A Review of Some Approximate Methods Used in Aerodynamic Heating Analyses

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Nomenclature

b	= wing span of Shuttle Orbiter
c	= wing root chord
H	= enthalpy
k	= ratio of principal radii of curvature
L	= vehicle length
M	= Mach number
p	= pressure
q	= heating rate
R	= radius of curvature
Re	= Reynolds number
St	= Stanton number
T	= temperature
u_e	= velocity at edge of boundary layer
V	= velocity
$\bar{V}$	= hypersonic viscous interaction parameter, $M/Re^{1/2}$
x,y,z	= Cartesian coordinates
α	= angle of attack
θ_H	= hyperboloid half angle
ρ	= density

Subscripts

e	= boundary-layer edge
L	= laminar
n,N	= nose
s,sp	= stagnation point
w	= wall
x,z	= in x and z directions, respectively
∞	= freestream
2	= aft of normal shock

Introduction

PRELIMINARY design and optimization studies for new aerospace vehicles require techniques which can calculate aerodynamic heating rates accurately and efficiently. The method used to calculate the flowfield is strongly dependent on the shape of the vehicle, Mach number, Reynolds number, and Knudsen number. Vehicles of the Apollo and Space Shuttle classes achieved significant heating rates at sufficiently low altitudes where the flowfield could be considered a continuum and chemical nonequilibrium effects are insignificant.

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However, the Aeroassisted Orbital Transfer Vehicle (AOTV) will fly hypersonically within the Earth's atmosphere at altitudes from 250,000 to 500,000 ft.[1] A substantial portion of its flight will be in the transitional regime between continuum and free molecule flow. In addition, chemical nonequilibrium effects within the shock layer can be significant.[2]

More sophisticated techniques for solving the flowfield include that of Moss and Bird.[3] Their technique uses a Monte Carlo method to calculate low density flows. For the continuum regime, finite-difference solutions to the Viscous Shock Layer (VSL)[4,5,6] or Parabolized Navier-Stokes (PNS)[7,8] equations give accurate predictions of surface heating rates. However, these methods require far more computational effort than can be tolerated for preliminary design or optimization of three-dimensional vehicles. This paper discusses some approximate methods which have been used to calculate heating rates on high-speed vehicles. Both flowfield methods and gas models are considered.

The method used to calculate the flowfield for the continuum flow regime is strongly dependent on the Reynolds number. For relatively large Reynolds numbers the flowfield can be divided into an inviscid outer flow and a boundary layer. The classical approach is to solve the inviscid flowfield first, and then to use the properties on the body surface as edge conditions for a boundary-layer solution. Although this approximation is valid for high Reynolds numbers, its accuracy decreases downstream of the nose region of blunt bodies. As the boundary layer grows along the surface, more and more inviscid mass flow is entrained into the boundary layer. The streamlines which passed through the nearly normal portions of the bow shock-wave are "swallowed" by the boundary layer. When this occurs, the inviscid streamline at the edge of the boundary layer will have crossed an oblique part of the upstream shock wave. The entropy of this streamline differs from normal-shock entropy (see Fig. 1). This phenomenon is called entropy-layer swallowing and it can have a significant effect on the heating rates.[9] Although the pressure at the edge of the boundary layer is nearly the same as the surface pressure, the other properties can be quite different from those corresponding to normal-shock entropy. To account for entropy-layer swallowing effects, the boundary-layer solution must be coupled with the inviscid flowfield solution. For these applications, approximate methods have been developed for both the boundary-layer and inviscid flows.

At relatively low Reynolds numbers, the viscous layer is a significant portion of the shock layer and accurate predictions of surface heating rates require solutions to either the parabolized Navier-Stokes or viscous shock-layer equations. There is a need for approximate methods to solve these equations.

Most flowfield methods have been developed for a perfect gas or equilibrium air. As shown in Fig. 2 (from Ref. 10), nonequilibrium flow can be important for both inviscid and viscous flowfield calculations for re-entry and AOTV vehicles.[2,10,11] Dissociated particles tend to recombine near cold walls in a rate-controlled process. If the surface is catalytic, the reaction goes to completion and the released heat of dissociation increases the surface heating. The tiles on the Space Shuttle have been found to be nearly noncatalytic to the recombination of dissociated air. This caused lower than expected temperatures and heating rates because nonequilibrium and surface catalysis effects were ignored in the earlier prediction methodologies.[11] However, recombination will probably be important on vehicles like the AOTV. Existing chemical nonequilibrium gas models increase the computational effort more significantly than calculations using a perfect gas or equilibrium air. Approximate relations are needed to model chemical nonequilibrium effects.

