) provide a review to developments in this area as well as in the solutions of the full Navier-Stokes solutions, which eventually must be solved for realistic flow conditions.

5.2 Wing-body and fin-body interactions

In supersonic flow, the interaction of the strong bow shock of a blunt fin or wing with laminar and turbulent boundary layers causes widespread upstream and lateral flow separation. This topic has been the object of concentrated research recently by groups at AFFDL, Princeton University, College of Aeronautics at Cranfield and VKI. The flow is characterized by a lambda shock structure in the plane of symmetry, which spreads out laterally downstream to form a broad three dimensional interaction region with strong vortical type flow. Local regions of high heating resulting from the fin interaction appear near the base and extend downstream at a fixed angle to the fin (Fig. 10).

These high heat transfer rates have been related to flow reattachment or impingement along a line associated with a vortex type flow (Ref. 50). Another explanation lies in the sweeping away laterally of the low energy flow in the lower boundary layer illustrated in figure 11 by the strong transverse pressure gradient thus thinning the shear layer near the fin (Ref. 57). Again the maximum heating has been related to the maximum presssure through a power law in the same way as for the two dimensional case (eq. 17)but the average value of n in this case seems to be closer to 0.8. This demonstrates that much higher heating rates are found in the three dimensional case than in the equivalent two dimensional case for the same pressure rise. Some experiments carried out at AFFDL (Ref. 58) indicated that the maximum heating was reduced by tripping the boundary layer with a row of roughness elements. This has been checked more thoroughly in some experiments at M=5.4 at VKI in an as yet unpublished work, but not found in some later AFFDL work (Ref. 59). The mechanism for this phenomenon is yet unexplained. There abounds much more information on the three dimensional flow field and surface pressure field than on heat transfer measurements for this configuration.

5.3 Axial corner flow

Supersonic flow in axial corners formed by two intersecting flat surfaces has been the subject of experimental investigations for a decade and a half. A corner interaction occurs as a result of initial flow compression on one or both surfaces due either to boundary layer displacement effects or to the inclination of the surfaces with respect to the free stream or both. The flow field is quite complex as first found by Charwat and Redekopp (Ref. 60) as seen in figure 6.

The corner problem may be viewed as one involving a strongly disturbed inviscid flow field, and, in turn, the interaction of this flow field with surface boundary layers. Sharp peaks seen have been associated with the strong vortex or vortices attached to the corner. The corner heat transfer distribution has distinct features in common with two dimensional shock wave boundary layer interactions - a drop in heat transfer rates beyond separation (the trough) followed by a rise to high values (peaks) at reattachment. (See Fig. 12 taken from Ref. 61). Studies on corners with angles between surfaces different than 90° have also been made as reviewed in reference 50. Summarizing, regions of high heat transfer are obtained due to corner interactions, these are typically several times larger than undisturbed values, but not usually as high as stagnation point heating values.

5.4 Base flows and vortex flows on expansion surfaces

Vortex flows as may arise on the lee -side of a fuselage or wing-body at angle of attack, causing possible regions of high heating and interference with control surfaces, have been well documented by Rainbird (Ref. 62) and Peake, Rainbird and Atraghji (Ref. 63). This problem of heating was brought into prominence with the space shuttle as reviewed in Ref. 17. Again heat transfer rates several time larger than undisturbed values are obtained around the centerline of wings alone, or on the body sides above the wing centered about the wing leading edge root. Furthermore, heat transfer rates do not usually achieve values typical of stagnation regions, but under certain circumstances may be considered important. A more recent review of the subject related to space shuttle is given in reference 64 with emphasis on the effects of surface temperature and Reynolds number on leeward shuttle heating.

Another review by Bertin (Ref. 65) focusses on the definition of the flow field in the base region and in the leeward separated flow behind a vehicle at high angle of attack (e.g., space shuttle) as an aid to determining the convective heat transfer distribution. Generally, the heat transfer due to the axisymmetric flow in the base region of a body at zero angle of attack are relatively small, however, there is a large change due to the onset of transition. Smaller effects due to Mach number, mass addition and wall temperature are also discussed. The free vortex layer type separation causes much higher heating, as noted above, which appears to be connected to the thinning of the viscous shear layer as a result of the outflow caused by the vortices. Again the effects of Reynolds number, Mach number, configuration and wall temperature on the flow field and heat transfer rate are discussed. The interesting feature in some of these latter references is in the approach in which the researcher is forced to make studies related to a specific project rather than with a fundamental objective in mind.

5.5 Shock impingement

The most severe aerodynamic heating is undoubtedly due to shock impingement or the interaction of an externally generated shock with the bow wave at the leading edge of a blunt body (for example, that caused by an aircraft on its store). Examples of practical configurations in which shock impingement arises are shown in figure 7. This problem was identified by investigators at an early period, but it was not until the extensive study of Edney (Ref. 66) who obtained very detailed schlieren photographs of a remarkable quality, that understanding of the interference flow field was gained. Edney measured the shock impingement heating on a hemisphere, a blunted cone and a flat faced cylinder at Mach numbers of 4.6 and 7. Using the quasi steady techniques of laterally sweeping the models relatively slowly across a wedge generated shock in a test section, he obtained accurate local measurements of heat rates and pressures. Very localized heat transfer rates of the order of ten times stagnation heat transfer rates are achieved by such impinging flows. Peak pressures follow very nearly the same distribution of peak heating rates, with maximum values at essentially the same location away from the stagnation point of the hemisphere. This fact again suggests the validity of pressure-heat transfer correlations for this case. Edney found that peak heating rate increases with increasing impinging shock strength and Mach number.

A more recent study of shock interference heating directly following on Edney's work is that of Hains and Keys (Ref. 67). The study encompasses many shock impingement cases on hemispheres, wedges at high angle of attack and on swept fins as well as discussing the application of such results to the design of the space shuttle. Peak heating values of 17 times the stagnation value were recorded and it is predicted that up to 37 times it could occur on the space shuttle attached to the booster during the higher altitude part of the ascent (M = 10, 80,000 meters altitude). Further diagnosis of flows caused by impingement of a shock generated by a body on the leading edge of swept and unswept wings is presented by Bertin (Ref. 68).

6. MEASUREMENTS ON CONFIGURATIONS IN FLIGHT AND IN WIND TUNNELS

In this section, a series of brief statements on results of wind tunnel tests or flight tests and their comparison with engineering correlations are presented, in which surface heating on configurations is featured. There is a severe lack of information on tests on missile shapes at supersonic speeds (as opposed to reentry) and so the information has generally been gleaned from high speed research aircraft, and lifting reentry vehicle sources. It is believed, however, that these remarks are also relevant to missile heating.

Heat transfer measurements measured on the wing of the X-15 airplane at low and high angles of attack and on the fuselage at low angles of attack (Ref. 69) are overestimated by Eckert's reference enthalpy method (Ref. 21) and the theory of van Driest (Ref. 41). Adequate prediction by Eckert's method, however, can be obtained if the effect of wall temperature is neglected. This is further confirmed by reference 70. These results have been under some dispute but later results by Cary (Ref. 71) in ground facilities have substantiated the flight measurements.

Selected incompressible flat plate turbulent heating expressions used in an appropriate manner produce good correlation with flight test data over large ranges of Mach number, Reynolds number and wall temperature ratios for bodies with various cone angles and bluntness ratios (Zoby & Sullivan, Ref. 72). For Re = 10^7 calculated heating rates based on Blasius or Schultz-Grunow friction factors were within +22 and -10% of measurements.

For $10^7 < Re < 8 \times 10^7$ the Schultz-Grunow method gave better correlation.

