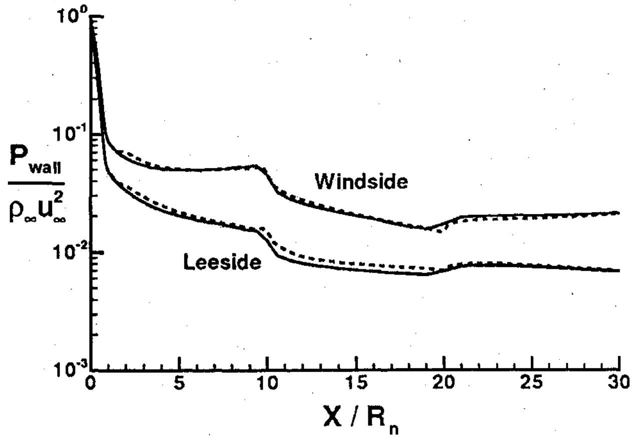

Fig. 2 Symmetry-plane surface pressures: $M = 28, R_n = 0.1524$ m, fully catalytic wall, $\alpha = 5$ deg; ——, LAURA and - - - -, LAURA–UPS.

Concluding Remarks

Two proven, existing solvers have been combined for the aerothermodynamic solution of hypersonic air flowfields with finite-rate chemistry. The robustness of the TLNS solver LAURA has been joined with the speed of the PNS solver UPS. The class of vehicles to which the method is applicable are blunt-nosed configurations with afterbody flowfields free of streamwise-separated flow. The method offers the potential benefits of obtaining efficient solutions with second-order accuracy in the crossflow planes, while requiring only a fraction of the computer time and memory that a full-body LAURA solution would require.

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T. C. Lin
Associate Editor

Reevaluation of Flight-Derived Surface Recombination-Rate Expressions for Oxygen and Nitrogen

R. N. Gupta*
NASA Langley Research Center, Hampton, Virginia 23681-0001

Introduction

FOR the efficient design of future transportation systems, an accurate description of the re-entry aerothermal environment about a spacecraft is important. Since the maximum heating pulse for such a lifting body is likely to occur at high altitudes, surface recombination rates for the dissociated air species (in a flowfield with nonequilibrium chemistry) are required in the evaluation of spacecraft heating. Unfortunately, direct measurement of surface recombination rates (on surface coatings) is not possible from ground or flight tests. These rates are generally inferred from the surface heat-flux measurements.

In the study of Ref. 1, an expression for oxygen recombination coefficient was obtained for the Space Shuttle tile material based on the heat-flux measurements (obtained from the early Shuttle flight test missions). Unfortunately, some discrepancies in the version of the viscous shock-layer (VSL) code[2] used in Ref. 1 were discovered during the studies of Refs. 3–5. The present investigation accounts for those discrepancies in the reevaluation of surface recombination-rate expressions for oxygen and nitrogen from the flight data.

Analysis

Procedures used to develop the present expressions for the surface reaction-rate coefficients are the same as those employed in Ref. 1. The surface recombination-rate values that give the best fit to flight heating rates at STS-2 entry conditions for altitudes of 71.29, 74.98, and 77.91 km are correlated as a function of the surface temperature in an Arrhenius form. The reaction-rate values are incorporated in a detailed VSL code[3,4] for the finite reacting flowfields to compute laminar heating rates for the partially catalytic surfaces. A brief outline of the procedure, as well as the relative merits of the flowfield method for computing the flow in the Shuttle windward symmetry plane, is given in Ref. 1. Only the changes implemented in the code employed for the study of Ref. 1 are provided here.

First, the shock boundary condition for enthalpy is correctly specified[5] and referenced to a temperature of 298 K. Next, a coding error in implementation of the constant Prandtl number option in the code[6] used in Refs. 1 and 2 has been rectified. Also, new transport and thermodynamic properties[7] are employed in place of those from Refs. 8 and 9. Finally, a grid-independent solution is obtained by employing a stepsize ΔS of 0.1 in the streamwise direction for the entire Shuttle length and a variable stepsize $\Delta\eta$ (with a value of 1×10^{-4} at the surface) in the direction normal to the surface. References 1 and 2 used similar values of $\Delta\eta$; however, ΔS varied from 0.1 in the stagnation region increasing through the nose region to a value of 3.0 at the tail end of Shuttle. Although the influence of each change on the predicted values of Ref. 1 is not quantified here, the effect of ensuring axial grid independence, in general, was the largest in developing the correlations. This is because the surface distributions of pressure and heating vary considerably in the nose

Received Oct. 5, 1995; revision received Nov. 24, 1995; accepted for publication Dec. 14, 1995.

