AERODYNAMIC HEATING OF MISSILE/ROCKET – CONCEPTUAL DESIGN PHASE

Duarte, Guilherme Felipe Reis, flpduarte@gmail.com
Instituto de Aeronáutica e Espaço, Divisão de Sistemas de Defesa, DCTA, Brasil

Silva, Maurício Guimarães da, maugsilva@iae.cta.br
Instituto de Aeronáutica e Espaço, Divisão de Sistemas de Defesa, DCTA, Brasil

Castro, Breno de Moura,
Instituto de Aeronáutica e Espaço, Subdepartamento Técnico do DCTA, DCTA, Brasil

Abstrac. *This report describes the methodology which can be adopted to estimate the aerodynamic heating in missile/rocket aerodynamic configurations in a conceptual design level. A MATLAB® computer code has been developed to calculate the transient missile aerodynamic heating parameters utilizing basic flight parameters such as altitude, Mach number, and angle of attack. The insulated skin temperature of a vehicle surface on either the fuselage (axisymmetric body) or wing (two-dimensional body) is computed from a basic heat balance relationship throughout the entire spectrum (subsonic, transonic and supersonic) of flight. This calculation method employs a finite difference procedure which considers radiation, forced convection, and non-reactive chemistry. Eckert's reference temperature method is used as the forced convection heat transfer model. The surface pressure estimative is based on a modified Newtonian flow model and CFD method. The CFD analysis is conducted by use of the NSC2KE software, a free code developed to solve 2D and axisymmetric flow. The principal options of the software are* $k-\varepsilon$ *turbulence model and non-structured/structured grid. The convective heat transfer coefficient can be obtained from two methodologies: Semi-empirical method or Parametric System Identification (PSI) which uses experimental temperature in the estimative process.*

Keywords: *Aerodynamic Heating; Conceptual Design; Missile; Rocket; Dome*

1. INTRODUCTION

As a body moves through the air at speeds many times that of sound, the air ahead of the body receives very little “warning” that a body is approaching. As a result, there is a sudden compression of the air very near the surface of the body and a flowing around the body of a thin layer of high speed, high temperature air. A supersonic missile, for example, experiences aerodynamic heating as a result of the conversion of air velocity into heat via the compressibility of the air through which it moves and the viscous forces acting in the boundary layer that surrounds the missile. The amount of heat generation depends directly on the speed of the missile and increases with speed. The ratio between the two modes of heat generation varies along the missile or rocket in an atmospheric flight. Near the stagnation point, the heat generated by direct compression of the air will dominate while on the sides of the missile the conversion of velocity into heat by the viscous forces within the boundary layer will be dominant. For a high-speed missile, this heat generation can be significant. However, it is not the only consideration when determining the effects of aerodynamic heating. The influence of the aerodynamic heating on the dome of a missile flying at supersonic speed has a negative effect on optical system performance since the thermal characteristics of material of optical components, dome and air inside of dome change during the flight (transient behavior). Another question is related to the electrical/electronic devices inside of missile body. These components have a number of power sources which constitute another contribution in terms of heating. This study looks at the interaction of these various factors and their application to the missile/rocket heating problem.

A MATLAB computer code has been developed to calculate the transient missile aerodynamic heating parameters utilizing basic flight parameters such as altitude, Mach number, and angle of attack (or pressure coefficients). The insulated skin temperature of a vehicle surface on either the fuselage (axisymmetric body) or wing (two-dimensional body) is computed from a basic heat balance relationship throughout the entire spectrum (subsonic, transonic and supersonic) of flight. This calculation method employs a simple finite difference procedure which considers radiation, forced convection, and non-reactive chemistry. The surface pressure estimates are based on a modified Newtonian flow model and CFD method. The CFD analysis is conducted by use of the NSC2KE software, a free code developed to solve 2D and axisymmetric steady flow. The principal options in terms of use of the software are $k-\varepsilon$ turbulence model and non-structured/structured grid. Eckert's reference temperature method is used as the forced convection heat transfer model. The code was developed as a tool to enhance the conceptual design process of high speed missiles and rockets. Recommendations are made for possible future development of the software to further support the design process.