A convenient correlation of local heat transfer rate with the local surface pressure for blunt cones at various angles of attack was derived by Widhopf (Ref. 73). A method is given by Wing (Ref. 74) for calculating the aerodynamic heating and shear stresses at the wall for tangent ogive noses that are slender enough to maintain an attached nose shock through that portion of flight during which heat transfer is significant. A method is developed by De Jarnette and Hamilton (Ref. 75) which calculates laminar, transitional and turbulent heating rates on space shuttle type configurations at angle of attack in hypersonic flow. Results for blunted circular cones and a typical delta wing space shuttle orbiter at angle of attack indicate that this method yields accurate laminar heat transfer rates and reasonably accurate transition and turbulent heat transfer rates. Improved heat protection of tail fin leading edges was sought by Overmier (Ref. 76) following a sounding rocket flight failure.

Heat transfer studies to an airfoil in oscillating flow reveal at large angle of attack, including those in which stall would occur in steady flow, a strong periodic starting vortex from the leading edge causing a dramatic reattachment of the flow and an increase in local Nusselt numbers by as much as a factor of 5 (Ref. 77). Experimental studies of vortex induced heating to a cone-cylinder body at Mach 6 by Hefner (Ref. 78) indicate that the most severe lee-surface heating need not occur as a result of the interaction of the primary vortices with the lee-surface. Vortex heating studies to cone flaps also at Mach 6 by Hefner and Whitehead (Ref. 79) indicate that locally high heating can occur on leeside flaps at vehicle angle of attack. This heating is generally less than the maximum heating at zero angle of attack (for the same flap deflection) but can be greater than the vortex induced heating on the configurational forebody at the same angle of attack.

In connection with space shuttle, the range of Mach number and Reynolds number has been expanded by Creel (Ref. 80) to assess their effect on orbiter/tank interference heating. The primary effect of interference in these tests is to cause transition to fully developed turbulent flow. As a result of extensive experimental studies, a technique has been developed for predicting shock shapes; pressures and turbulent heating rates on the leading edge of a fin, swept wing antenna or similar highly swept near its intersection with a high speed vehicle (Ref. 81).

There are significant increases in heat transfer rate over large regions of the Apollo command module due to the presence of protuberances and cavities (present in its reentry configuration) at $M = 10$ and $1.4 \times 10^6 < 2.6 < 10^6$ (Ref. 82). For the evaluation of the thermal performance of RSI (reusable surface insulation) tiles for protection of space shuttle surfaces, results show that unfilled gaps between tiles seem hotter than comparable undisturbed areas and for transverse and axial gaps a trend of decreasing temperatures with decreasing gap width is evident (Brewer, Saydah, Nestler, Florence, Ref. 83). Dunavant and Throckmorton (Ref. 84) also show that heat transfer is significantly increased when one tile protrudes above another. Further information on gaps and many other topics has been discussed in relation to the ASSET reentry vehicle program by Neumann (Ref. 85).

A series of tests have been made to examine three dimensional shock wave/turbulent boundary layer interactions on a finned missile configuration by Hayes (Ref. 86) with aerodynamic heating providing the main focus of interest. The work is a continuation of fin/flat plate studies discussed earlier. The model was basically an ogive cylinder instrumented with 200 thin skin heat transfer sensors. During the test series several types of control surfaces (fins and canards) were mated to the body and test at different roll angles and an angle of attack ranges from 0 to 12°. The effect of gaps between the control surface and the bodies were examined. For the "sealed" control surfaces, the peak heating in the interaction region was correlated and found to be predicted by a pressure interaction theory with an exponent of 0.8 (Ref. 58).

Flight and wind tunnel measurements were conducted on a pylon-mounted store by Matthews and Keys (Ref. 1).The main objective of the tests was to substantiate the extrapolation of procedures used to assess the heating rates on a full scale store mounted on an F-111 in flight with those measured on a small scale model in a wind tunnel model. Some substantiating evidence was obtained; however, scatter in the flight test data prohibited any definite conclusion.

7. TESTING TECHNIQUES

7.1 Wind tunnels

The prediction of local heating over the whole surface of a missile (or aircraft) configuration can be stated as still a large way from possible, because of the complexity of the flow. Wind tunnel testing is hence likely to play a continuing important role for a period far enough in the future that new facilities are being considered. For flow Mach numbers below 5, there exist very few heated flow facilities available for studying kinetic heating. The fact that there are many tunnels with heated flows above Mach 5 is only for the reason of preventing flow condensation. For simulating flow over missiles at high supersonic Mach numbers, one could thus envisage replacing the hypersonic nozzles of these latter tunnels with lower Mach number versions. Because of the power levels needed to heat such flows, usually these hypersonic tunnels, thus also such converted tunnels, tend to have small test sections, so small that configurational testing is difficult to accomplish. Nevertheless, these facilities provide a useful service in testing simple models fo single components of a configuration.

Two types of facilities could be of value in future development of high speed missiles in which heating plays a critical role. The first one would be used for testing a complete missile including its structure and systems. Such a tunnel should have a test section of large enough size to test a complete missile, a flow temperature range the same as the stagnation temperature range of future projected missiles and the correct Mach number and Reynolds number range. A call for such an "aerothermal wind tunnel" with a Mach number performance of 2 to 6 with actual flight time and temperature duplication and full scale Reynolds numbers for missiles is given in reference 87 (Fig. 13). A 6 ft (1,80 m) diameter test section would be sufficiently large to test most full scale tactical missiles. The operating envelope that would be required to provide flight duplication in a 6 ft diameter wind tunnel is given in figure 14. It can be seen that an air supply of 3000 lb/sec (1360 kg/sec) heated to 3000°R (1670°K) at a pressure of approximately 1500 lb/m^2 (100 atmospheres) would be required. This would constitute a major facility but reference 88 discusses that the technology for such a tunnel is available.

The other type of tunnel is one in which one wishes to test the aerodynamics and heating to the external surface of a configuration. In this case, both the flight time and true temperature do not have to be duplicated, hence one can fall back on economic short duration tunnels operating near the flow condensation temperature (although one should have sufficient flow temperature to create a heat transfer to models at ambient temperature for the lower Mach number cases). The important parameters to simulate are Mach number and Reynolds numbers. Provided the wall to recovery temperature ratio is more than a few per cent higher than unity, the measured value of the heat transfer coefficient, h, (or St or Nu) will be applicable for other wall temperatures. Higher accuracy can be achieved in h by using as high a T_w/T_r as possible. Although such facilities are on a scale which is modest compared to the "aerothermal tunnel" mentioned above (i.e., approximately 600 K, 2 m test section, 40 atm, $2 < M < 6$), such tunnels do not exist at present, mainly because of a recognition of the requirement of such a facility only recently.

An interim "aerothermal" simulation technique for testing materials on simple flat plate or hemisphere shapes has been devised by Matthews and Stallings (Ref. 89). Conventional continuous hypersonic tunnels, M = 6 and 10, were used to test materials mounted on a wedge at such an angle of attack and such supply conditions to get M, Re and T_{stag} correct (see Fig. 15). Through such a technique it was possible to simulate the flow on the body surface and around a stabilizing fin of a missile at M = 4, altitude of 60,000 ft (18,000 m). A novel optical technique was described to measure the recession rate on tested materials.

For aerodynamic and heating simulation, to overcome the difficulties associated with the high power levels required and materials and instrumentation problems involved with continuously hypersonic operating tunnels, short duration tunnels were devised (for a review of such developments, see Lukasiewicz, Ref. 90) and have been found indispensable in the Apollo and space shuttle programs. For heat transfer studies, a further advantage is that fast response, sensitive and accurate transient methods of measurement can be used (Schultz and Jones, Ref. 91). The modest temperature requirement needed for aerodynamic heating studies of missile configurations suggest the application of a class of facility recently devised for the different application of heated turbine component testing (which also requires only a modest temperature) by Jones, Schultz and Hendley (Ref. 92) called the isentropic light piston tunnel. A large version of this turbine facility has been built at VKI (Refs. 93, 94). This VKI tunnel has been designed with a future possibility of using it for studies of heat transfer flows over vehicle configurations at high Mach numbers and its application to missile testing may be most appropriate. Its characteristics are for it to generate flow temperatures up to 600 K at pressures of 40 atmospheres for running times up to 1 second over which time it develops a power of 7 MW. It is driven economically from the Institute's 250 atm compressed air supply. With a Mach 4 nozzle, a test section size of 0.5 m diameter and a flow Reynolds number of 1.8×10^8 per meter could be achieved.