*Senior Research Engineer, Aerothermodynamics Branch, Gas Dynamics Division. Associate Fellow AIAA.

region. After developing the correlations, however, larger step sizes (similar to those of Refs. 1 and 2) may be used.

Following the procedure identical to that used in Ref. 1, the following forms of the oxygen and nitrogen surface recombination-rate expressions in centimeter per second are obtained:

$$k_{W,\mathrm{O}} = 7276.0\sqrt{T_W}\exp(-8600/T_W) \tag{1}$$

$$k_{W,\mathrm{N}} = 6.8\sqrt{T_W}\exp(-2219/T_W) \tag{2}$$

or, in terms of the energy-transfer recombination coefficient,

$$\gamma_{\mathrm{O}} = 8.0\exp(-8600/T_W) \tag{3}$$

$$\gamma_{\mathrm{N}} = 0.007\exp(-2219/T_W) \tag{4}$$

Equation (2) for the nitrogen surface recombination rate as well as Eq. (4) for the recombination coefficient have the exponential factor similar to the one contained in the expression based on the ground-measured data.[10] However, the preexponential factors in Eqs. (2) and (4) has been fine tuned to obtain the best agreement between the predicted and the flight-measured heating rates. The fine tuning is done by treating the preexponential factor as an independent variable and varying it parametrically from its base value (obtained from the ground-measured data). The base value is used to obtain the oxygen recombination-rate equation [Eq. (1)] and the recombination coefficient expression [Eq. (3)]. This procedure is adopted by keeping in view that the surface recombination of nitrogen atoms occurs at temperatures higher than those for the oxygen atoms. Therefore, the oxygen recombination rates play a major role in controlling the diffusive heat-flux component near the cooler Shuttle surface ($800 < T_W < 1400$). Further, it is difficult to quantify separately the effects of the recombination rates of atomic oxygen and nitrogen on surface heating for the flight case (in air). Hence, only the preexponential factor has been fine tuned rather than obtaining an entirely new expression for the nitrogen surface recombination rate.

Results and Discussion

Figure 1 shows the catalytic energy recombination coefficient γ_{O} obtained from Eq. (3) as a function of surface temperature. Also shown in this figure are Scott's[10] ground-measured and extrapolated values from arc-jet flow, as well as the values obtained in Ref. 1. The surface temperatures of interest for the Shuttle applications are about 1400 K (computed radiative equilibrium values in the nose region) or less. Present recombination rates are larger for temperatures greater than about 1200 K and smaller for temperatures less than 1200 K as compared to the results of Ref. 1. Further, the present rates are somewhat larger but with approximately the same slope as those extrapolated from Scott's[10] measurements for the entire temperature range shown in Fig. 1.

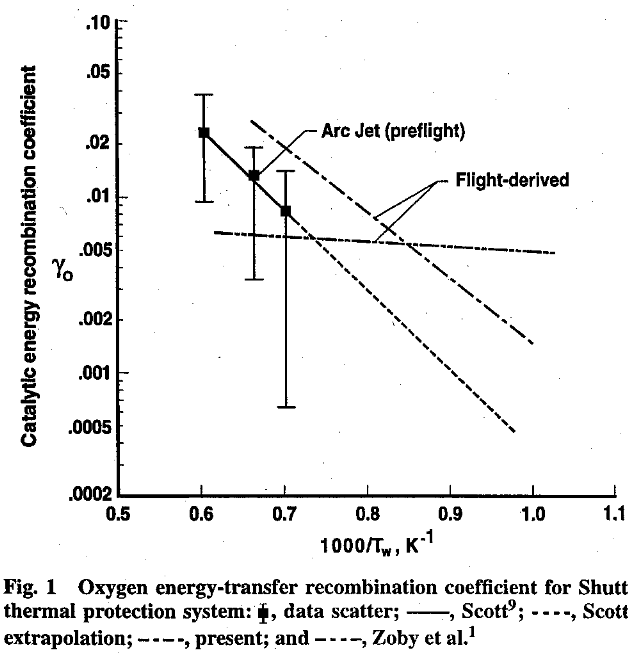

Fig. 1 Oxygen energy-transfer recombination coefficient for Shuttle thermal protection system: ⫲, data scatter; ——, Scott[9]; - - - -, Scott's extrapolation; – – – –, present; and – - – -, Zoby et al.[1]

