Stagnation-Point Radiative Heating Relations for Earth and Mars Entries

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Stagnation-point radiative heating rate expressions are presented for use in air and an approximate Martian atmosphere consisting of 97% CO_2 and 3 % N_2. Thermochemical equilibrium is assumed throughout. The flight conditions and body dimensions that are modeled are representative of both manned and unmanned missions to Mars and return to Earth. Comparisons between the heating rates computed using the expressions presented here and independent computations yielded maximum differences of about 20 to 30%.

Nomenclature

a = exponent (see Eq. 1)
b = exponent (see Eq. 1)
C = constants (see Eq. 1)
C_{H_r} = radiative heat-transfer coefficient, dimensionless
E = Earth atmosphere
M = Mars atmosphere
p = shock-layer pressure, atm
$\dot{q}_r$ = radiative heat transfer at wall, W/cm^2
r_n = hemispherical nose radius, m
V = flight velocity, m/s
ρ = freestream density, kg/m^3

Introduction

DIRECT atmospheric entries at Mars and on return to Earth subject vehicles to significant amounts of radiative heating from the incandescent shock-layer gases in the stagnation-point region. For example, vehicles returning from Mars will enter the Earth's atmosphere at velocities of 12 km/s, or greater. At these speeds, the heating from radiation will exceed the boundary-layer convective heating at the stagnation point. Several computer codes have been developed to calculate the stagnation-point radiative heating in air as a function of the flight conditions.[1-3] The codes are complex since the chemical composition must be calculated and the emissions of the individual species have to be computed and integrated over all wavelengths. Also, the emission of radiation as the gas travels from the shock to the body makes the flow nonadiabatic. Computing radiative heating of the body is, therefore, a complex procedure and heating rates are usually tabulated as a function of flight speed, freestream density, and body radius. Since the radiative heating is a highly nonlinear function of the foregoing parameters, the triple interpolation that must be made to apply the tabulated values is awkward and computationally inefficient.

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In this paper, analytic expressions, supplemented by one easily interpolated table, are presented for calculating the stagnation-point radiative heating in air and in the Martian atmosphere. Thermochemical equilibrium is assumed throughout and only the "cold-wall" heating occurring in the absence of ablation is considered. A multiband radiative model was used. Also presented is a comparison with an independent computation of stagnation-point radiative heating in air.

Entries into the Martian atmosphere, which consists of 95.6% CO_2, 2.7% N_2, and 1.6% Ar,[4] occur at lower speeds than on return to Earth with peak velocities below 10 km/s. However, the stagnation-region shock layer contains CO and CN molecules that are intense radiators, and the radiative heating can be comparable to the convection for large bodies. The dominant source of radiation (80–90%) is CO since its concentration is three orders of magnitude greater than that of CN. Because most of the CO emission is in the ultraviolet part of the spectrum,[5] self-absorption is very important and has been accounted for. An analytic expression, again supplemented by one simple table, is presented. The expression is valid for body sizes ranging from unmanned vehicles to the large aerobrakes needed for manned configurations.

The heating relations that are presented can be efficiently used for aerobraking mission studies in which many repetitive calculations are performed. More detailed analyses, which take into account radiative transport in boundary layers containing ablation products and possible thermochemical nonequilibrium, will be required to design mass-efficient heat shields.

Brief Description of Radiation Model

The radiation model is based on the assumption that thermochemical equilibrium exists. The assumption of equilibrium is somewhat supported by the observation that, during manned return from Mars, the shock-layer pressures at the time of maximum radiative heating range from about 0.2 to 0.4 atm.[6] The radiation model is coupled to an inviscid flowfield code that computes the nonadiabatic flow along the stagnation streamline of a hemisphere.[2] For nonspherical configurations, equivalent hemispherical nose radii can be found that will give the same shock standoff distances in an adiabatic flow. The multiband radiation model of Nicolet[7,8] is used. The model includes atomic line, continuum, and molecular band transitions and covers the wavelength spectrum beginning at 0.0775 μm. Extensive comparisons are presented in Ref. 9 of the calculated radiative intensities with values measured in shock tubes, ballistic ranges, constricted arcs, and two well-instrumented flight tests. The radiative heating was calculated

in air for flight velocities from 9 to 18 km/s, freestream densities of 1.84×10^{-3} kg/m^3 to 6.66×10^{-5} kg/m^3, or altitudes from about 30 to 72 km, and nose radii from 0.05 to 5 m. These calculated values were tabulated, but have not yet been published.

The stagnation-point, equilibrium radiative heating in the Martian atmosphere was calculated assuming a composition of 97% CO_2 and 3% N_2, which is nearly the same as the values previously listed that were measured by the Viking Mars landers.[4] Cases were computed for velocities ranging from 6 to 9 km/s over a large range of altitudes and for nose radii from 1 to 23 m; these results have been tabulated, but not yet published.