7.2 Heat transfer measurements

In the case of testing in "aerothermal" test facilities, the main emphasis on measurement techniques will focus on temperature measurements of internal parts of the missile rather than heat transfer measurements to the exterior surface. The subject of temperature measurement is dealt with in detail in reference 95 and no further mention will be made on this except in relation to heat transfer measurements.

7.2.1 Steady state technique

Heat transfer measurements using steady state techniques are extremely difficult to carry out. The most complete and up-to-date review on this subject is given by Winter in reference 96 (which also, however, contains a review of transient techniques) based on work carried out at RAE in England. Another useful reference is that of Eckert and Goldstein (Ref. 97).

To make heat transfer measurements under steady state conditions it is necessary to supply power to a model equivalent to that transferred to the air stream or to extract power from the model accordingly as the model is maintained at a temperature higher or lower than the recovery temperature of the air stream. The latter technique is normally carried out by passing a cold liquid through the interior of the model. An example of the former is that carried out in the VKI 0.40 m continuous supersonic tunnel S-1 on a wedge (Ref. 98). The wedge was cast in an epoxy resin around thermocouples and then machined to produce a smooth surface and expose the thermocouple junctions. The heating element was provided by silver plating the whole surface using a mirror-silvering process resulting in a film of fairly uniform thickness of about 1 μm. The film was connected by silver paint to electrodes running along the sides of the model. If it is assumed that the film is of uniform thickness and resistivity and that the current is supplied uniformly, then the heat dissipated will be uniform over the whole surface. Thus measurements of the surface temperature, T_w, of the heated model and of the temperature, T_r, of the unheated model and a knowledge of the electrical power, q, dissipated per unit area enable a heat transfer coefficient, h, to be determined from

$$h = \frac{q}{T_w - T_r}$$

In any steady state technique, care is needed to ensure the equilibrium conditions have been obtained. For conditions of M = 2.2 and Reynolds numbers of 1.6 and 3.3×10^6 per m, a running time of 1 hour was required to reach equilibrium.

Although the method as described gives information at constant hear transfer, using multiple elements on a surface, constant model surface temperature can more easily be obtained and with appropriate automatic control the response time of the system could be as fast as the flow establishment or tunnel starting time, since no model temperature change is involved. This method would be much more complicated than the "semi-infinite slab" transient method to be described, but would be applicable in cases in which rapid tunnel start or rapid model injection was impractical (since the method is not dependent on the running time) and would prove useful in activating existing continuous or blowdown ambient temperature supersonic wind tunnels for kinetic heating studies (A further philosophy justifying this latter approach would be that less energy would be needed to heat the model than heat the flow). The heat transfer coefficients thus obtained would be applicable to the real situation as long as M and Re were simulated and the main conclusion would arise from the boundary layer characteristics (e.g., position of transition, boundary layer thickness, etc.) being not well simulated.

The more classic approach to steady state heat transfer measurement involved internally cooling a model fitted with a thin skin uniform skin and measuring the temperature gradient across the skin. The choice of material skin is a balance between a good conductor (high heat transfer rate, but low temperature difference) and a poor conductor (low heat transfer rate, but high temperature difference across the skin). Winter (Ref. 96) shows that for maximum sensitivity the temperature drop through the skin should be half the temperature drop in temperature from the aerodynamic recovery temperature to the coolant temperature. An extensive review is made by Winter on heat flow meters appropriate to these steady flow techniques, many developed at RAE - Bedford.

Another possible approach is through the use of the analogy of heat transfer with mass transfer. This has been applied in the form of measuring the sublimation rate of naphalene (Eckert and Goldstein, Ref. 97).

7.2.2 Transient techniques

Measurements in aerodynamic heating facilities are more popularly carried out using transient techniques. These methods depend normally on a quick start tunnel flow (<0.1 secs) such as produced in such short duration facilities as shock tunnels, piston tunnels, etc, or on the rapid injection of the model into a blowdown or continous tunnel. A typical range of techniques used in presented in reference 99.

The thin skin or calorimeter technique involves the measurement of the temperature of the thin skin made of a good conductor which has been thermally insulated on the back side (Refs. 91, 100). For a perfectly constructed gauge, the heat transfer rate is then given simply by the relation

$$q = \rho b c \frac{dT_w}{dt} \tag{18}$$

where ρ, b and c are the denstiy, thickness and specific heat of the skin material. In practice, imperfections in construction are such that the effective thickness, b, must be found by direct calibration. For the flows of interest, it is quite possible that the skin temperature rise is important with relation to the driving temperature, $T_r - T_w$, which must be taken into account in defining the heat transfer coefficient, h, i.e.

$$h = \frac{\rho b c \; dT_w/dt}{T_r - T_w} \tag{19}$$

Trimmer et al describe an analysis of the data to assess and eliminate errors due to heat conduction along the skin. Errors due to thermal radiation can normally be neglected. The temperature sensor can be a thermocouple, thin film resistance thermometer etc.. Response times of such heat sensors are of the order of 1-100 msec. and measurement can be made in a time of 10 msec - 1 second.

Another method was the measurement of the surface temperature of a thick layer of a poor conductor to assess the heat transfer rate. The technique is decribed in detail in reference 91. The measurement time is controlled by the thickness of the material, usually pyrex glass or quartz, required to cause the temperature change at the back surface to be small. Running times of up to 1 second have been achieved using this method (Ref. 101).

The response time is controlled by the characteristics of the temperature sensor which is usually selected to be a thin film platinum resistance gauge or thermocouple painted and baked on to the glass or ceramic backing material. This time can be as low as several microseconds. The heat transfer is then calculated using the following appropriate solution of the one dimensional heat conduction equation :

$$q = \frac{\sqrt{\rho c k}}{\sqrt{\pi}} \left[\frac{T(t)}{\sqrt{t}} + \int_0^t \frac{T(t)-T(\tau)}{(t-\tau)^{3/2}} \, d\tau \right] \tag{20}$$

which can be carried out numerically or by the use of analogues. The analysis of numerical data from an on-line data acquisition system is reviewed in reference 102. A recent development is the use of MACOR ^{R} machinable ceramic available from Corning Glass from which complicated models can be machined and on which can be easily fired platinum film gauges (Ref. 103).

In the space shuttle program, considerable use was made of thermographic techniques for determining heat transfer patterns on these complicated configurations. Such techniques were the thermo-sensitive paints, phase change paints, liquid crystals and thermographic phosphors. These techniques are mentioned in references 91, 97 and 99 and their application to the space shuttle program in reference 17. Models were made of STYCAST, a moulding compound with low thermal conductivity which was selected to enhance the surface temperature variations on the model. For application to the above-mentioned short duration piston tunnel, most of the thermographic techniques will respond to the changes in surface temperature to a sufficient extent in the relatively long running time. This suggests an economic but effective way of generating experimental data of a developmental nature.

The most sophisticated technique available that has the attributes of the thermographic techniques with the accuracy of the transient techniques discussed earlier involves the use of an infra-red scanning camera (see paper 22 by D.L. Compton in Ref. 17). Combined with an on-line data acquisition system to deal with the vast amount of data collected, accurate overall surface measurements could be achieved on configurations. The method would require, however, some substantial development.

8. CONCLUDING REMARKS

In this report, experimental and theoretical aspects of kinetic heating of missiles are reviewed. Simple correlation methods appear to be widely used and reasonably accurate for predicting heat transfer rates due to attached flow or near two dimensional or axi-symmetric surfaces. There is sufficient literature available to be aware that surface flow interactions can cause severe heating. Crude correlations have been devised to predict the level of the maximum heating rates achieved, provided the pressure field is known, but little information is available on the location of such regions of heating.