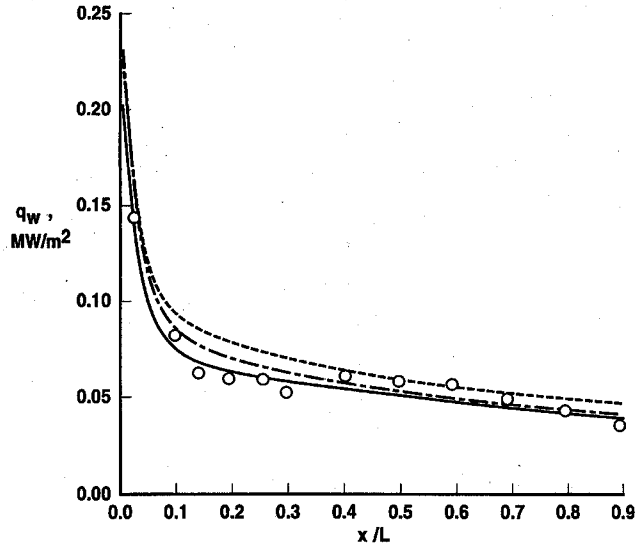

Fig. 2 Comparison of predicted and experimental heating rates for different values of the surface energy-transfer coefficients; STS-2, altitude 77.91 km: U_∞ = 7.42 km/s; M_∞ = 26.3; $\rho_\infty = 2.335 \times 10^{-5}$ kg/m³; α = 40.2 deg; ○, experimental data; ——, present coefficients; - - - -, Zoby et al.[1] coefficients; – – – –, Scott[9] coefficients; and 1440 K $\le T_W \le$ 813 K.

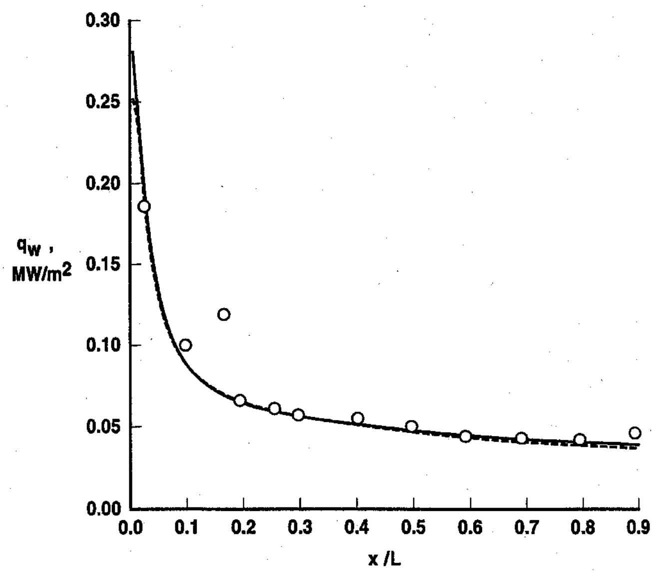

Fig. 3 Predictions of surface heating rates from VSL2D and VSL3D codes using the present recombination rate expressions; STS-2, altitude = 60.56 km: U_∞ = 4.99 km/s; M_∞ = 15.7; $\rho_\infty = 2.621 \times 10^{-4}$ kg/m³; α = 42.0 deg; $\gamma_O = 8e^{-8600/T_W}$ and $\gamma_N = 0.007e^{-2219/T_W}$; ○, experimental data; ——, VSL2D; and - - - -, VSL3D.

The predicted surface heat-transfer rates with the present oxygen and nitrogen recombination coefficients are shown in Fig. 2. These heating rates (obtained with the current code) are in better agreement with the data as compared to those based on existing[1,10] recombination-rate expressions. Note that the present correlations are determined (similar to Ref. 1) from three selected STS-2 entry conditions ranging from 71 to 78 km altitude, where the nonequilibrium chemistry effects are dominant. At 60.56-km altitude (see Fig. 3), the agreement between the predicted and measured heating values is quite good, even though the flow is considerably outside the range of altitudes chosen for the correlation of Eqs. (3) and (4). Therefore, this comparison should provide sufficient validity for the application of present surface recombination rates for flow conditions, which have considerably different flow properties than those employed to obtain these rates.

Using Scott's[10] rates in the present code gives better results (within 15% of the experimental data) than those presented in Ref. 1.