All flowfield methods require a subprogram to calculate the geometric properties of a vehicle. Numerous techniques have been developed for surface-fitting three-dimensional bodies, but many of them require more computational effort than approximate flowfield methods. The Coons "patching" method[12] is too complex, while the relatively simple method of DeJarnette and Ford[13] is generally inaccurate in the longitudinal description of a body. The QUICK method[14] is reasonably accurate but requires large setup times and greater computational effort. A fast and reliable geometry technique that is user-friendly is still needed for approximate flowfield calculations.

Inviscid Methods

When it is possible to neglect coupling between the inviscid flow and the boundary layer, the surface pressure distribution and normal-shock entropy are all that are needed to calculate properties at the edge of the boundary layer for perfect and equilibrium gases. The modified Newtonian method is frequently used to calculate the pressure distribution over blunt-nosed bodies. Varner et al.[9] found that it is also reasonably accurate for predicting pressures on the windward region of an aircraft body in subsonic as well as supersonic flow. Other simple techniques for estimating surface pressures include the tangent-wedge and tangent-cone methods. These two methods are limited to regions where the inclination of the surface relative to the freestream velocity is less than the shock detachment angle. The tangent-cone method has been found to yield reasonably accurate surface pressures along the windward plane of symmetry of the afterbody of vehicles like the Space Shuttle.[15,16] The tangent-wedge method is applicable to the windward side of wings with relatively flat surfaces. Additional approximate techniques, including the blast wave theory and viscous interaction effects, are discussed by Anderson.[17]

When entropy-layer swallowing effects are important, the inviscid solution off the surface must be coupled with the boundary-layer method in order to calculate the entropy at the edge of the boundary layer. The simplest approximation to this effect is to calculate the entropy aft of the shock wave using the tangent-wedge or tangent-cone methods. Edwards and Cole[16] found this technique to work well along the windward centerline of the Space Shuttle when using the tangent-cone method. Several investigators have used variations of the Maslen method[18] for calculating inviscid flowfield properties. The basic Maslen method[18] is applicable to hypersonic flow over smooth, axisymmetric bodies. It is an inverse method and the solution involves the Von Mises transformation with the stream function and distance along the shock wave as independent variables. A simple, approximate integral of the lateral momentum equation gives the pressure as a linear function of the stream function across the shock layer. The adiabatic energy equation is approximated by neglecting the velocity component normal to the shock wave and the shock-layer thickness is calculated by quadrature. Maslen extended this method to three-dimensional flows,[19] but that technique was found to be too complicated and cumbersome to use in many cases. Zoby[20] used Maslen's second-order pressure relation[19] and an approximate expression for the normal component of velocity to develop an accurate inviscid flowfield method for axisymmetric flows. It was found to predict shock shapes and surface pressures and agreed well with finite-difference methods but with much less computational effort. DeJarnette and Hamilton[21] extended Maslen's basic method[18] to calculate the shock shape corresponding to inviscid surface streamlines over three-dimensional bodies. This technique is a direct one in that a prescribed surface pressure is used to calculate the shock-wave shape.

The approximate inviscid techniques previously discussed are not very accurate when applied away from the nose region and windward plane of symmetry for three-dimensional vehicles. Several finite-difference inviscid flowfield codes that give accurate predictions of the flowfield properties have been developed. Most of these techniques are limited in Mach number[22] or angle of attack[22-24] range. However, vehicles like the Space Shuttle, which operate at large angles of attack (greater than 30 deg), have a local subsonic flow region that extends over a large portion of the windward surface of the vehicle. A time asymptotic computational code[25] called HALIS has been developed to calculate the high angle of attack inviscid flowfield for shuttle-like vehicles. It has been shown to predict surface pressures that compare well with both wind tunnel and flight data for a nearly complete shuttle vehicle[26] at $6.0 < M_\infty < 21.6$ and $26.6 \text{ deg} < \alpha < 40$ deg.

