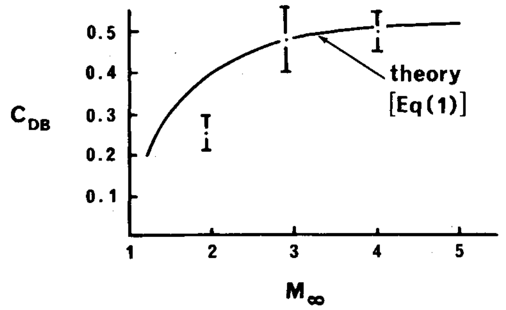

Fig. 3 Ball drag coefficient deduced from experiments. Comparison with simple theory.

Table 1 Ball open ($\theta = 0°$) and ball closed drag coefficients

	Freestream Mach number, M_∞		
Ball position	1.94	2.88	4.00
Open, C_{DO}	0.390	0.250	0.210
Closed, C_{DC}	0.645	0.738	0.714
$C_{DB} = C_{DC} - C_{DO}$	0.255	0.488	0.504

The experiments also show that the dependency of drag on Mach number reverses as the projectile passage changes from an open to a blocked condition. Since the projectile configuration changes in this evolution from a relatively slender body to that of a blunt body, the dominant drag mechanism may be expected to become one which is largely blunt body form drag dominated by the pressure rise across the normal bow shock wave. (The ratio of pressures across a normal shock is about 4 times higher at Mach 4 than at Mach 2). This also explains the relatively large benefits in drag reduction at the higher Mach numbers where opening of the passage results in the removal of a relatively strong bow shock.

As the ball moves, the change in the total drag acting upon the projectile is largely attributable to the change in the drag force acting upon the ball itself. There is some validity, therefore, to the approximation that the drag on the ball at a non-zero value of θ is given by the difference between the total projectile drag at that ball position and the projectile drag with unblocked flow ($\theta = 0$). The drag on the ball in the fully closed ($\theta = 90°$) position is especially useful in prediction of the motion of the ball during projectile flight and, with the above approximation, this is given by:

$$C_{DB} = C_{DC} - C_{DO}$$

In Ref. 1 this was approximated using normal-shock static pressure on the face of the ball and negligible base pressure recovery. Under these assumptions, the ball drag coefficient is given as:

$$C_{DB} = 4\bar{r}^2 (1 - 1/M_\infty^2) / (\gamma + 1)$$

where $\bar{r}$ is the ratio of ball hole radius to the projectile radius and γ is the isentropic exponent. Using the parameters pertinent to these experiments ($\bar{r} = 0.573$ and $\gamma = 1.4$) the drag coefficient for the fully-closed ball is given theoretically as:

$$C_{DB} = 0.55 (1 - 1/M_\infty^2) \quad (1)$$

This expression is compared with the experimental values (Table 1) in Fig. 3.

Drag-coefficient uncertainty bands are also shown in Fig. 3 for reference purposes. These bands are thought to be conservative in that they are based upon estimates of maximum errors in the measurements of drag force and freestream static pressure at the ball angle giving the greatest scatter of data. The initial series of measurements was conducted at Mach 2.88, and the smaller uncertainties at the other Mach numbers reflect improvements in experimental apparatus (particularly in mounting the projectile) and techniques. Given the experimental uncertainties the formula recommended above appears to be well-supported by the data. The departure of theory from experiment at low Mach numbers is not unexpected, since the validity of the theoretical prediction is sensitive to differences between normal-shock downstream static and stagnation pressures. These differences become insignificant at the higher Mach numbers. In addition, at the lower supersonic Mach numbers, it is expected that an improved theory would require considerably more sophistication in modelling the recirculation regions both upstream and downstream of the ball.