Next, expressions for the stagnation-point radiative heating will be presented and their range of applicability discussed.

Heating Rate Expressions

It was found that a heating rate expression having the form

$$\dot{q}_r = Cr_n^a \rho^b f(V) \tag{1}$$

could be used to represent the calculated values over a range of flight conditions representative of Mars missions. In Eq. (1), C is a constant that depends on the atmosphere, r^n is the hemispherical nose radius in m, ρ is the freestream density in kg/m^3, and $f(V)$ are tabulated values that are functions of flight velocities V and the atmospheric composition. The exponents a and b can be either constants, or functions of ρ and also V, in m/s. For air, the following values are used, where $\dot{q}_r$ is in W/cm^2:

$$C = 4.736 \times 10^4$$

$$a = 1.072 \times 10^6 \, V^{-1.88} \rho^{-0.325}$$

$$\text{if} \quad 1 \leq r_n \leq 2, \qquad a \leq 0.6$$

$$\text{if} \quad 2 < r_n \leq 3, \qquad a \leq 0.5$$

$$b = 1.22 \tag{2}$$

For air, $f_E(V)$ is listed in Table 1; linear interpolation can be used.

The values given by Eqs. (1) and (2) apply in the speed range from 10 to 16 km/s and for freestream densities from 6.66×10^{-5} kg/m^3 to 6.31×10^{-4} kg/m^3, or altitudes of about 72 to 54 km, respectively. The nose radii can vary from 0.3 to 3 m. (Note that $a \leq 1$ must always be met.) For manned vehicles returning from Mars, peak radiative heating typically occurs at altitudes between 64 and 68 km and at speeds from 0.9 to 0.95 of the entry velocity.[6] These flight conditions are well within the range of Eqs. (1) and (2). A comparison is shown in Table 2 of the rounded-off percentage differences between the radiative heating as calculated using Eqs. (1) and (2) and the previously mentioned tabulated values. The differences are largest at the lowest altitude of 54 km and vary from being 20% too high to 18% too low. At the important 66-km altitude, the heating rates are off by a maximum of 8%. The differences shown in Table 2 are within the uncertainties of the radiative heating calculations.[9]

For the Martian atmosphere, also with $\dot{q}_r$ in W/cm^2, the following constants are used:

$$C = 2.35 \times 10^4$$
$$a = 0.526$$
$$b = 1.19 \tag{3}$$

and $f_M(V)$ is given in Table 1. The constants given by Eq. (3) apply over the speed range from about 6.5 to 9 km/s and for freestream densities from 10^{-4} kg/m^3 to 10^{-3} kg/m^3. The corresponding altitudes vary from about 51 to 30 km, respectively. The above flight conditions cover the range over which peak radiative heating occurs for manned vehicles.[10] The nose radii can vary from 1 to about 23 m. The differences between previously mentioned tabulated radiative heating values and those calculated using Eqs. (1) and (3) are shown in Table 3. The values given by the equation vary from being 12% too high to 17% too low. Although the comparison presented in Table 3 is somewhat limited, the most important flight conditions are covered.

Table 1 Radiative heating velocity functions for Earth and Mars

V, m/s	$f_E(V)$	V, m/s	$f_M(V)$
9000	1.5	6000	0.2
9250	4.3	6150	1.0
9500	9.7	6300	1.95
9750	19.5	6500	3.42
10,000	35	6700	5.1
10,250	55	6900	7.1
10,500	81	7000	8.1
10,750	115	7200	10.2
11,000	151	7400	12.5
11,500	238	7600	14.8
12,000	359	7800	17.1
12,500	495	8000	19.2
13,000	660	8200	21.4
13,500	850	8400	24.1
14,000	1065	8600	26.0
14,500	1313	8800	28.9
15,000	1550	9000	32.8
15,500	1780	—	—
16,000	2040	—	—

Table 2 Percent difference (rounded off) between heating rates from Eqs. (1) and (2) and computed values

ρ, kg/m^3 Approx. altitude	r_n(m)	V, km/s 10	11	12	14	16
6.659×10^{-5}	0.3	1	10	13	15	11
(72 km)	1.0	10	12	9	3	−8
	3.0	0	11	18	9	−5
$1.471 / 10^{-4}$	0.3	4	6	5	6	0
(66 km)	1.0	0	0	0	0	−6
	3.0	2	6	8	4	0
3.059×10^{-4}	0.3	8	3	7	10	7
(60 km)	1.0	−6	−3	0	5	−2
	3.0	−7	−1	3	5	5
6.314×10^{-4}	0.3	10	9	15	20	20
(54 km)	1.0	−8	−1	5	16	17
	3.0	−18	−6	0	11	13

Table 3 Percent difference (rounded off) between heating rates from Eqs. (1) and (3) and computed values