There have been recently dramatic advances in computational methods and computers to solve the full Navier-Stokes equations towards predicting the flow over complicated shapes, but even the most optimistic workers in this field expect one to wait another 10 or 15 years to achieve such solutions at a developmental level, even in well equipped establishments. There is thus a call for economic wind tunnel tests using easy-to-build models and heat visualization measuring techniques to privode rapid answers on complete configurations. Suitable measurement techniques are reviewed.

REFERENCES

1. MATTHEWS, R.K. & KEY, J.C.: Comparison of wind tunnel and flight test heat transfer measurements on a pylon mounted store.
J. Aircraft, Vol. 14, No 6, June 1977, pp 565-568.
2. CRABTREE, L.F.; DOMMETT, R.L. & WOODLEY, J.G.: Estimation of heat transfer to flat plates, cones and blunt bodies.
RAE TR 65137, July 1965.
3. CARSLAW, H.S. & JAEGER, J.C.: Conduction of heat in solids.
Oxford University Press, 1947.
4. JOHNS, D.J.: Thermal stress analyses.
Pergamon Press, September 1965.
5. SCHUH, H.: Heat transfer in structures.
Pergamon Press, 1965.
6. JOHNS, D.J. & WOOLRIDGE, M.J.: Some approximate formulae for transient temperature prediction in aircraft type structures.
J. Royal Aeron. Soc., Vol. 71, 1967, p. 223.
7. BEELER, D.E.: The X-15 research program.
AGARD Report 289, October 1960.
8. CHAPMAN, A.J. & DOW, M.B.: Arc tunnel evaluation of some ablative heat shield materials for the X-15-2.
NASA TN D 3753, February 1967.
9. ERB, R.B.; GREENSHIELDS, D.H.; CHAUVIN, L.T.; PAVLOVSKY, J.E.; STATHAM, C.L.: Apollo thermal protection system development.
J. Spacecraft & Rockets, Vol. 7, No 6, June 1970, p. 727.

10. MASEK, R.V.; HENDER, D.; FORNEY, J.A.: Evolution of aerodynamic heating uncertainties for space shuttle.
J. Spacecraft & Rockets, Vol. 11, No 6, June 1974, p. 368.
11. HURWICZ, H.; KRATSCH, K.M.; ROGAN, J.E.: Ablation.
AGARDograph 161, March 1972.
12. LIEPMANN, H.W. & ROSHKO, A.: Elements of gasdynamics.
Wiley 1957.
13. HAYES, W.D. & PROBSTEIN, R.F.: Hypersonic flow theory. 2nd edition, Vol. I.
Academic Press, 1966.
14. KRAUS, W.: Panel methods in aerodynamics; in
"Computational Fluid Dynamics", VKI LS 87, March 1976; also
"Numerical Methods in Fluid Dynamics", Hemisphere Publ. Corp., 1978, pp 237-297.
15. CEBECI, T. & SMITH, A.M.O.: Analysis of turbulent boundary layers.
New York, Academic Press, 1974.
16. SAE Aerospace Applied Thermodynamics Manual. Part 1C - Engineering Fundamentals - Heat Transfer, and Part 4A - Boost Heat Transfer.
17. Space Shuttle Aerothermodynamics Technology Conference. Volume II - Heating.
NASA TM X 2507, February 1972.
18. FAY, J.A.; RIDDELL, F.R.; KEMP, N.H.: Stagnation point heat transfer in dissociated air flow.
Jet Propulsion, Vol. 27, No 6, June 1957, p. 672.
19. FAY, J.A. & RIDDELL, F.R.: Theory of stagnation point heat transfer in dissociated air.
J. Aero. Sc., Vol. 25, No 2, February 1958, p. 73.
20. BECKWITH, I.E. & GALLAGHER, J.E.: Local heat transfer and recovery temperatures on a yawed cylinder at a Mach number of 4.15 and high Reynolds numbers.
NASA TR R 104, 1961.
21. ECKERT, E.R.G.: Survey on heat transfer at high speeds.
WADC TR 54-70, April 1954.
22. LEES, L.: Laminar heat transfer over blunt-nosed bodies at hypersonic flow speeds.
Jet Propulsion, Vol. 26, No 4, April 1956, pp 259-269.
23. COHEN, C.B. & RESHOTKO, E.: The compressible laminar boundary layer with heat transfer and pressure gradient.
NACA TR 1294, 1956.
24. MICHEL, R.: Etude de la transition sur les profils d'aile. Etablissement d'un point de transition et calcul de la traînée de profil en incompressible.
ONERA Report 1/1578A, July 1951.
25. SMITH, A.M.O. & GAMBERONI, N.: Transition, pressure gradient and stability theory.
Proc. 9th Int. Congr. Appl. Mech., Brussels, Belgium, 1956.
26. CRABTREE, L.F.: Prediction of transition in the boundary layer on an aerofoil.
RAE TN Aero 2491, 1957.
27. HOPKINS, E.J.; JILLIE, D.W.; SORENSON, V.L.: Charts for estimating boundary layer transition on flat plates.
NASA TN D 5846, 1970.
28. BENEK, J.A. & HIGH, M.D.: Transition prediction technique.
AIAA J., Vol. 12, No 10, October 1974, pp 1425-1427.
29. MORKOVIN, M.V.: Technical evaluation report of the Fluid Dynamics Panel Symposium on Laminar-Turbulent Transition.
AGARD AR 22, June 1978.
30. DHAWAN, S. & NARASIMHA, R.: Some properties of boundary layer flow during transition from laminar to turbulent motion.
J. Fluid Mechanics, Vol. 3, Part 4, January 1958, pp 418-436.
31. MASAKI, M. & YAKURA, J.: Transitional boundary layer considerations for the heating analyses of lifting re-entry vehicles.
AIAA Paper 68-115, December 1968.
32. CHEN, K.K. & TYSON, N.A.: Extension of Emmons spot theory to flows on blunt bodies.
AIAA J., Vol. 9, No 5, May 1971, p. 821.
33. STAINBACK, P.; WAGNER, R.D.; OWEN, F.K.; HORSTMAN, C.C.: Experimental studies of hypersonic boundary layer transition and effects of wind tunnel disturbances.
NASA TN D 7453, March 1974.
34. HARRIS, J.E.: Numerical solution of the equations for laminar, transitional and turbulent boundary layers and comparison with experimental data.
NASA TR R 368, 1971.
35. ADAMS, J.C. Jr: Eddy viscosity - intermittency factor approach to numerical calculations of transitional heating on sharp cones in hypersonic flow.
AEDC TR 70-210, November 1970.
36. SHAMROTH, S.J. & McDONALD, H.: Assessment of a transitional boundary layer theory at low hypersonic Mach numbers.
NASA CR 2131, November 1972.
37. SOMMER, S.C. & SHORT, B.J.: Free flight measurements of turbulent boundary layer skin friction in the presence of severe aerodynamic heating at Mach numbers from 2.8 to 7.0
NACA TN 3391, 1955.
38. SPALDING, D.B. & CHI, S.W.: The drag of a compressible turbulent boundary layer on a smooth flat plate with and without heat transfer.
J. Fluid Mechanics, Vol. 18, Part 1, January 1964, pp 117-143.
39. WHITE, F.M. & CRISTOPH, G.H.: A simple new analysis of compressible turbulent two dimensional skin friction under arbitrary conditions.
AFFDL TR 70-133, WPAFB, Ohio, February 1971.
40. HOPKINS, E.J. & INOUYE, M.: An evaluation of theories for predicting turbulent skin friction and heat transfer on flat plates at supersonic and hypersonic Mach numbers.
AIAA J., Vol. 9, No 6, June 1971, pp 993-1003.
41. VAN DRIEST, E.R.: Problem of aerodynamic heating.
Aeron. Engrg. Rev., Vol. 15, No 10, October 1956, pp 26-41.