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Space Shuttle Heating Analysis with Variation in Angle of Attack and Catalycity

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Introduction

THE Shuttle has the aerodynamic capability of re-entering the Earth's atmosphere at steeper or shallower angles of attack for different cross-range requirements of landing. Depending on the angle of attack of the Orbiter, the Shuttle could have different stagnation point locations and, thus, different effective nose radii, since the local radius of curvature changes rapidly on the Shuttle nose. This, in turn, would affect the nonequilibrium/equilibrium characteristics of the flowfield as well as the surface heating. A study to analyze this effect, therefore, is desirable.

For a surface having large catalytic efficiency (or, surface recombination rate, $k_{w,i}^*$), the heating during the re-entry is increased[1-3] by the heat of dissociation released during the recombination of oxygen and nitrogen atoms (mostly the oxygen atoms) at the surface. The thermal protection system

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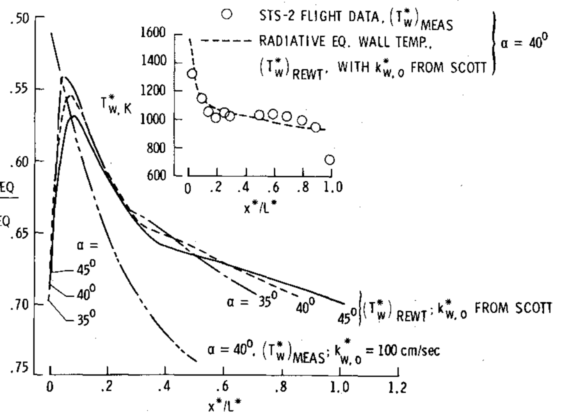

Fig. 1 Heating reduction due to nonequilibrium chemistry along windward centerline of the Orbiter for various angles of attack and surface conditions; altitude = 74.98 km, $U_\infty^* = 7.2$ km/s (STS-2 flight trajectory).

of the Shuttle has been designed conservatively by assuming the wall to be fully catalytic ($k_{w,i}^* = \infty$). In Refs. 2 and 3, values of $k_{w,o}^*$ between 80 and 200 cm/s have been used to generate nonequilibrium predictions to compare with flight data. It is, therefore, of interest to know how sensitive the surface heating is to the variation in catalytic efficiency of the thermal protection system.

The changes in nonequilibrium character of the viscous shock-layer flow during the re-entry period may be evaluated by analyzing the time history of the boundary-layer edge quantities and surface heat transfer. Some indication of the state of outer inviscid flow can be obtained from the boundary-layer edge quantities whereas the nature of the boundary-layer flow may be established by analyzing the surface heat transfer. This information may be useful in obtaining the surface heating from the boundary-layer methods and/or the heating correlations[2] for flowfields where the more detailed viscous shock-layer methods could not be applied.

Analysis

The flowfield structure and the surface heating rates along the Shuttle windward centerline plane have been obtained by seeking solutions of the reacting two-dimensional viscous shock-layer equations over the matching hyperboloid at zero incidence.[4] The details of the chemical kinetics, thermodynamic, and transport properties employed here are given in Refs. 3 and 5. The Prandtl and Lewis numbers are assumed to be 0.7 and 1.4, respectively.

The numerical method developed by Davis[6] for solving the nonreacting viscous shock-layer equations for the stagnation and downstream flow was applied by Moss[5] to reacting multicomponent mixtures. The same method, with the details given in Ref. 5, has been used here and is, therefore, not described.

Angle-of-Attack Variation Effects

Figure 1 shows the effect of variation in angle of attack on heating along the windward centerline of the Orbiter. This figure contains results for STS-2 flight conditions at an altitude of 74.98 km, where the nonequilibrium effects are greatest. Corresponding to this flight condition, the nominal value of the angle of attack α is 40 deg. The effect of the angle-of-attack variation shown in this figure has been obtained by using the radiative equilibrium wall temperature (REWT) based on a constant surface emissivity of 0.9. This assumption is used since it is a fairly good representation of the actual flight data and, thus, provides a consistent method for computing the nontrajectory points. The figure inset in Fig. 1 provides a comparison between the STS-2 flight data and the radiative equilibrium wall temperature. Predictions of the wall heat transfer based on REWT or T_w^* obtained from the STS-2 data are essentially the same.

