PRELIMINARY STUDY OF AERODYNAMIC HEATING AT HYPERSONIC VEHICLE 14-X

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Abstract. *Vehicles flying at high speeds undergo a process of temperature increase in the wall and surrounding. Knowledge of the phenomenon, as well as determining the intensity, is of paramount importance to the material specification to be used for its manufacture, its thermal protection and also prevent telemetry problems generated by ionization of the air. Since this is that this work fits, proposing a analytical methodology for calculating for aerodynamic heating of hypersonic vehicle 14-X and also estimating the heating generated in the fly regime.*
Keywords*: aerodynamic heating, hypersonic,14-X.*

1. NOMENCLATURE

C_p = specific heat at constant pressure;
C_v =specific heat at constant volume;
M=Mach number;
P = pressure;
P_r=Prandtl number;
$\dot{q}_c$ = *rate heat*;
R= radius of curvature in the stagnation point;
r =recovery factor;
S_t = Stanton Number;
T_{aw} = Temperature of the adiabatic wall;
T_w = wall temperature;
T_r = Monaghan reference temperature;
u = speed in the direction flow;

Greek Symbols

ρ= specific mass;
γ = ratio of specific heats;
μ= dynamic Viscosity.

Subscripts

s = condition outside boundary layer;
sp =stagnation point;
w = wall;
∞ = flight freestream.

2. INTRODUCTION

The 14-X is the project of Brazilian Air Force, developed in the Division of Aerothermodynamics and Hypersonic, of the Institute for Advanced Studies and consists of the study, project and manufacture of the a technological demonstrator of a reactor-combustion ramjet supersonic.

In the contemporary space research, the challenges posed by aerospace activities are the full knowledge of the environment and of phenomena founds in the hypersonic fly of the aerospace vehicles. In particular the atmosphere offers resistance to displacement of the vehicle, converting a large portion of kinetic energy in thermal energy. Thermal energy generated results in large heat flux causing a substantial temperature increase in the atmosphere surrounding the outer surface of the vehicle the order of hundreds degrees Celsius. The knowledge of phenomenon is important to design a system of thermal protection that keep the wall temperature and in the indoor temperature of space vehicle at acceptable levels.

3. METHODOLOGY

Preliminarily, considers the following assumptions:

1) Vehicle flying with zero angle of attack (α=0º);
2) Air behaves as perfect gas (Υ =cte);
3) Inviscid flow (effects of boundary layer are neglected);
4) Laminar Flow;
5) Regions formed by flat plates and semi-cylinder;
6) Coordinate system is aligned with the flat surface;
7) No dissociation occurs.

Based on hypotheses shown above, the methodology used in this study considers a flow state of two-dimensional on the vehicle and three distinct regions under different conditions (conical shock, oblique shock and region that does not occur shock waves) for calculation the aerodynamic heating, see Fig. 1.

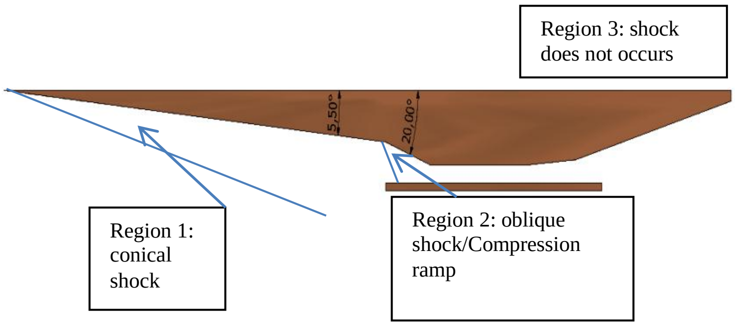

Figure 1: sketch of the vehicle regions.