The better heating rates presently computed using Scott's extrapolated values for γ_0 are a result of the correct implementation of constant Prandtl number option in the VSL code and the axial grid sensitivity studies as mentioned earlier. It may be indicated here that apart from the requirement of extrapolation of the Scott's measured values (for lower Shuttle surface temperatures away from the stagnation region), surface heating predictions based on Scott's rates approach equilibrium levels (and the flight data) at lower altitudes (<55 km) slower as compared with those based on the present rates.

To evaluate the code independency of the present recombination-rate coefficients, surface heating rates have been calculated by employing these coefficients in another VSL code[11] (VSL3DNQ). Present predictions, as well as those from the code of Ref. 11, using the recombination rate expressions of Eqs. (3) and (4), are almost indistinguishable and in excellent agreement with the experimental data as shown in Fig. 3. This comparison provides a test to judge the applicability of present surface recombination rates with other flowfield codes. No effort has been made in this work to compare the present rates with the original rates of Kolodziej and Stewart[12] and Stewart et al.[13] Stewart has also recently reevaluated these rates, and his new results have not been published yet.

Concluding Remarks

Existing oxygen and nitrogen surface recombination coefficient expressions (obtained from heating data from the Shuttle flights) have been reevaluated in view of the corrections, changes, and grid resolution study carried out with a recently developed partially coupled VSL code. The changes include correcting code errors and incorporation of transport and thermodynamic properties from a recent study that are considered to be more accurate than the previous values. Similar to the earlier study, the new oxygen and nitrogen recombination-coefficient expressions are determined from the STS-2 surface heat-flux measurements at altitudes above 70 km along the windward symmetry plane. The current oxygen recombination-coefficient expression is shown to be more temperature dependent than the previous expression. Quite good comparison is obtained between the predicted and measured heating values over the range of STS-2 entry conditions from approximately 78 to 61 km altitude. The previous recombination coefficient values result in larger heat-flux predictions as compared with the flight data, with a maximum difference of about 20% at an altitude of 60.56 km.

Acknowledgments

The author would like to thank K. P. Lee for help in obtaining the present results and R. A. Thompson for the VSL3D results used for comparison. E. V. Zoby provided helpful suggestions during the preparation of this manuscript.

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T. C. Lin
Associate Editor

Effect of Transpiration Cooling on Nozzle Heat Transfer

F. Chen,* W. J. Bowman,† and R. Bowersox‡
U.S. Air Force Institute of Technology, Wright–Patterson Air Force Base, Ohio 45433-7755

Introduction

RENEWED interest in liquid propellant space launch vehicles has sparked research to improve the performance of these systems.[1] Nozzle heat transfer and material thermal limitations remain limiting factors in the performance of modern rocket engines. Actively cooled rocket nozzles allow the high-combustion temperatures necessary for high performance while maintaining the structural integrity of the nozzle. Several methods of active cooling have been employed to address this problem, including regenerative, film, and transpiration cooling.

Regenerative cooling involves pumping a liquid through channels surrounding the outside of the combustion chamber. Film cooling involves injecting gas or liquid through one or several discrete holes in a nozzle wall to establish a protective film on the surface. Transpiration cooling is essentially the limiting case of film cooling because it involves pushing gas or liquid uniformly through an area of porous wall material.[2] Ideally, it can be thought of as an infinite number of film cooling ports with zero distance between them. This cooling has been successfully used for cooling injector faces in the upper stage engine (J-2) of the moon launch vehicle and the Space Shuttle main engines, but it has not been used for cooling the thrust chamber or nozzle regions of large rocket engines.[2]

It is expected that a heat transfer rate reduction similar to or better than film cooling can be realized with decreased flow disturbances, indicating that transpiration cooling could be a more attractive method than film or regenerative cooling. Therefore, this investigation proposes to study the effects of transpiration cooling in a supersonic nozzle on heat transfer. Compared to an earlier study,[1] a wider range of blowing ratios were examined.

The objective of this research was to investigate the differences between flow over a porous wall with blowing and a nonporous wall with no blowing. In particular, the effect of blowing ratio on the nozzle heat transfer characteristics were studied. These objectives

Received Oct. 23, 1995; revision received Dec. 13, 1995; accepted for publication Dec. 20, 1995. This paper is declared a work of the U.S. Government and is not subject to copyright protection in the United States.

*Research Assistant, Aero/Astronautical Engineering Department.

†Associate Professor, Aero/Astronautical Engineering Department. Senior Member AIAA.

‡Assistant Professor, Aero/Astronautical Engineering Department. Member AIAA.