42. von KARMAN, Th.: The analogy between fluid friction and heat transfer.
ASME Transact., Vol. 64, No 8, November 1939, pp 705-710.
43. BRADSHAW, P.: An improved Van Driest skin-friction formula for compressible turbulent boundary layers.
AIAA J., Vol. 15, No 2, February 1977, pp 212-214.
44. BUSHNELL, D.M.; CARY, A.M.; HARRIS, J.E.: Calculation methods for compressible turbulent boundary layers. in
"Compressible Turbulent Boundary Layers", VKI LS 86, March 1976.
45. REYNOLDS, W.C.: Computation of turbulent flows.
Annual Review of Fluid Mechanics, Vol. 8, 1976, p. 183.
46. BRADSHAW, P.: The turbulent structure of compressible boundary layers. in
"Compressible Turbulent Boundary Layers", VKI LS 86, March 1976.
47. CHAPMAN, D.R.: Status and prospects of computational fluid dynamics. in
"Computational Fluid Dynamics", VKI LS 87, March 1976.
48. BRADSHAW, P.B. & UNSWORTH, K.: An improved FORTRAN program for the Bradshaw-Ferris-Ahwell method of calculating turbulent shear layers.
Imperial College, Aero Report 74-02, February 1974.
49. CRAWFORD, M.E. & KAYS, W.M.: STAN5 - A program for numerical computation of two dimensional internal and external boundary layer flows.
NASA CR 2742, December 1976.
50. KORKEGI, R.H.: Survey of viscous interactions associated with high Mach number flight.
AIAA J., Vol. 9, No 5, May 1971, pp 771-784.
51. CHAPMAN, D.R.; KUEHN, D.M.; LARSON, H.K.: Investigation of separated flows in supersonic and subsonic streams with emphasis on the effect of transition.
NACA TR 1356, 1958.
52. FLETCHER, M.S.; BRIGGS, D.G.; PAGE, R.H.: A review of heat transfer in separated and reattached flows.
AIAA Paper 70-767, June 1970.
53. HANKEY, W.L. & HOLDEN, M.S.: Two dimensional shock wave/boundary layer interaction.
AGARDograph 203, 1975.
54. BUSHNELL, D.M. & WEINSTEIN, L.M.: Correlation of peak heating for reattachment of separated flows.
J. Spacecraft & Rockets, Vol. 5, No 9, September 1968, pp 1111-1112.
55. JOHNSON, C.B. & KAUFMAN, L.G. II: Interference heating from interactions of shock waves with turbulent boundary layers at Mach 6.
NASA TN D 7649, September 1974.
56. NEUMANN, R.D. & BURKE, G.L.: The influence of shock wave boundary layer effects on the design of hypersonic aircraft.
AFFDL TR 68-152, March 1969.
57. OSKAM, B.; VAS, I.E.; BOGDONOFF, S.M.: An experimental study of three dimensional flow fields in an axial corner at Mach 3.
AIAA Paper 77-689, June 1977.
58. NEUMANN, R.D. & HAYES, J.R.: Prediction techniques for three dimensional shock wave/turbulent boundary layer interactions.
AIAA J., Vol. 15, No 10, October 1977
59. SPENCER, G.D.: AFFDL trip effects on interference heating.
AEDC TR 77-61, 1977.
60. CHARWAT, A.F. & REDEKOPP, L.: Supersonic interference flow along the corner of intersecting wedges.
AIAA J., Vol. 5, No 3, March 1967, pp 480-488.
61. BERTRAM, M.H. & HENDERSON, A. Jr: Some recent research with viscous interacting flows in hypersonic streams.
Proc. Symp. Viscous Interaction Phenomena in Supers. & Hypers. Flow, USAF-ARL, May 1969, pp 1-30.
62. RAINBIRD, W.J.: Some problems of three dimensional separation about aircraft components.
AIAA Paper 69-662, 1969.
63. PEAKE, D.J.; RAINBIRD, W.J.; ATRAGHI, E.G.: Three dimensional flow separations on aircraft and missiles.
AIAA J., Vol. 10, No 5, May 1972, pp 567-580.
64. BERTIN, J.J. & GOODRICH, W.D.: Effects of surface temperature and Reynolds numbers on leeward shuttle heating.
J. Spacecraft & Rockets, Vol. 13, No 8, August 1976, pp 473-480.
65. BERTIN, J.J.: A study of parameters which influence surface pressure and heat transfer in separated regions - a literature survey.
U. Texas at Austin, Aerospace Engineering Report 74004, September 1974.
66. EDNEY, B.E.: Anomalous heat transfer and pressure distribution on blunt bodies at hypersonic speeds in the presence of an impinging shock.
Aeron. Res. Inst. Sweden, Report 115, February 1968; also
Effects of shock impingement on the heat transfer around blunt bodies.
AIAA J., Vol. 6, No 1, January 1968, pp 15-21.
67. HAINS, F.D. & KEYES, J.W.: Shock interference in hypersonic flows.
AIAA J., Vol. 10, No 11, November 1972, pp 1441-1447.
68. BERTIN, J.J.; GRAUMANN, B.W.; GOODRICH, W.D.: Aerothermodynamic aspects of shock interference patterns for shuttle configurations during entry.
J. Spacecraft & Rockets, Vol. 10, No 9, September 1973, pp 545-546.
69. QUINN, R.D. & PALITZ, M.: Compression of measured and calculated turbulent heat transfer on the X-15 airplane at angles of attack up to 19°.
NASA TM X 1291, September 1966.
70. GORD, P.R.: Measured and calculated structural temperature data from two X-15 airplane flights with extreme aerodynamic heating conditions.
NASA TM X 1358, March 1967.

71. CARY, A.M. Jr: Turbulent boundary layer heat transfer and transition measurements for cold wall conditions at Mach 6.
AIAA J., Vol. 6, 1968, p. 958.

72. ZOBY, E.V. & SULLIVAN, E.M.: Correlation of free flight turbulent heat transfer data from axisymmetric bodies with compressible flat plate relationship.
NASA TN D 3802, 1967.

73. WIDHOPF, G.F.: Heat transfer correlation for blunt cones at angle of attack.
J. Spacecraft & Rockets, Vol. 8, No 9, September 1971, pp1002-1004.

74. WING, L.D.: A method for calculating aerodynamic heating on sounding rocket tangent ogive noses.
J. Spacecraft & Rockets, Vol. 11, No 6, June 1974, pp 357-362.

75. DE JARNETTE, F.R. & HAMILTON, H.H.: Inviscid surface streamline and heat transfer on shuttle type configurations.
J. Spacecraft & Rockets, Vol. 10, No 5, May 1973, pp 314-321.

76. OVERMIER, D.K.: Development of airframe heat protection for a high performance sounding rocket.
J. Spacecraft & Rockets, Vol. 4, No 11, November 1967, pp 1500-1504.

77. MILLER, J.A. & PUCCI, P.F.: Heat transfer to an airfoil in oscillating flow.
ASME Transact., Series A : J. Engrg for Power, Vol. 93, No 4, October 1968, pp 461-468.

78. HEFNER, J.N.: Vortex induced heating to a cone-cylinder body at Mach 6.
J. Spacecraft & Rockets, Vol. 11, No 2, February 1974, pp 127-128.

79. HEFNER, J.N. & WHITEHEAD, A.H. Jr: Vortex induced heating to cone flaps at Mach 6.
J. Spacecraft & Rockets, Vol. 11, No 3, March 1974, pp 200-201.

80. CREEL, T.R. Jr: Mach number and Reynolds number effect on orbiter/tank.
J. Spacecraft & Rockets, Vol. 10, No 8, August 1973, pp 535-537.

81. COLEMAN, H.W. & LEMMON, E.C.: Prediction of turbulent heat transfer and pressure on swept leading edges.
J. Spacecraft & Rockets, Vol. 11, No 6, June 1974, pp 376-381.

82. BERTIN, J.J.: The effect pf protuberances, cavities and angle of attack on the wind tunnel pressure and heat transfer distribution for the Apollo commande module.
NASA TM X 1243, 1966.