3.1 Mathematical Equations

The mathematical modeling used the following sets of equations for rate heat:

3.1.1 Stagnation Point

The analytical equation for the rate heat in the Stagnation Point was developed by Fay and Riddell (1958), as shown below:

$$\dot{q}_c = \frac{0.57}{(P_r)^{0.6}} * (\rho_s \mu_s)^{0.4} (\rho_w \mu_w)^{0.1} C_p (T_s - T_w) \left(\frac{du}{dx}\right)^{0.5} \tag{1}$$

and:

$$\left(\frac{du}{dx}\right) = \frac{1}{R} * \sqrt{\frac{2(P_s - P_\infty)}{\rho_s}} \tag{2}$$

3.1.2 Off of Stagnation Point

In the region outside the stagnation point was used the theory of the Lees (1956):

$$\frac{\dot{q}_c}{\dot{q}_{sp}} = \frac{2\theta sin\theta\left\{\left[1-\frac{1}{\gamma M_\infty^2}\right]\cos^2\theta + \frac{1}{\gamma M_\infty^2}\right\}}{[D(\theta)]^{0.5}} \tag{3}$$

$$D(\theta) = \left[1 - \frac{1}{\gamma M_\infty^2}\right]\left[\theta^2 - \frac{\theta sin4\theta}{2} + \frac{1-cos4\theta}{8}\right] + \frac{4}{\gamma M_\infty^2}\left[\theta^2 - 2sin2\theta + (1-cos2\theta)/2\right] \tag{4}$$

And θ is the angle between stagnation point and the desired point.

3.1.3 Flat Plate

The mathematical equation of the problem follows the simplified analytical model developed by R. Michel (1970), where the rate heat is calculated as:

$$\dot{q}_c = \rho_s * u_s * c_p * S_t * (T_{aw} - T_w) \tag{5}$$

$$T_{aw} = T_s * \left[1 + r * \frac{\gamma-1}{2} * M_s^2\right] \tag{6}$$

$$S_t = \frac{m}{Re_x^n} * \Delta \tag{7}$$

$$T_r = T_s + 0{,}54 * (T_w - T_s) + 0{,}16 * (T_{aw} - T_s) \tag{8}$$

$$\Delta = \left(\frac{T_r}{T_s}\right)^{\frac{5n}{2}-1} * \left[\frac{T_s+S}{T_r+S}\right]^n \tag{9}$$

For a flat plat, under laminar regime, “*m*” and “*n*” are respectively 0,411 e ½. Wall temperature was set at 300ºK.

3.2 Calculating the properties

The flow properties do not disturbed were calculates using the program Hypersonic Airbreathing Propulsion (HAP) developed by Heiser (1994), see Fig. 2.

Figure 2: properties calculated in the HAP.

The shock conical properties were calculated using a computer code developed in MATLAB by Giannino (2012). The results obtained through the code (ConShock.m) were also validated in HAP. The properties of oblique shock were calculated by Anderson Jr. (2006) according to the theory of oblique shock waves.

4. RESULTS AND DISCUSSIONS

With the information show above was created a computational routine (Heating14Xpontual.m) in MATLAB to calculate the rate heat at a set of points of each region and for each flight regime. The input dates are altitude and flying speed was: h_1=23000m, v_1=1782m/s; h_2=34000m; v_2=2142m/s. The results obtained for rate heat in the regions, shown in figure 1, were plotted in figures 3, 4, 5, 6, 7 and 8. In the figure 9, the data obtained from the program were assigned on the geometry of the vehicle, using the software ANSYS.

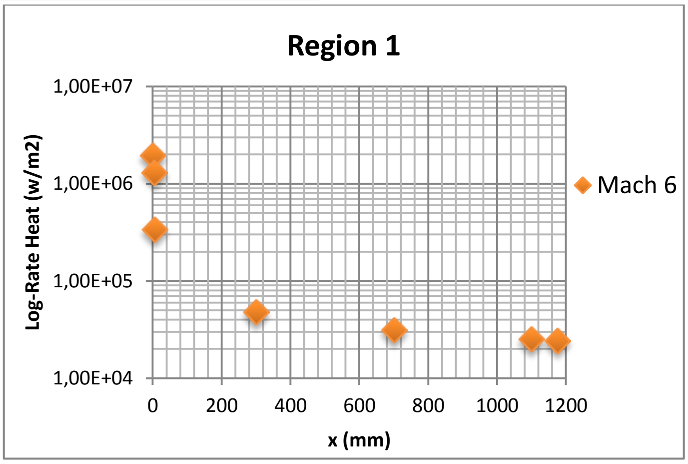

Figure 3: rate heat in the region 1 for Mach 6.

