Calculated Heating on the Afterbody Nozzle of a Hypersonic Aircraft Concept

N. A. Talcott Jr.* and J. L. Hunt*
NASA Langley Research Center, Hampton, Va.

Introduction

HYPERSONIC scramjet-powered aircraft offer attractive potential solutions to future civil and military needs (Ref. 1). Current concepts use the entire lower fuselage of the aircraft as part of the propulsion system. The vehicle forebody provides inlet precompression and the lower aft-end of the vehicle acts as a high-expansion-ratio external nozzle. Extreme care must be exercised in designing the external nozzle to assure optimum thrust and lift while minimizing adverse pitching moments that could lead to large aircraft trim penalties or instabilities.

Recent developments using substitute gas techniques to simulate high-temperature, real-gas scramjet exhaust flows are reported in Ref. 2. Limited nozzle heat transfer data have been obtained in the Grumman detonation tube, but fundamental questions still exist as to what the absolute level of aerodynamic heating will be in flight. This Note presents the results of a parametric analysis to estimate the heating levels on the afterbody nozzle of a typical hypersonic research concept. The aircraft concept examined is approximately 24.4 m (80 ft) long with a 5.6 m (18 ft) long planar exhaust nozzle incorporating a 20-deg initial expansion angle (Ref. 3).

Analysis Methods

Conditions representative of cruise at freestream Mach numbers of 5 to 7 and freestream dynamic pressures ranging from 2.4×10^4 to 7.2×10^4 N/m^2 (500 to 1500 lb/ft^2) were examined. Two methods were used to calculate the heating rate distribution on a 20-deg afterbody scramjet two-dimensional nozzle for stoichiometric hydrogen-air ratios: 1) the correlation method of Spalding and Chi (Ref. 4), and 2) the compressible, ideal-gas, nonsimilar boundary-layer code of Price and Harris (Ref. 5). The explicit, finite-difference, ideal-gas nozzle code of Ref. 6 which incorporates the Spalding and Chi correlation was used to compute the nozzle's inviscid boundary-layer edge conditions for input to both heat-transfer prediction methods. Earlier calculations with the code of Ref. 7 using equilibrium and frozen real-gas expansions have shown that the assumption of an average-γ ideal gas does not significantly alter the flow (Ref. 2).

Results and Discussion

A boundary-layer reference length must be selected to compute the nozzle heating rate. Since the airframe-integrated scramjet swallows the forebody boundary layer, the appropriate reference length for the nozzle boundary layer is not known a priori. For this Note the boundary-layer reference length is assumed to start in the scramjet combustor immediately downstream of the fuel injectors. For the configuration studied, this corresponds to a reference length of approximately 1.07 m (3.5 ft), where the combustor is approximately 1.83 m (6 ft) in length. The sensitivity of the reference length on the heating rate distribution will also be discussed.

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*Aero-Space Technologist, Hypersonic Aerodynamics Branch, High-Speed Aerodynamics Division.

Figure 1a shows a comparison between heating-rate data and the turbulent boundary-layer theory of Ref. 5 for a Mach 6, $q=7.2\times10^4$ N/m^2 (1500 lb/ft^2) flight case. A laminar comparison was also made using Ref. 5 and the absolute levels of the heating-rate distribution clearly indicate that the boundary layer is turbulent. These results are presented to give confidence to calculating the heating-rate distributions by the methods outlined above. The H_2/air combustion products data were obtained in the Grumman detonation tube at test conditions matching flight enthalpy, chemistry, and Reynolds number (Ref. 7). $\dot{Q}_{ref}$ and Y_{ref} are taken at the detonation tube combustor exit plane. For this test the reference length was 1.37 m (4.5 ft).

Figure 1b also contains a comparison of the two methods for a Mach 7, $q=2.4\times10^4$ N/m^2 (500 lb/ft^2) flight case, where $\dot{Q}_{ref}$ and Y_{ref} are taken at the scramjet combustor exit plane. The Spalding-Chi distribution shows a good agreement with the turbulent distribution of Ref. 5 except for a higher initial level $\dot{Q}_{ref}$. Since the Spalding-Chi computation incorporated in Ref. 6 requires minimal user input and half the computational time of that required by Ref. 5, this method was chosen to estimate the heating rates for this study.

A parametric examination of the effects of Mach number, reference length, and wall temperature was conducted, and cases similar to that shown in Fig. 1b were computed for Mach 5, 6, and 7 flight.

Figure 2 shows the heating-rate distributions obtained by varying flight Mach number for a fixed dynamic pressure,

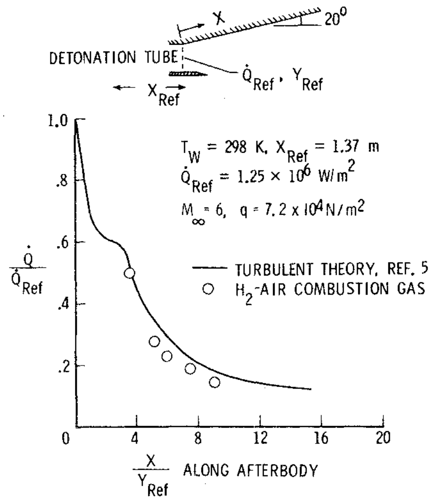

Fig. 1a Heating-rate data compared with turbulent theory, $M_\infty=6$.