Figure 1 gives the ratio of predicted nonequilibrium, finite catalytic wall heating to predicted equilibrium heating as a function of vehicle length along the windward centerline for three values of the angle of attack, assuming Scott's expression[8] for wall catalycity. A ± 5-deg variation from the 40-deg nominal value of angle of attack does not appear to affect this ratio much. Also shown in the same figure is the curve for 40-deg angle of attack by employing a constant value of 100 cm/s for the oxygen recombination rate at the wall, $k_{w,o}^*$. This curve has a monotonically increasing trend with the vehicle length. This trend (with the constant value of wall catalycity) is basically due to the nature of the gas-phase chemistry, i.e., the chemistry relaxes toward the equilibrium state as the flow travels further downstream. For the cases assuming Scott's expression for wall catalycity, a different trend is noted in the forward region. This may be the result of the highly temperature-dependent characteristic of surface catalycity (see Fig. 2 of Ref. 7) overtaking the influence from gas-phase chemistry. In the aft portion of the vehicle, considerable reduction in heating is noticeable when using the Scott's expression for the surface recombination rate parameter in place of a constant value. For this region, assuming that the gas-phase chemistry has the same influence in the two cases, the temperature-dependent Scott's values for $k_{w,o}^*$ are much lower than the constant value of 100 cm/s and, therefore, the wall heat-transfer values based on the Scott's expression are lower. It may be mentioned here that employing a constant value of $k_{w,o}^*$ is not physically realistic. A temperature-dependent expression similar to that of Scott's, but less strongly dependent, might be more realistic.

Actual values of the predicted nonequilibrium surface heating rates along the windward centerline based on the Scott's expression for wall catalycity are shown in Fig. 5 of Ref. 7. The heat-transfer rate, $q_{w,NEQ}^*$, like the ratio, $q_{w,NEQ}^*/q_{w,EQ}^*$, is not significantly (less than 8%) affected by a ± 5-deg variation in the angle of attack about the nominal value of 40 deg.

The nonequilibrium-to-equilibrium surface heating ratio and the nonequilibrium heating-rate distribution at a lower altitude of 47.67 km also indicate that the effect of a ± 5-deg angle-of-attack variation is not significant as given in Ref. 7. Once again, there is more reduction in the value of the nonequilibrium surface heating when Scott's expression is used in place of a constant value of 100 cm/s for the recombination rate parameter $k_{w,o}^*$.

Wall Catalycity Effects

The effect of variations in the surface recombination rate or wall catalycity on heat transfer at an altitude of approximately 75 km is given in Fig. 2. The predictions are obtained for different body locations along the Orbiter windward centerline by using the STS-2 flight conditions and a range of constant values of the recombination rate parameter, including the noncatalytic ($k_{w,o}^* = 0$) and fully (or equilibrium) catalytic ($k_{w,o}^* = \infty$) wall values. The data from the STS-2 flight are also given for the same body locations. Surface heating is most sensitive to the recombination rate parameter near the nose ($x^*/L^* \approx 0.025$) and at an altitude of 75 km, where the nonequilibrium effects are most significant for the STS-2 flight. As pointed out in the Introduction, values of $k_{w,o}^*$ used by the other investigators[2,3] for predicting the STS-2 surface heating range between 80 and 200. For this range of variation in $k_{w,o}^*$, the surface heat transfer changes by about 12% at $x^*/L^* \approx 0.025$ and 75 km in altitude. At higher or lower altitudes the change in surface heat transfer is less when $k_{w,o}^*$ varies from 80 to 200 cm/s. However, values of $k_{w,o}^*$ larger than 200 are required for predicting the surface heating at 61-km altitude for $x^*/L^* \approx 0.025$ as may be noticed from Fig. 8c of Ref. 7. In fact, no single value of the recombination rate parameter can be used to predict heating