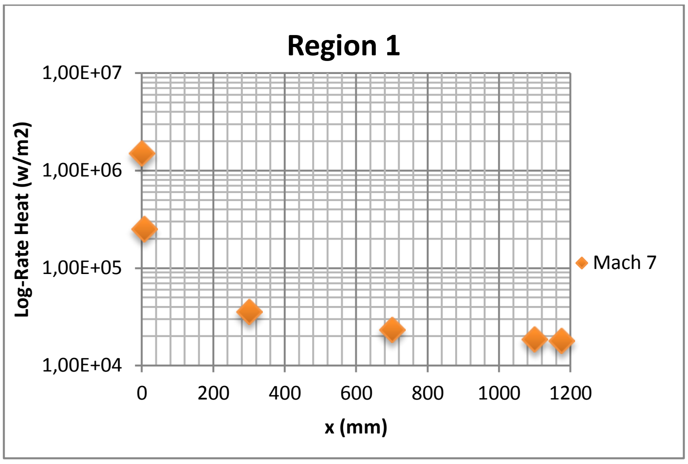

Figure 4: rate heat in the region 1 for Mach 7.

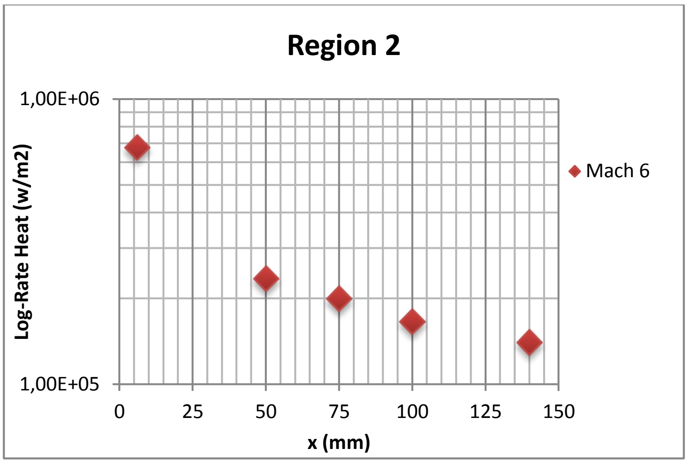

Figure 5: rate heat in the region 2 for Mach 6.

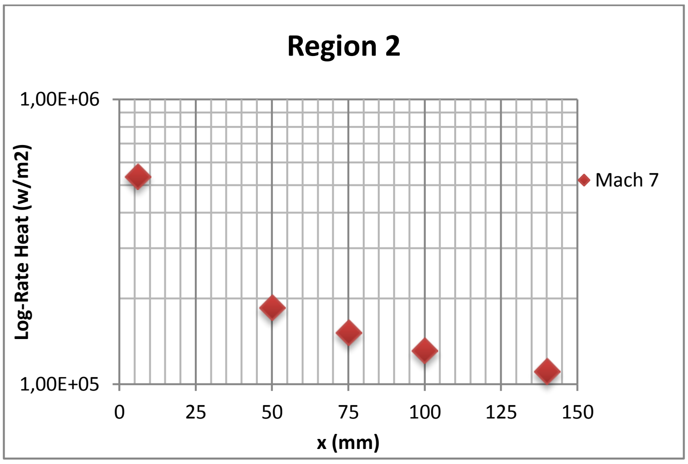

Figure 6: rate heat in the region 2 for Mach 7.