83. BREWER, R.A.; SAYDAH, A.R.; NESTLER, D.E.; FLORENCE, D.E.: Thermal performance evaluation of RSI panel gaps for space shuttle orbiter.
J. Spacecraft & Rockets, Vol. 10, No 1, January 1973, pp 23-28.

84. DUNAVANT, J.C. & THROCKMORTON, D.A.: Aerodynamic heating to RSI tile surfaces and gap interactions.
J. Spacecraft & Rockets, Vol. 11, No 6, June 1974, pp 437-440.

85. NEUMANN, R.D.: Special topics in hypersonic flow.
AGARD LS 42, Vol. I, 1972.

86. HAYES, J.R.: An overview of the test program and representative data taken on the modular missile configuration.
AFFDL TM 77-90-FXG, October 1977.

87. Technical Staff, AEDC & ARO: Aerodynamic testing - a look at future requirements.
AIAA Paper 78-765, AIAA 10th Aerodynamic Testing Conference, San Diego, April 1978.

88. TRIMMER, L.L. & MATTHEWS, R.K.: Supersonic aerothermal testing - a new requirement.
AIAA Paper 78-733, AIAA 10th Aerodynamic Testing Conference, San Diego, April 1978.

89. MATTHEWS, R.K. & STALLINGS, D.W.: Materials testing in the VKF continuous flow wind tunnels.
Paper presented at the AIAA 9th Aerodynamic Testing Conf., Arlington, Texas, June 1976.

90. LUKASIEWICZ, J.: Experimental methods of hypersonics.
Marcel Dekker Inc., New York, 1973.

91. SCHULTZ, D.L. & JONES, T.V.: Heat transfer measurements in short duration hypersonic facilities.
AGARDograph 165, 1973.

92. JONES, T.V.; SCHULTZ, D.L.; HENDLEY, A.D.: On the flow in an isentropic free piston tunnel.
ARC R&M 3731, January 1973.

93. RICHARDS, B.E.: Isentropic light piston facilities for simulation of hot flows through turbines. in
Advanced Testing in Turbomachines, VKI LS 78, April 1975.

94. CONSIGNY, H.; VILLE, J.P.; RICHARDS, B.E.: The VKI hot cascade tunnel CT2.
VKI TN to be published.

95. HERZFELD, C.M. (ed.): Temperature, its measurement and control in science and industry. Volume 3.
Rheinhold Publishing Company, 1962.

96. WINTER, K.G.: Notes on the measurement of aerodynamic heat transfer in model-testing facilities. in
Compressible Turbulent Boundary Layers, VKI LS 86, March 1976.

97. ECKERT, E.R.G. & GOLDSTEIN, R.J.: Measurements in heat transfer. 2nd edition.
McGraw Hill, 1976.

98. GINOUX, J.J.: A steady state technique for local heat transfer measurement and its application to the flat plate.
J. Fluid Mechanics, Vol. 19, Part 1, 1964, pp 21-29.

99. TRIMMER, L.L.; MATTHEWS, R.K.; BUCHANAN, T.O.: Measurements of heat rates at the AEDC von Karman Facility.
ICIASF'73 Record pp 35-44; published by IEEE.

100. NAYSMITH, A.: Measurement of aerodynamic heat transfer in intermittent wind tunnels.
ARC CP 780, January 1964.

101. RICHARDS, B.E.: Developments in heat transfer measurements using transient techniques.
ICIASF'77 Record; published by IEEE, September 1977.

102. OLDFIELD, M.L.G.; SCHULTZ, D.L.; JONES, T.V.: An on-line interactive minicomputer system for heat transfer measurements in a transient turbine cascade tunnel.
ICIASF'77 Record; published by IEEE, September 1977.

103. SCHULTZ, D.L.; JONES, T.V.; OLDFIELD, M.L.G.; DANIELS, L.G.: A new transient cascade facility for the measurement of heat transfer rates.
AGARD CP 229, Paper 31, February 1978.

Further references

MICHEL, R.: Caractéristiques thermiques des couches limites et calcul pratique des transferts de chaleur en hypersonique.
Progress in Aeronautical Sciences, Vol. 9, pp 123-214 ; Pergamon Press, 1968.
DORRANCE, W.H.: Viscous hypersonic flow.
New York, McGraw-Hill, 1962.
POPE, A. & GOIN, K.L.: High speed wind tunnel testing.
New York, Wiley, 1965.
BRADSHAW, P.B.: Compressible turbulent boundary layers.
Annual Review in Fluid Mechanics, Vol. 9, 1977.

ACKNOWLEDGEMENTS

The author would like to thank the following individuals who made suggestions after kindly reading the first draft of this paper prepared for an earlier von Karman Institute Lecture Series (VKI LS 86) in March 1976 : J.J. Bertin, G.G. Brebner, C. Mathews, R. Michel, R.D. Neumann, W. Oberkampf and K.G. Winter. Thanks are also due to Mme Toubeau who typed the manuscript.

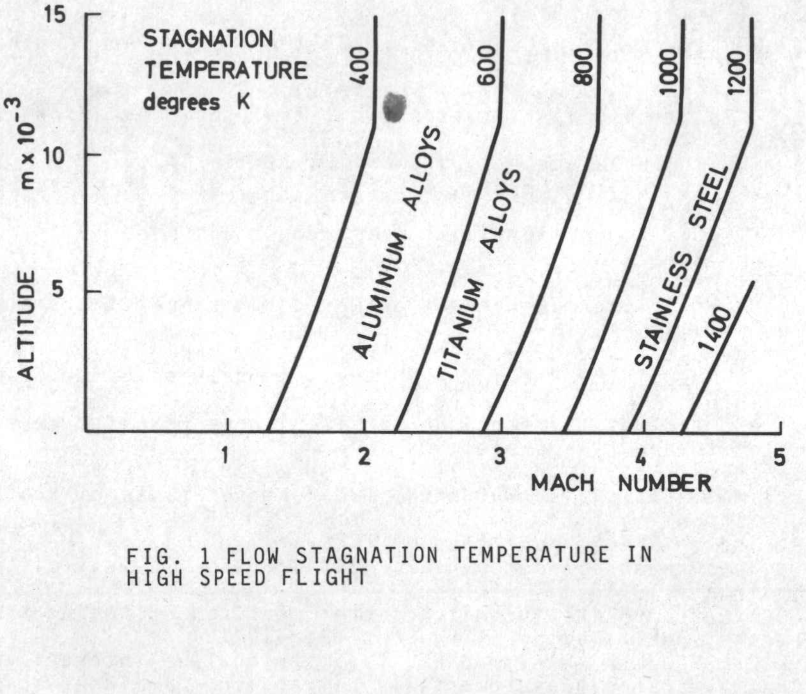


FIG. 1 FLOW STAGNATION TEMPERATURE IN HIGH SPEED FLIGHT

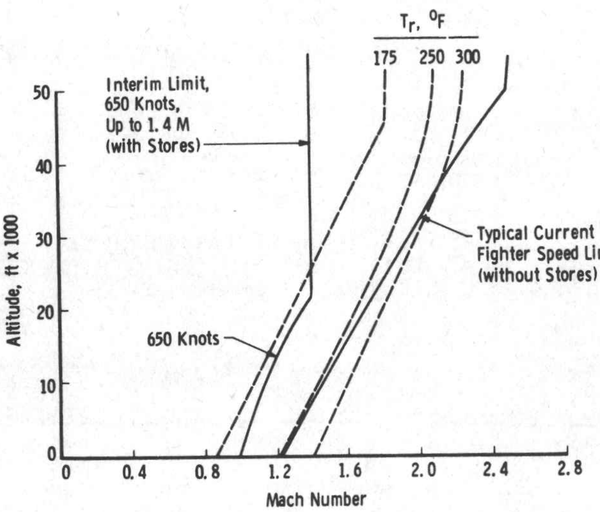


FIG. 2 PERFORMANCE ENVELOPE OF PRESENT-DAY AIRCRAFT (From Ref. 1)