Approximate Heating Methods

Stagnation Point and Leading Edges

For axisymmetric stagnation points, the methods of Fay and Riddell[27] and Cohen[28] have been proven to yield reliable heating rates. Sutton and Graves[29] developed an axisymmetric stagnation-point heat-transfer rate equation of the form

$$(q_s)_{\mathrm{AXI}} = K\left(\frac{p_s}{R_x}\right)^{1/2} (H_s - H_w)$$

where K can be determined using a simple but accurate method over a wide range of gas mixtures. This method compares well with results from Cohen's method.[28] Reshotko[30] developed a simple method for calculating the heating rate at general three-dimensional stagnation points which is applicable to cold wall flight conditions. Hamilton[31] and others have shown that these methods could be combined to give the following expression for the general three-dimensional stagnation point during atmospheric entry:

$$(q_s)_{3D} = \sqrt{\frac{1+k}{2}}(q_s)_{\mathrm{AXI}}$$

The three-dimensional effect enters through the factor

$$k = R_x / R_z$$

which is the ratio of the two principal radii of curvature at the stagnation point. For a Lewis number of unity, this method for the stagnation-point heat-transfer rate is similar to that developed by DeJarnette and Hamilton.[32] Adams and associates[33] found that the velocity gradient at the stagnation point could be calculated more accurately by the relation

$$\left(\frac{\partial u_e}{\partial x}\right)_s = \frac{V_\infty}{R_x}\sqrt{1.85\frac{\rho_\infty}{\rho_s}}$$

rather than the result obtained from modified Newtonian pressures:

$$\left(\frac{\partial u_e}{\partial x}\right)_s = \frac{1}{R_x}\sqrt{\frac{2(p_s - p_\infty)}{\rho_s}}$$

The leading edges of blunted, swept wings have been approximated as infinite swept or yawed cylinders.[34] Even for relatively large sweep angles, the ratio of local heat-transfer rates to the leading-edge heat-transfer rate along a plane perpendicular to the leading edge is insensitive to the angle of yaw.

Downstream Region

McWherter et al.[35] showed that the classical approach, where the outer inviscid flow is calculated independent of the boundary layer, is applicable except for cases of high viscous interaction. Although solutions of the general three-dimensional boundary-layer equations have been obtained for relatively simple shapes,[36-38] this approach is generally too costly for preliminary design calculations.

Axisymmetric Analog

A simpler method to compute the viscous flow uses the "axisymmetric analog" for three-dimensional boundary layers developed by Cooke.[39] Following that approach, the general three-dimensional boundary-layer equations are written in a streamline coordinate system and the cross-flow velocity (tangent to the surface and normal to the streamline direction) is assumed to be zero. This reduces the three-dimensional boundary-layer equations to a form that is identical to those for axisymmetric flow, provided that 1) the distance along a streamline is interpreted as distance along an "equivalent body," and 2) the metric coefficient that describes the spreading of the streamlines is interpreted as the radius of the equivalent body. This allows any existing axisymmetric boundary-layer program to be used to compute the approximate three-dimensional heating along a streamline in regions where the small cross-flow assumption is valid. By considering multiple streamline paths, an entire vehicle can be covered.

Hayes[40] has shown that the cross flow in the boundary layer is small when the streamline curvature is small; and, when the wall is highly cooled (as it is in many practical applications), Vaglio-Laurin[41] has shown that the cross flow in the boundary layer is small even when the streamline curvature is not small. Further, it has been found[42,43] (from comparisons with experimental data and other theoretical calculations, including some cases where the cross flow in the boundary layer is not necessarily small) that reasonably accurate heating rates can be obtained by using this approach.

The most difficult part of applying this technique is computing the inviscid surface streamline paths and the metric coefficient associated with the spreading of the streamlines. DeJarnette and Davis[44] calculated the streamlines as the lines of steepest descent (also called simplified streamlines), emanating from the stagnation point. These streamlines are consistent with the Newtonian concept that a fluid particle loses its normal component of momentum upon striking a body surface. DeJarnette and Hamilton[21,32] developed a simple method for calculating streamlines from a known pressure distribution. However, this approach has proven difficult to apply,[45] unless the surface pressures and geometry can be described analytically.[9] More success has been achieved when the streamline information is derived from complete three-dimensional inviscid flowfield calculations.[46-49]

Axisymmetric Heating Rates

Numerous finite-difference and integral methods have been developed to accurately calculate axisymmetric laminar, transitional, and turbulent boundary layers.[50] However, since the major concern here is three-dimensional heating rates, approximate two-dimensional and axisymmetric heating rates that have been modified for three-dimensional effects will be considered in this section.