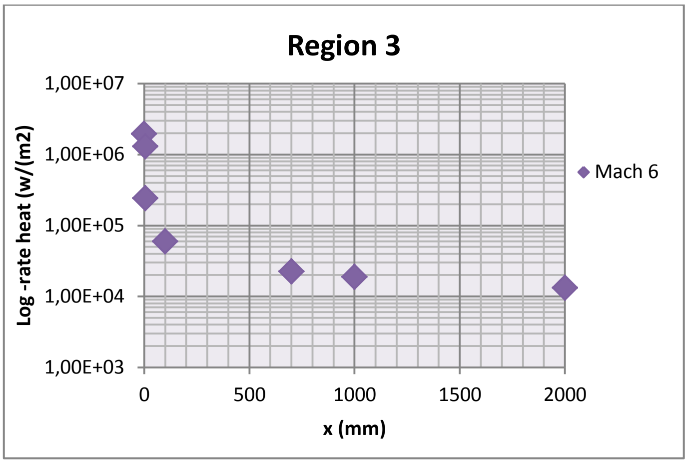

Figure 7: rate heat in the region 3 for Mach 6.

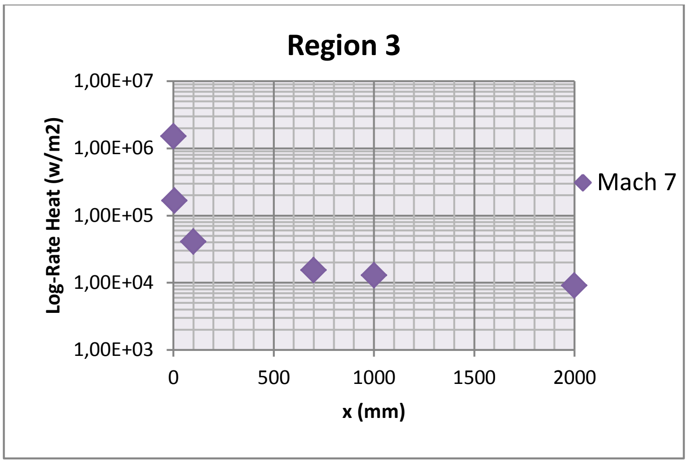

Figure 8: rate heat in the region 3 for Mach 7.

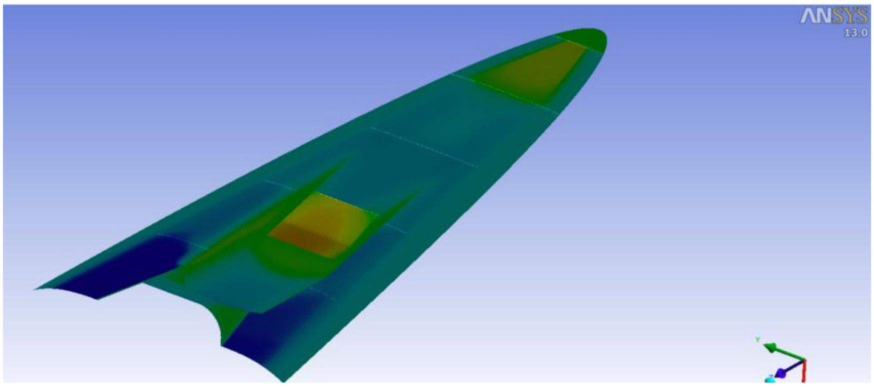

Figure 9: rate heat under vehicle for Mach 6.

The results are same order of magnitude found by Machado and Pessoa Filho (2007) for the fins of the vehicle VSB30. Note that the rate heat decreases for condition 2, because, although the velocity increases, the air density is low. It is also evident that the heating in the region of stagnation point is grater because this location is under the influence of normal shock waves, where the energy conversion is more pronounced. The results also shown that the closer to the leading edge, grater the rate heat, this is due to of the greater portion of the kinetic energy of the vehicle is converted into thermal energy in this region, as shown in figs 3-8, these results are agreement with Odam *et al* (2005) in their studies of heating aerodynamic of Hyshot.

This work, although analytical, indicates the most critical regions and trajectories that more thermally request structure. Works for determining surface temperature map of the vehicle should be developed and studies in CFD (Computational Fluid Dynamics) and ground experiments are needed to increase the results validation and the accuracy of the thermal protection project in order to determine an optimal thickness for the coating.

5. REFERENCES

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