2. MATHEMATICAL FORMULATION

2.1. Recovery Temperature

In the absence of any internal cooling or heating process, forced convection is the most significant factor involved in the heating analysis of a high-speed missile. Areas near to the stagnation point derive most of their heat from the compression of the air. These regions of the missile will not have a large variation of temperature normal to the surface. This also means that there is a large supply of heat energy, so that any conduction of heat away from the skin will be quickly replenished. It is likely that in this region the flow will be laminar, especially for an IR missile where the sapphire dome is very smooth. This will affect the ability of the boundary layer to transfer heat energy to the missile surface; a laminar boundary layer will conduct less heat than a turbulent one. For a supersonic missile, the stagnation temperature is strongly dependent on the Mach number. The stagnation temperature (T_0) is a function of the free stream static temperature (T_∞) and Mach number (M_∞) only:

$$\frac{T_0}{T_\infty} = 1 + \frac{\gamma - 1}{2} M_\infty^2 \tag{1}$$

Areas on the missile far away from the stagnation point will derive most of their heating from the viscous deceleration of the air in the boundary layer. For a high-speed boundary layer, the air is brought to rest at the wall in a thermodynamically irreversible process. Part of the kinetic energy is converted to heat and part is dissipated as viscous work. The boundary layer is very thin in comparison to the missile and there can be a large temperature gradient in the boundary layer, especially at the larger Mach numbers. The thin nature of this boundary layer implies that there is little heat generation capacity and as a result, the amount of heat energy conducted away from the boundary layer will strongly affect the temperature at the surface of the missile. The variable that can be used to estimate the influence of these parameters on the flight conditions is the recovery temperature. The recovery temperature is a function of the nearby temperature and Mach number just outside of the boundary layer, and it is dependant of the type of boundary layer. The recovery temperature is higher for a turbulent boundary layer. The recovery temperature (T_R) at each flight condition is obtained from the trajectory profile for the baseline model of the missile. Given this data, the following equation is solved to determine the ratio between the recovery temperature and the stagnation temperature (T_0) at the boundary layer edge (subscript L):

$$\frac{T_R}{T_0} = r + \frac{1 - r}{1 + \frac{\gamma - 1}{2} M_L^2} \tag{2}$$

The variable r is defined as recovery parameter, which is:

$$r = \sqrt{\mathrm{Pr}} \quad \text{for Laminar flows and}$$
$$r = \sqrt[3]{\mathrm{Pr}} \quad \text{for Turbulent flows,} \tag{3}$$

Pr is the Prandtl number. The most of tactical missiles have a relatively short time of flight. The heat transfer is usually in a transient state condition. Aerodynamic heating/thermal response prediction requires considerations of two different types of prediction method applicable to two different types of surfaces: thermally thin and thermally thick, Fleeman (2006). Both methodologies consider the evolution of temperature on the skin (at transient regime) as function of recovery temperature and recovery parameter. A thermally thin surface can be approximated as one-dimensional heat transfer with nearly uniform internal temperature that increases with time, eventually, approaching the recovery temperature. The testing of the assumption that the rocket baseline uninsulated airframe is thermally thin surface is given by Biot number, which is:

$$h\left(\frac{z}{k}\right)_{Surface} < 0.1 \tag{4}$$

where h is the convection heat transfer coefficient of air (Btu/s/ft^2/R), z is the local thickness of airframe (ft) and k is the thermal conductive of airframe material (Btu/s/ft/R). In this work it is adopted the assumption thermally thin surface.

2.1. Thermal energy balance

The fundamental concepts used to build the theoretical model involve two basic modes of heat transfer: convection and radiation. Forced convection is assumed because the vehicle is propelled through the air by the release of chemical energy. During supersonic flight the local stagnation pressure at the edge of the boundary layer is assumed to be defined by the stagnation pressure behind a normal shock wave.