Laminar Heating Rates

Lees[51] showed that the pressure-gradient effect on laminar heating rates is small for cold walls. Therefore, the compressible flat plate solution, using the Mangler transformation for bodies of revolution, can be applied locally to obtain a relatively simple expression for the heating rate.[44] Beckwith and Cohen[34] modified the heating rate of Lees to account for equilibrium air conditions.

More accurate heating rates can be calculated for noncold walls by using methods that base boundary-layer properties on the local pressure and Eckert's reference enthalpy.[16,52] In the heating rate expressions, Edwards and Cole[16] used the local Reynolds number based on distance from the stagnation point, while Zoby, Moss, and Sutton[52] used the momentum thickness Reynolds number and a Levy-Lees type transformation for surface distance.

Turbulent Heating Rates

Numerous turbulent heating techniques and procedures have been developed. Earlier methods included Spalding-Chi, Sommer and Short, Van Driest I, and Van Driest II. (These methods are described in Ref. 53.) Many of them are modifications of flat plate methods that obtain heating rates from the local skin-friction coefficient by some form of Reynolds analogy. Combinations of turbulent skin-friction techniques and Reynolds analogy factors have been compared for flat plates and cones with experimental data by Hopkins and Inouye.[53] They concluded that the Van Driest II analysis with a Reynolds analogy factor of 1 is best. Zoby and Graves[54] compared results from several techniques with both wind tunnel and flight data. They found that the best overall agreement was obtained when the Spalding-Chi method used the modified Colburn Reynolds analogy with the Reynolds number based on distance from the peak heating point and when the Schultz-Grunow method used the same analogy with the Reynolds numbers based on the distance from the transition location. More recently Zoby, Moss, and Sutton[52] developed a method based on Eckert's reference enthalpy relation to calculate both laminar and turbulent heating rates. This method has been found to compare well with both wind tunnel and flight data. Several investigators have found that generally more accurate heating rates can be obtained by using Reynolds analogy and skin-friction theories or procedures based on momentum-thickness Reynolds number rather than using the Reynolds number based on surface distance from the stagnation point or leading edge.

Transition Heating Rates

The location of transition from laminar to turbulent boundary layers is still difficult to predict. However, Harris and Blanchard[50] used a relatively simple expression to locate the end of transition based on a known beginning of transition. Transitional heating rates can be calculated reasonably well by using an exponential distribution of the weighting of local laminar and turbulent values given by Dhawan and Narasimha.[55]

Gas Models

Earlier heating analyses used a perfect gas or Cohen's equilibrium air model.[56] It was found that Cohen's model gave poor predictions of the density and coefficient of viscosity at some flight conditions, although the product was reasonably accurate.[33] The equilibrium air model of Zoby and Moss[57] was found to give better accuracy with very little increase in computational effort.[33] An even more accurate prediction of equilibrium air properties, with some increase in computational effort, can be obtained by using the correlation of Tannehill and Mugge,[58] which is a curve fit of the Mollier diagram.

Three-Dimensional Applications

Earlier design studies were usually conducted using adaptations of simple two-dimensional heating techniques like those applied in the MINIVER code.[59] These techniques are

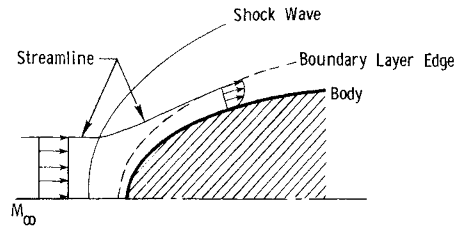

Fig. 1 Entropy-layer swallowing for axisymmetric flow.