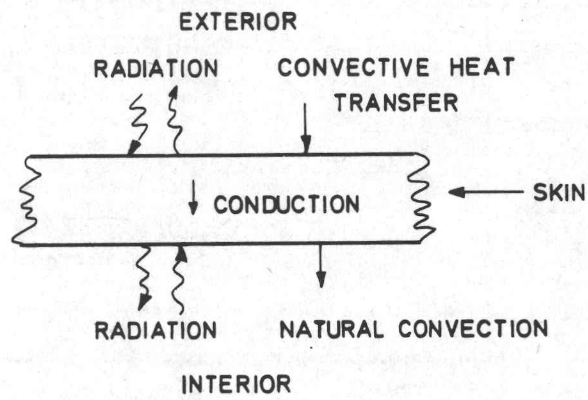


FIG. 3 THE GENERAL HEAT TRANSFER PROBLEM

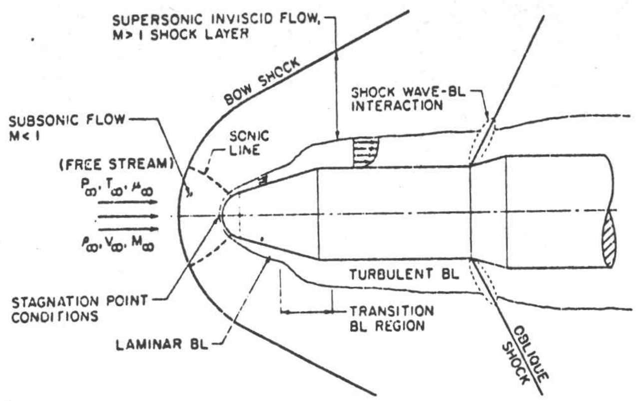


FIG. 4 TYPICAL MISSILE CONFIGURATION AND FLOW FIELD STRUCTURE

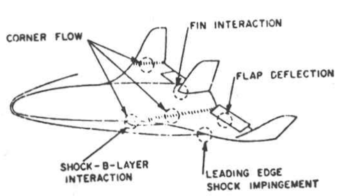


FIG. 5 REGIONS OF STRONG VISCOUS INTERACTION ON A HIGH MACH NUMBER VEHICLE

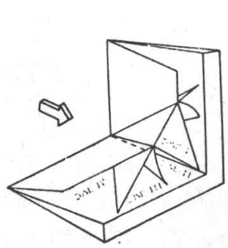


FIG. 6 CORNER FLOW STRUCTURE

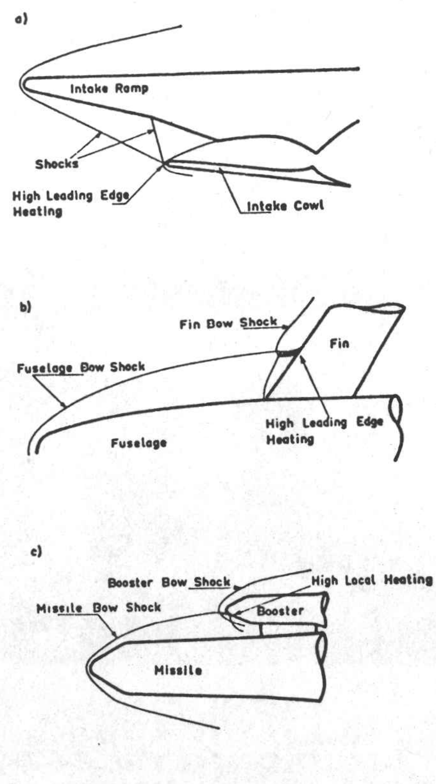


FIG. 7 PRACTICAL EXAMPLES OF SHOCK IMPINGEMENT HEATING

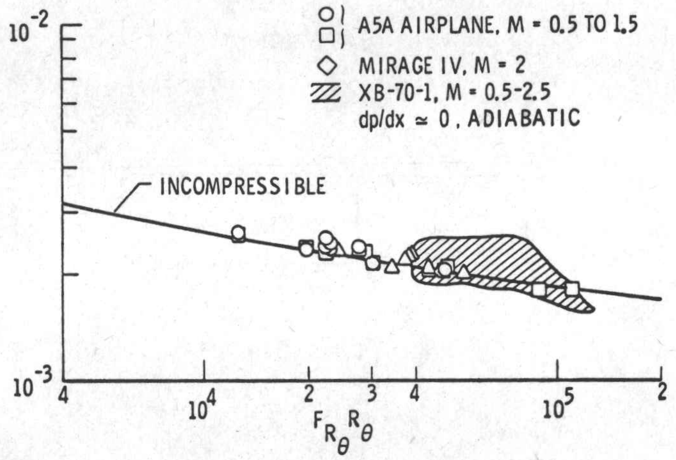


FIG. 8 PREDICTION OF TURBULENT SKIN FRICTION ON CONVENTIONAL FLIGHT VEHICLES BY ECKERT'S METHOD (From Ref. 44)

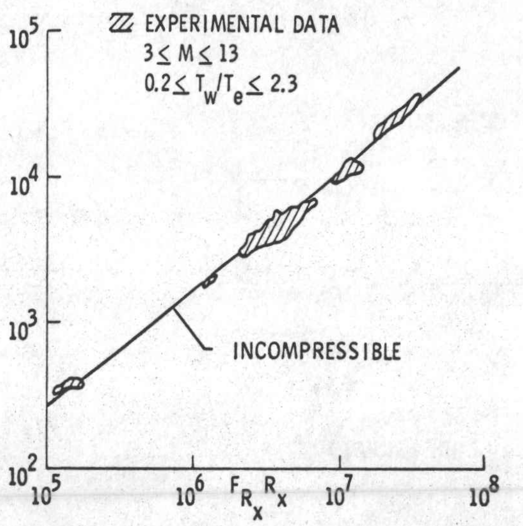


FIG. 9 PREDICTION OF TURBULENT HEATING TO SHARP AND BLUNT TIPPED AXISYMMETRIC BODIES IN FLIGHT BY ECKERT'S METHOD (From Ref. 44)

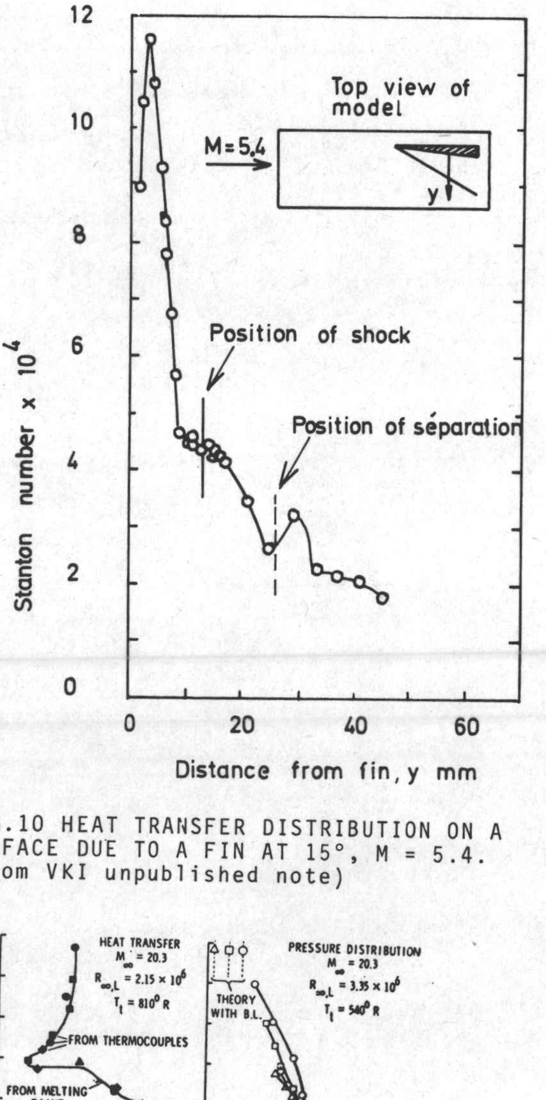


FIG.10 HEAT TRANSFER DISTRIBUTION ON A SURFACE DUE TO A FIN AT 15°, M = 5.4. (From VKI unpublished note)