ρ, kg/m^3 Approx. altitude	r_n(m)	V, km/s 7	8	9
1.000×10^{-4}	1.0	9	9	15
(51 km)	2.3	−3	−6	−9
	10.0	−1	−1	−12
3.162×10^{-4}	1.0	−1	−8	−14
(41 km)	2.3	2	−6	−17
	10.0	5	6	2
1.000×10^{-3}	1.0	12	—	—
(30 km)	2.3	12	—	—
	10.0	−1	—	—

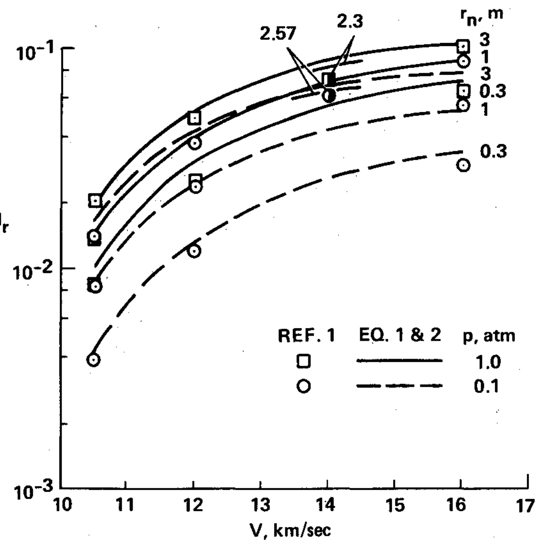

Fig. 1 Radiative heat-transfer comparison in air.

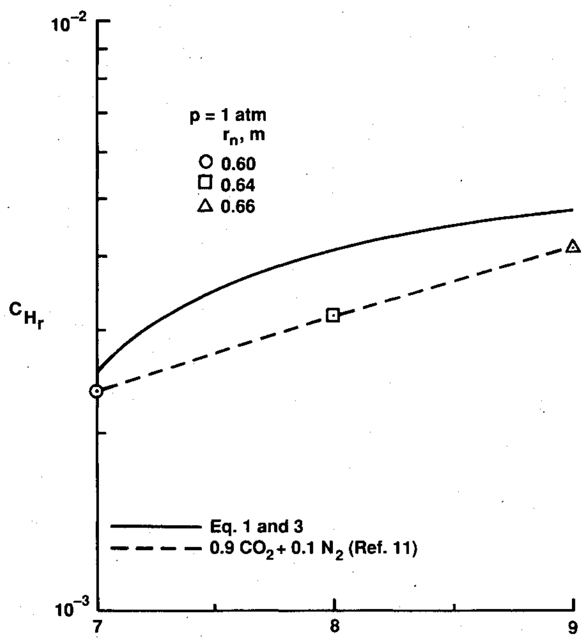

Fig. 2 Radiative heat-transfer comparison in two $CO_2 - N_2$ mixtures.

Comparison with Other Methods

The stagnation-point radiative heating expression given above is now compared with an independent computation. A comparison with values from Ref. 1 for radiative heating in air is shown in Fig. 1. The radiative heat-transfer coefficient, defined as

$$C_{H_r} = \dot{q}_r/(1/2\rho V^3) \tag{4}$$

is shown as a function of flight velocity for shock-layer pressures of 0.1 and 1 atm, and for nose radii from 0.3 to 3 m. The average difference between the heating rates from Eqs. (1) and (2) and the values from Ref. 1 is 8.5%. The maximum difference is 23% higher than Ref. 1; the largest underprediction is 12%. The differences between heating rates from Eqs. (1) and (2) and the values from Ref. 1 are well within the current uncertainties in the knowledge required for computing radiative heating rates.

For Mars, the only similar, independent computations of radiative heating are those by Page and Woodward[11] for a 90% CO_2 and 10% N_2 mixture. In addition, the values listed in Ref. 11 are for higher pressures, or densities, and smaller nose radii than the range of applicability of the expressions presented here. Nonetheless, a comparison with values from Ref. 11 is shown in Fig. 2 to lend credence to the present results. The present formulation yields values that are within 10–30% of those listed in Ref. 11. Note, again, that this relatively good agreement occurs despite differences in the composition of the gas mixtures and the fact that Eqs. (1) and (3) are used beyond their intended range of applicability.

Concluding Remarks

Expressions for stagnation-point radiative heating rate are presented for use in air and an approximate Martian atmosphere consisting of 97% CO_2 and 3% N_2. Thermochemical equilibrium is assumed throughout. The flight conditions and body dimensions that are modeled are representative of both manned and unmanned missions to Mars and return to Earth. Comparisons between the heating rates computed using the expressions presented here and independent computations yielded a maximum difference of about 23% for air. The difference is within uncertainties currently inherent in the computation of stagnation-point equilibrium radiative heating rates.

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