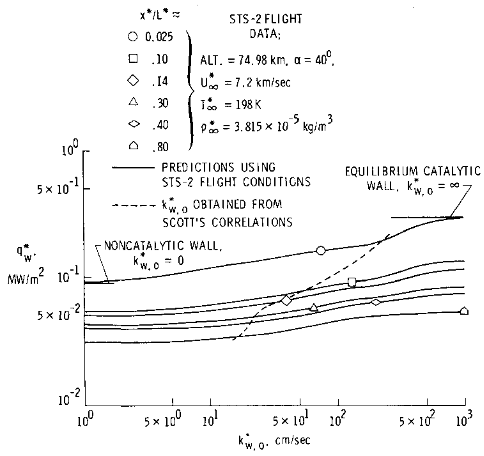

Fig. 2 Surface heat transfer as a function of oxygen recombination rate at various locations along the Orbiter windward centerline.

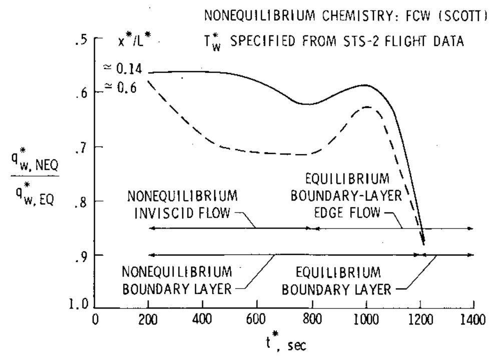

Fig. 3 Time history of heating reduction due to nonequilibrium chemistry at various locations along the Orbiter windward centerline for STS-2 flight.

comparable with the flight data for all locations along the Orbiter windward centerline for a given altitude and the values differ from altitude to altitude. The recombination rate $k^*_{w,0}$ obtained from the Scott's correlations, is also shown in Fig. 2.

At the lower altitude of 48 km (as shown in Fig. 3d of Ref. 7), q^*_w becomes relatively insensitive to the changes in $k^*_{w,0}$ in the range of 80-200 cm/s. For this altitude the flight data suggest the surface heating to be closer to the equilibrium predictions (with $k^*_{w,0} = \infty$). There does not appear to be any reduction in heating due to the nonequilibrium chemistry at 48-km altitude. A maximum reduction of 49% in heating due to the nonequilibrium chemistry is obtained at an altitude of about 75 km in the nose region ($x^*/L^* \approx 0.025$) when the flight data are compared against the equilibrium value. At other body locations along the windward centerline, as well as at higher and lower altitudes, the reduction in heating due to the nonequilibrium chemistry is much less.

Time History of Boundary-Layer Edge Quantities and Nonequilibrium Heating

The nonequilibrium effects at the boundary-layer edge appear to become less significant after the entry time of about 800 s (or below an altitude of 65 km) as shown in Fig. 9 of Ref. 7. The tangential velocity U^*_e, temperature T^*_e, and the constituent air species $(C^*_i)_e$ at the boundary-layer edge are predicted reasonably close from the equilibrium as well as nonequilibrium chemistry after this time. The boundary-layer flow, however, is still in chemical nonequilibrium as suggested by the time history of wall heat-transfer ratio, $q^*_{w,\mathrm{NEQ}}/q^*_{w,\mathrm{EQ}}$, of Fig. 3. Figure 3 also indicates that beyond 1200 s (or below 50 km), both the boundary-layer edge quantities and the boundary-layer flow appear to approach the equilibrium chemistry very rapidly. It may be mentioned here that the results of Fig. 3 are based on Scott's surface recombination rates[8] and may require some modification when more appropriate values of the recombination rates are available.

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Thermodynamic Considerations in Bipropellant Blowdown Systems

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Nomenclature

A = surface area
c = specific heat at constant volume
D = propellant tank diameter
g = gravity constant
Gr = Grashof number
h = heat transfer coefficient
J = Joule proportionality factor
k = thermal conductivity
m = mass
Nu = Nusselt number
P = total tank pressure
Pr = Prandtl number
$\dot{Q}$ = heat transfer rate
R = gas constant
Re = Reynolds number
T = temperature
V = ullage volume

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