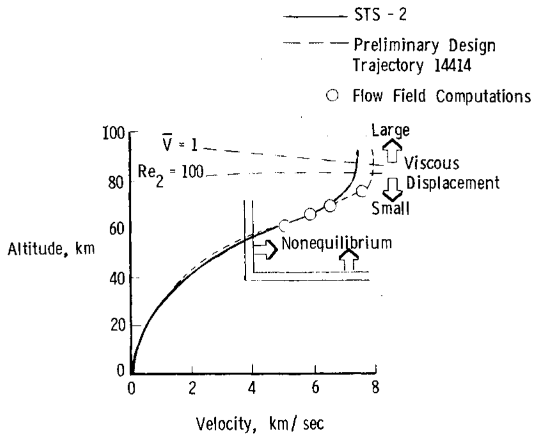

Fig. 2 Flow regimes.[10]

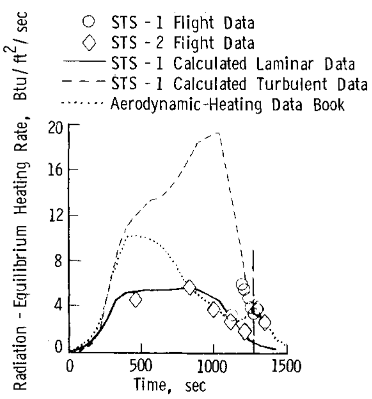

Fig. 3 Comparison of estimated and measured flight heating rates at $Z/L = 0.5$ on the lower surface centerline of Shuttle Orbiter.[19]

not sufficiently accurate for current vehicle designs. This section gives some results from more accurate techniques for estimating heating rates. For brevity, the number of applications is limited to only a few.

Windward Centerline Methods

Tangent Cone

Laminar and turbulent radiation equilibrium heating rates on the windward centerline of Space Shuttle flights STS-1 and STS-2 are shown in Fig. 3 (from Ref. 16). The method of Edwards and Cole[16] was used to calculate the radiation equilibrium heating rates. For the time span from about 450 to 1350 s, α varied from about 42 deg down to 26 deg, M decreased from about 25 to 8, and the altitude dropped from 240,000 to 130,000 ft. Figure 3 shows that the aerodynamic heating data book overpredicted the heating rates for flight times less than 800 s. The method of Edwards and Cole gave results that compared well with flight data.

Equivalent Axisymmetry Body

Adams et al.[60] proposed that an appropriate axisymmetric body at zero angle of incidence that modeled the windward centerline flowfield over the Shuttle at an angle of attack could be determined. Zoby[61] used this concept to determine hyperboloids that matched the windward centerline of the Shuttle at angles of attack from 25 to 45 deg. Heating rates were calculated by combining his modified Maslen inviscid method[20] with the reference enthalpy method of Zoby, Moss, and Sutton.[52] Variable entropy at the edge of the boundary layer was calculated by interpolating the inviscid flowfield at a distance equal to the boundary-layer thickness away from the wall. Figure 4 (from Ref. 61) shows that this technique predicts laminar heating rates that compare well with wind tunnel experimental data extrapolated to flight conditions and with the methods of Refs. 46 and 47, which use the axisymmetric analog, finite-difference axisymmetric boundary-layer and the three-dimensional inviscid flowfield methods. This figure illustrates the effect of entropy-layer swallowing on heating rates. Zoby's method[61] has also been shown to predict laminar and turbulent heating rates along the windward centerline. These compare well with experimental data, viscous shock layer calculations, and flight data from Space Shuttle flights STS-1[62] and STS-2.[63]

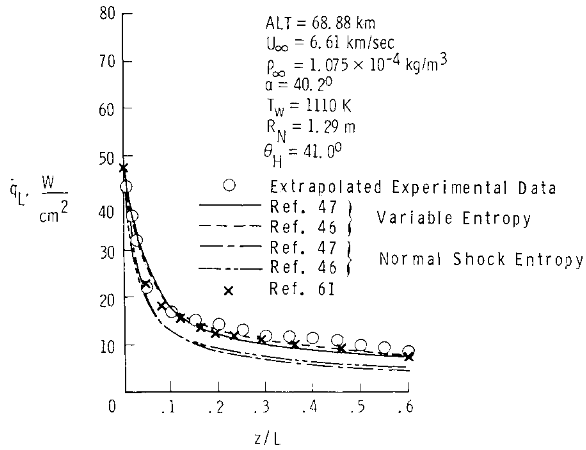

Fig. 4 Comparison of predicted shuttle windward-ray heating distributions at entry condition for maximum heating design trajectory.[61]