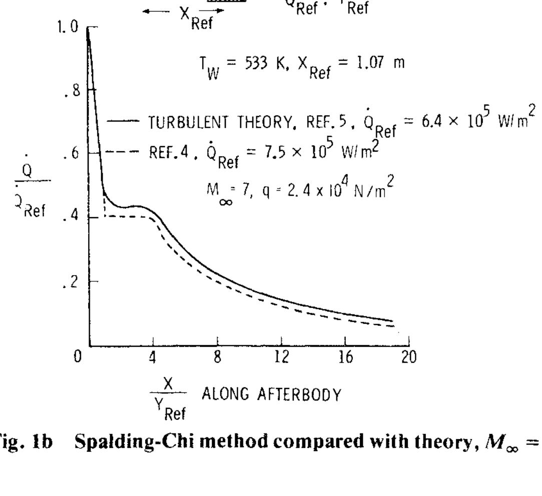

Fig. 1b Spalding-Chi method compared with theory, $M_\infty=7$.

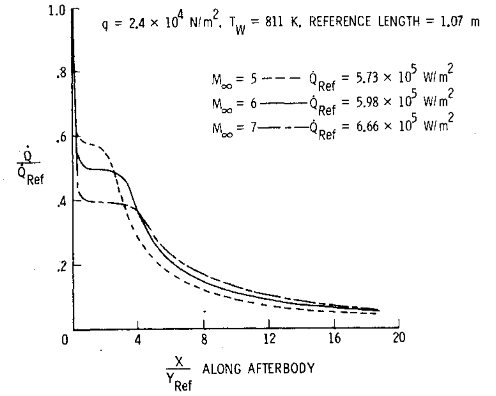

Fig. 2 Heating rate distributions for Mach 5, 6, and 7 flight.

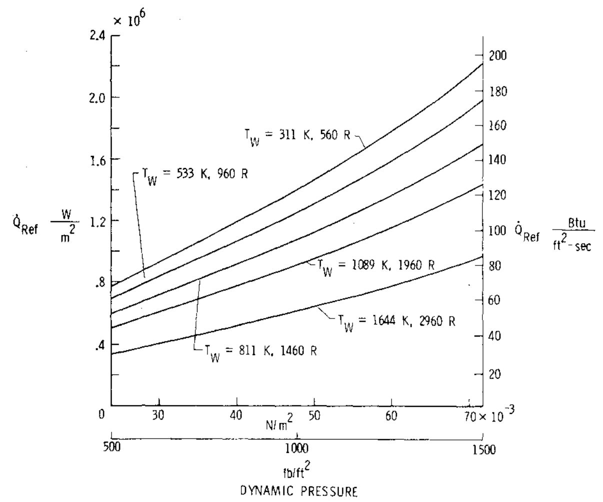

Fig. 3 Initial heating-level variation due to changes in dynamic pressure, $M_\infty = 6$.

wall temperature, and reference length. The results also showed that variations in wall temperature and reference length shifted the distributions by less than 5% for a fixed flight Mach number. Therefore, $\dot{Q}_{ref}$ appears to be a good normalizing factor for this study. The absolute effects of changing the dynamic pressure, wall temperature, and reference length can now be examined.

The effect on the afterbody nozzle wall initial heating level of varying the reference length from 0.76 m to 1.52 m for a Mach 6 flight condition [$q = 2.4 \times 10^4$ N/m^2 (500 lb/ft^2)] was examined. For a given wall temperature, a 12% variation in $\dot{Q}_{ref}$ over the stated reference length range was obtained. This variation could be approximated by a one-sixth power law.

Figure 3 presents the variation in $\dot{Q}_{ref}$ due to changes in the dynamic pressure for Mach 6 flight where a reference length of 1.07 m has been assumed. Curves are shown for five wall temperatures that span the range of possible wall values. The actual wall temperature will be a function of the thermal protection system selected for the vehicle and may well be a function of location along the afterbody as well. The tripling of the initial heating level shows that, as would be expected, dynamic pressure is the major parameter affecting the nozzle heating rate. [This variation (Fig. 3) could be approximated by a 0.98 power law.]

Conclusions

A study has been conducted to estimate the heating levels on the external nozzle of a scramjet/airframe-integrated research aircraft. A parametric examination of the effects of Mach number, reference length, and wall temperature showed that the heating-rate distributions are independent of reference length and wall temperature. The initial heating rates obtained for a Mach 6 flight case are in the 3-8 × 10^5 W/m^2 (30-70 Btu/ft^2-sec) range.

Underlying the entire study is the question of nozzle boundary-layer formation and growth and what corresponding reference length should be used in the computation. Our results have shown that reference length is not the dominant factor setting the heating levels and we have tried to bound the actual length. Further work will be required to obtain a better understanding of the combustor exit boundary layer before more detailed calculations of the rates can be obtained.

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Errata

Remarks on Thin Airfoil Theory

Rajendra K. Bera*
National Aeronautical Laboratory, Bangalore, India
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It has been brought to my attention by N. H. Kemp that there were sign errors in some of the equations in the above Engineering Note. The corrections are as follows:

$$J_{n+1} + J_{n-1} = 2\cos\phi J_n + (2/n)[1-(-1)^n] \tag{11}$$

$$J_1 = -\int_0^\pi \cot\left(\frac{\theta+\phi}{2}\right)d\theta = -2\log\cot(\phi/2) \tag{15}$$

$$A = \frac{4}{\sin 2\phi} - 2\frac{\log\cot(\phi/2)}{\sin\phi} \tag{16}$$

In Eqs. (17), (19), and (20) the right-hand sides should be multiplied by (-1) so that the final result, Eq. (20), reads

$$\int_0^\pi \frac{\sin n\theta d\theta}{(\cos\theta - \cos\phi)} = \frac{4\sin n\phi}{\sin 2\phi} - 2\frac{\sin n\phi}{\sin\phi}\log\cot\frac{\phi}{2} - [1-(-1)^n]\frac{1}{n\cos\phi} \tag{20}$$

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*Scientist, Aerodynamics Division.