Boundary layer flow is assumed to be turbulent due to the magnitude of the Reynolds number which is generally greater than 500,000. The gas dynamic relations are based on inviscid flow, no reacting chemistry assumptions. The primary objective of the computer code is to rapidly compute the transient insulated skin temperatures along the trajectory of the missile at a given point somewhere on the surface. To accomplish this, the thermal energy balance equation, Eq.(5), is solved for T_S using a finite difference method. However, before this can be accomplished, the adiabatic wall temperature and the heat transfer coefficient must be determined. The required data for the skin temperature calculations include trajectory parameters (time, Mach number, angle of attack, altitude), material properties (density, material specific heat, emissivity), the location and skin thickness at the point of interest, and a radiation reference temperature. This radiation temperature is the temperature of a distant surface seen by the insulated skin element, which for this program, is either space or the earth.

The governing equation which serves as the basis for the computer program incorporates both radiative and forced convective heat transfer processes. The thermal energy balance equation for this insulated skin heat transfer case is given by:

$$mc_p \frac{\delta T_S}{\delta t} = hA(T_{AW} - T_S) - \sigma \varepsilon A(T_S^4 - T_R^4) \tag{5}$$

where the variable m is the mass of the structure and c_p is the specific heat of the structure material. The parameters σ and ε are the drivers for radiation transfer calculations, Boltzman constant and emissivity of the body, respectively. The convective heat transfer coefficient is represented by the variable *h*. The equation written in this form assumes that the adiabatic wall temperature (T_{AW}) is greater than the insulated skin temperature (T_S) which is greater than the radiation reference temperature (T_R). Once the complex heat transfer coefficient calculation is performed and the other significant heat transfer parameters are determined, a finite difference numerical method can be applied, yielding a skin temperature profile for the specified point of interest.

2.2. Adiabatic wall temperature

The first parameter needed to determine skin temperature is the adiabatic wall temperature, given by Fleeman (2006):

$$T_{AW} = T_L\left[1 + \frac{r(\gamma-1)}{2} M_L^2\right] \tag{6}$$

The local Mach number (M_L) is dependent on local pressure and stagnation pressure. For real viscous flow over a missile body, with heat transfer, the local Mach number and temperature at the edge of the boundary layer (subscript L) can be approximated by using the inviscid isentropic calculations for the values at the surface. For isentropic inviscid flow the relationship between the local Mach number and static pressure is given by the following equation, ESDU item 82018.

$$M_L = \sqrt{\left[\left(\frac{P_{0L}}{P_L}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]\frac{2}{\gamma-1}} \, . \tag{7}$$

For isentropic inviscid flow with no heat transfer to or from the surface the local surface temperature is given by the isentropic formulation:

$$T_L = \frac{T_{0L}}{\left(1 + \frac{\gamma-1}{2} M_L^2\right)} \, . \tag{8}$$

The speed of the missile, whether subsonic or supersonic will determine the local stagnation pressure. If the trajectory profile calls for subsonic flight, then the local stagnation pressure will be (Zucker, 1977):

$$P_{0L} = P_\infty\left[1+\frac{\gamma-1}{2}M_\infty^2\right]^{\gamma/\gamma-1}. \tag{9}$$

If the Mach number is in the supersonic range, the local stagnation pressure is then (Zucker, 1977):

$$P_{0L} = P_{0\infty}\left[\frac{\frac{\gamma+1}{2}M_\infty^2}{1+\frac{\gamma-1}{2}M_\infty^2}\right]^{\gamma/\gamma-1}\left[\frac{2\gamma}{\gamma+1}M_\infty^2-\frac{\gamma-1}{\gamma+1}\right]^{1/1-\gamma}, \tag{10}$$

where

$$P_{0\infty} = P_\infty\left[1+\frac{\gamma-1}{2}M_\infty^2\right]^{\gamma/\gamma-1}. \tag{11}$$