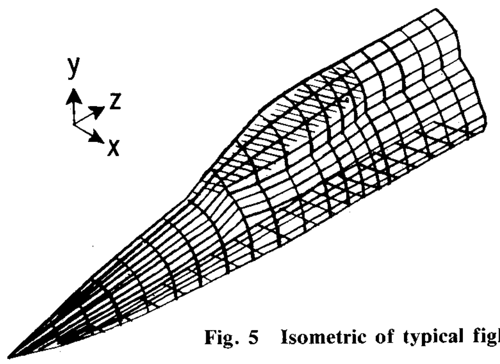

Fig. 5 Isometric of typical fighter forebody.[9]

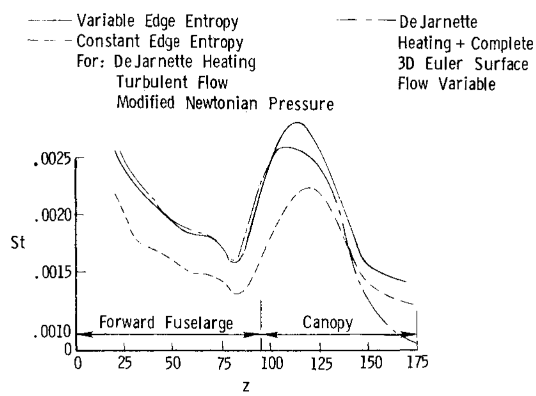

Fig. 6 Effect of variable-edge entropy on Stanton number distribution for the upper-surface centerline of the fighter forebody at Mach 2.5, $T_w/T_s = 0.8$, and 75,000 ft altitude.[9]

Methods Using the Axisymmetric Analog

Modified Newtonian Pressure Distribution

The earlier method of DeJarnette and Hamilton[32] was extended to include entropy-layer swallowing effects.[21] The inviscid solution was coupled with the boundary-layer solution by using a modified Maslen method to calculate the shock-wave shape corresponding to each inviscid surface streamline. A mass balance technique was used to determine where the streamline at the edge of the boundary layer crossed the shock wave. The original method of Ref. 21 required iterations at each marching step along a streamline to determine the boundary-layer edge properties. That method was replaced with a noniterative technique described by Fivel.[64] This method was found to predict laminar and turbulent heating rates reasonably well over the windward regions of sphere-cones,[21] blunted elliptical cones,[65] and the Space Shuttle.[32] The method for calculating the inviscid surface streamlines from the pressure distribution was replaced by the simplified streamline technique,[44] which developed a method for calculating heating and surface temperatures of vehicles for computer-aided design studies.[33] Although less accurate, this technique allows for rapid computations of heating rates and surface temperatures throughout the trajectory of a vehicle.

Jacocks and Kneile Euler Code

The heating method of DeJarnette and Hamilton[21] was modified by Varner et al.[9] to calculate heating rates from both the modified Newtonian pressure distribution and the three-dimensional Euler code of Jacocks and Kneile.[22] Heating rates were calculated for the typical fighter forebody shown in Fig. 5. Figure 6 shows the effect of variable entropy on turbulent Stanton numbers along the upper surface centerline at $M_\infty = 2.5$. This figure shows that entropy-layer swallowing effects are significant even at $M_\infty = 2.5$. It also shows that heating calculations using the modified Newtonian pressure distribution compared well with those generated by the Euler code. However, caution must be exercised when calculating inviscid surface streamlines from surface pressure data. DeJarnette[45] found that surface streamlines calculated using this technique were unsatisfactory unless the surface pressure distribution was modeled accurately.