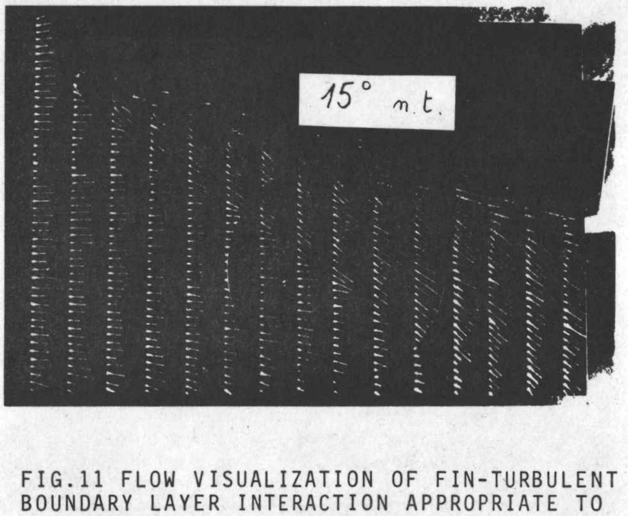


FIG.11 FLOW VISUALIZATION OF FIN-TURBULENT BOUNDARY LAYER INTERACTION APPROPRIATE TO FIG. 10 (From VKI unpublished note)

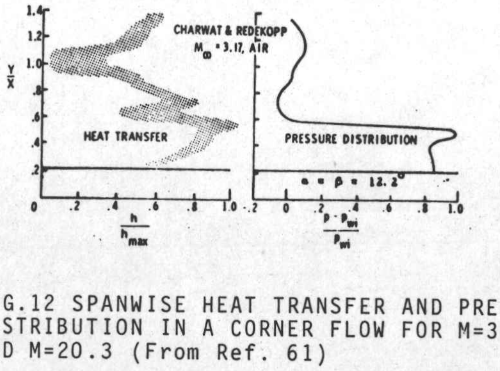


FIG.12 SPANWISE HEAT TRANSFER AND PRESSURE DISTRIBUTION IN A CORNER FLOW FOR M=3.17 AND M=20.3 (From Ref. 61)

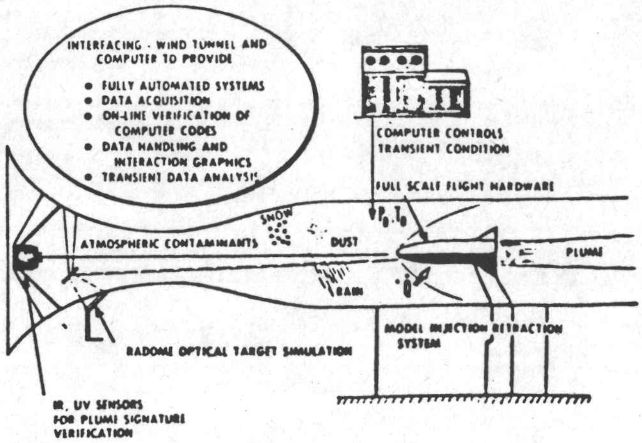


FIG.13 AEROTHERMAL WIND TUNNEL (M NUMBER 2 TO 6) (From Ref. 87)

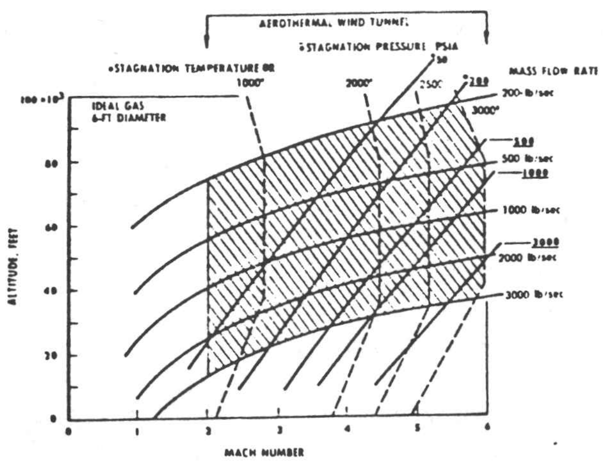
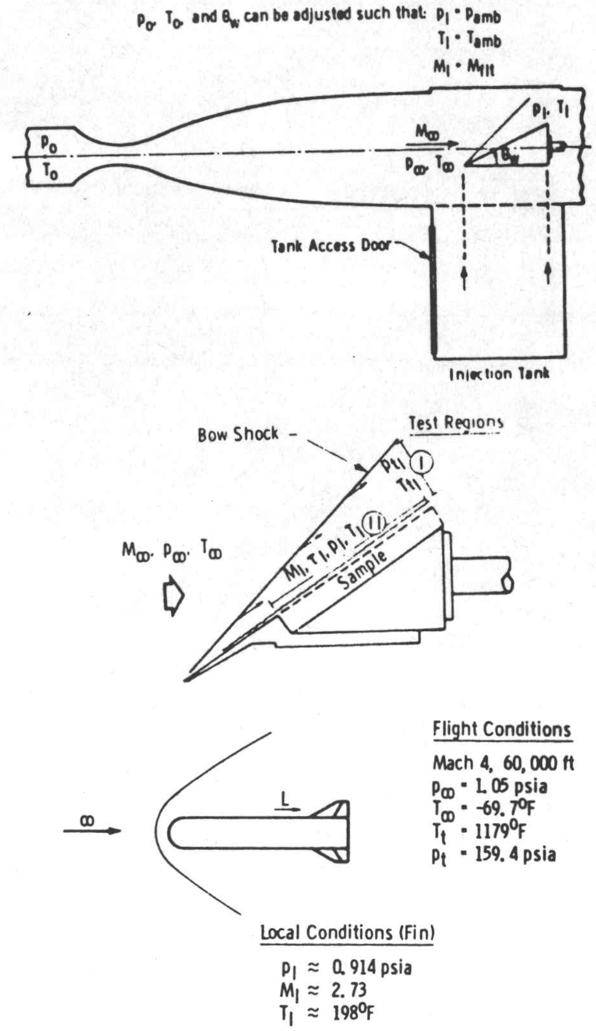


FIG.14 REQUIRED OEPRATING ENVELOPE FOR AN ADVANCED AEROTHERMAL WIND TUNNEL - DUPLICATED FLIGHT (From Ref. 87)



These conditions can be determined from analytical or experimental techniques.

a. Flight Environment

Utilizing AEDC-VKF Tunnel C, $M_\infty = 10$ (Real Gas Corrections < 4%)

Nose	Fin
Test Region I	Test Region II
for $\theta_w = 21$ deg	for $\theta_w = 29.7$ deg
p_o = 2000 psia & T_o = 1179°F	p_o = 965 psia & T_o = 1265°F
The Flow Conditions Are:	The Flow Conditions Are:
$M_{l_{wt}} = 4.0 = M_{\infty_{flt}}$	$M_{l_{wt}} = 2.73 = M_{l_{flt}}$
$T_{t_{wt}}$ = 1179°F = $T_{t_{flt}}$	$p_{l_{wt}}$ = 0.914 psia = $p_{l_{flt}}$
$p_{t_{wt}}$ = 159.4 psia = $p_{t_{flt}}$	$T_{l_{wt}}$ = 198°F = $T_{l_{flt}}$

b. Wind Tunnel Conditions Required to Duplicate Flight Environment

FIG.15 TECHNIQUE FOR MATERIALS TESTING IN THE AEDC CONTINUOUS FLOW WIND TUNNELS (From Ref. 89)