HALIS Code

Hamilton et al.[49] used the HALIS code[25] to calculate the inviscid flowfield and the method of Zoby, Moss, and Sutton[52] to calculate laminar and turbulent heating rates on three-dimensional vehicles. Although the heating rates could be obtained from a finite-difference solution of the full axisymmetric boundary-layer equations,[46,47] it was found that accurate results could be obtained more easily, and with much less computational effort, from the approximate heating relations developed in Ref. 52. Variable boundary-layer edge entropy was calculated by interpolating the inviscid flowfield at a distance equal to the boundary-layer thickness away from the wall. This method has been used to accurately predict heating rates for simple shapes, such as a spherically blunted cone, and for more complex shapes, such as the Shuttle Orbiter, for a variety of wind tunnel and flight conditions.[49]

Figure 7 (from Ref. 49) compares laminar, transitional, and turbulent heating rates with wind tunnel data along the windward symmetry plane of the Space Shuttle at $M_\infty = 8$ and $\alpha = 35$ deg. The calculated data are in excellent agreement with the experiment. Comparisons with flight data are

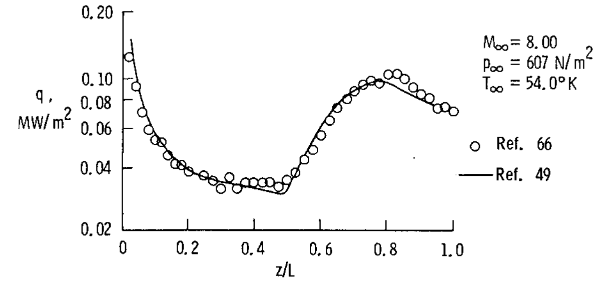

Fig. 7 Heating distribution along windward-symmetry plane of 0.0175 scale Shuttle Orbiter for $\alpha = 35°$, $Re_\infty = 13.1 \times 10^6$/m.[49]

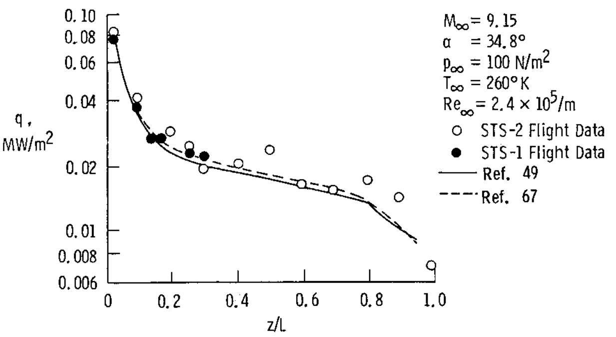

Fig. 8 Heating distribution along windward-symmetry plane of full-scale Shuttle Orbiter.[49]

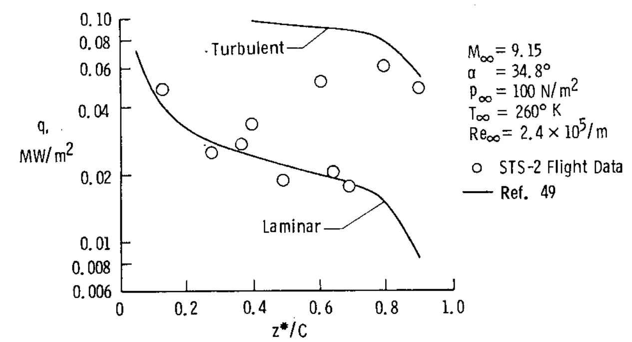

Fig. 9 Streamwise distribution of heating on wing of full-scale Shuttle Orbiter at $2x/b = 0.5$.[49]

given in Fig. 8 (from Ref. 49). Although there is considerable scatter in the flight data, the calculations agree reasonably well. In addition to the method of Ref. 49, Fig. 8 shows that Hamilton's swept-cylinder method[67] compares well with the data.

The primary advantage of the method of Ref. 49 over most other methods is that it can be easily used to make calculations for regions off the windward-symmetry plane. A comparison of calculated heating with flight data for the "mid-wing" location ($2x/b = 0.5$) is presented in Fig. 9 (from Ref. 49). Two theoretical curves are shown, one for laminar flow and one for turbulent flow. The turbulent calculations were made for a starting transition at $z/L = 0.2$. At first glance the flight data appear to behave very strangely; initially, they are laminar, then transitional, then laminar, then transitional, then laminar again, and finally fully turbulent near the trailing edge of the wing. This behavior is quite easily explained when it is realized that the flow has traveled along different streamlines at different cord locations on the wing. This can be seen from the inviscid surface streamline pattern for this case that is shown in Fig. 10 (from Ref. 49). Thus, the flow along one streamline can be transitional, or

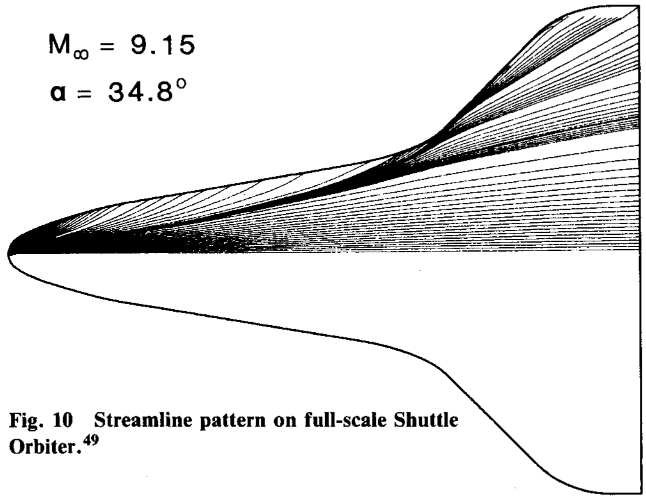

Fig. 10 Streamline pattern on full-scale Shuttle Orbiter.[49]

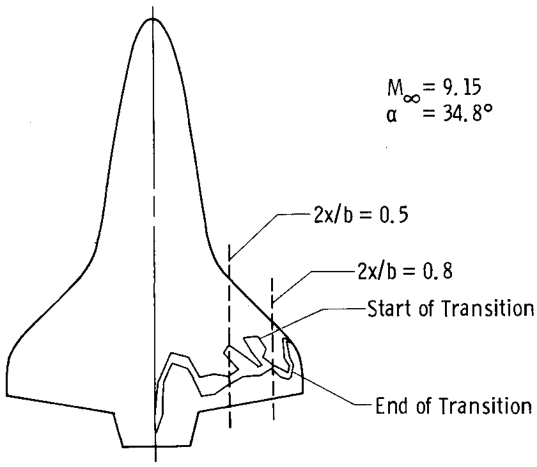

Fig. 11 Transition front on Shuttle Orbiter wing from STS-2 flight.[49]

even turbulent, while the flow on adjacent streamlines remains laminar. In fact, this behavior fits the transition pattern observed for this case as shown in Fig. 11 (from Ref. 49).

Concluding Remarks

Several approximate techniques that can be used to calculate laminar and turbulent heating rates along the windward plane of symmetry of three-dimensional vehicles were reviewed. From hypersonic Mach numbers down to values as low as 2.5, variable entropy at the edge of the boundary layer was found to have a significant influence on the calculated heating rates. All of the techniques reviewed for the windward centerline compared reasonably well with experimental data.

For regions off the windward plane of symmetry, fewer methods are available. The use of the axisymmetric analog to replace fully three-dimensional boundary-layer methods has proven to be accurate and requires considerably less computational effort. Inviscid surface streamlines calculated from prescribed pressure distributions appear to work well for some body shapes. However, for winged three-dimensional vehicles, difficulties have been experienced when using that technique. More success has been achieved when streamlines are calculated using the surface velocity components obtained from a complete three-dimensional inviscid flowfield code. The HALIS code has the advantage of being capable of calculating the inviscid flowfield for shuttle-like vehicles at large angles of attack. Heating rates along inviscid surface streamlines can be calculated rapidly and accurately with the approximate relations of Zoby, Moss, and

Sutton. This method requires much less computational effort than do the finite-difference solutions of the axisymmetric boundary-layer equations. The method of Hamilton et al. was found to calculate accurate heating rates over the wing as well as the body of the Space Shuttle. Simpler techniques using the modified Newtonian pressure distribution (but with streamlines calculated by the "simplified" method) are useful for computer-aided design applications where heating rates are needed throughout the trajectory of a vehicle.

There are several areas where fast, reliable, approximate methods are still needed. The methods reviewed here account for entropy-layer swallowing effects, but the interaction of the boundary layer on the inviscid solution is neglected. This viscous interaction effect is important in some flight profiles. None of the approximate methods reviewed here included chemical nonequilibrium gas models or catalytic wall effects. For low Reynolds number flows, approximate viscous shock-layer and noncontinuum flow methods are needed. Also, approximate heating methods for separated flows need attention. Finally, the age-old problem of predicting the location of transition still remains.

Acknowledgments

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