The subscripts 0 and ∞ denote stagnation conditions and free stream conditions, respectively. Free stream pressure (P_∞), temperature (T_∞), and density (ρ_∞) are calculated using standard atmosphere property value approximations found in Anderson (1991). Local surface pressure (P_L) is determined using modified Newtonian theory (Grimminger *et al.*., 1950) or from CFD analysis:

$$P_L = qCp_L + P_\infty\ , \tag{12}$$

where

$$Cp_L = Cp_{\max}\cos^2\theta \quad \text{(Newton Theory).} \tag{13}$$

The variable θ is related to the angular position on the body. $Cp_{\max}$ is the maximum pressure coefficient on the body surface and q is the dynamic pressure obtained during the flight, Equations (14) and (15), respectively.

$$Cp_{\max} = \frac{P_{0L}-P_\infty}{q}\text{, and} \tag{14}$$

$$q = \frac{1}{2}\rho_\infty V_\infty^2 = \frac{1}{2}\gamma P_\infty M_\infty^2. \tag{15}$$

Modified Newtonian flow theory has been shown to be applicable for the prediction of local surface static pressures over all surfaces experiencing non-separated flow (DeJarnett *et al.*., 1985 and Newberry and Roserfield, 1961). The accuracy of Modified Newtonian flow theory is quite reliable at hypersonic speeds. Its use at subsonic and low supersonic speeds is justified by the continuity and simplicity it provides in the lower Mach number regions where aerodynamic heating rates are so low as to be negligible.

2.3. Thermodynamic properties

The specific heat (c_p) of air used in the Prandtl number is given by Chapman (1960). It is used the local temperature as reference temperature:

$$c_p = 0.219 + 0.342.10^{-04}T_L - 0.293.10^{-08}T_L^2 \tag{16}$$

Analogously, thermal conductivity (Brown et al.., 1958) and dynamic viscosity (Chapman, 1960) are also temperatures dependent and are given respectively by:

$$k = \frac{0.0141}{3600}\left(\frac{707}{T_L + 225}\right)\left(\frac{T_L}{492}\right)^{3/2} \tag{17}$$

$$\mu = \frac{2.27.10^{-08} T_L^{1.5}}{T_L + 198.7} \tag{18}$$

Prandtl number is calculated based on specific heat, thermal conductivity and dynamic viscosity, which is:

$$\mathrm{Pr} = \frac{\mu c_p g}{k} \tag{19}$$

It is emphasized that the temperature T_L is given in Rankine (R) and the distance from the nose of body in feet (ft). The parameter g is related to the conversion of unit. In this case, $g =$ 32.174 ft/s^2.

2.4. Heat transfer coefficient

In order to estimate the heat transfer coefficient, one uses the concept of reference temperature. The reference temperature is the temperature whose magnitude is typically between that of the wall temperature and in low speed flow, the reference temperature does become the average of the two. This reference temperature more accurately models the temperature distribution that occurs within the boundary layer which has a significant impact on the skin temperature. The transport properties should be evaluated at the reference temperature, which is defined by Eckert (1955):

$$T^* = T_L + 0.5(T_S - T_L) + 0.22(T_{AW} - T_L) \tag{20}$$

Based on experimental data, the Nusselt number can also be written as a function of Reynolds and Prandtl number, (Chapman, 1960):

$$Nu = 0.0292\,\mathrm{Re}^{*0.8}\,\mathrm{Pr}^{*1/3} \tag{21}$$

where the boundary layer is assumed to be turbulent and the Reynolds number is defined as:

$$\mathrm{Re}^* = \frac{\rho^* M_L \left[1 + 0.5(\gamma - 1) M_L^2\right]^{\frac{\gamma+1}{2(\gamma-1)}} \sqrt{\gamma R T_{0L}}}{\mu^*} x, \tag{22}$$

and

$$\mathrm{Pr}^* = \frac{\mu^* c_p^*}{k^*}. \tag{23}$$

With the formulations above mentioned, it is possible to write the convective heat transfer coefficient using:

$$h = \frac{k^* Nu^*}{x} \tag{24}$$

With these significant variables defined, the thermal energy balance equation can be solved using a simplified finite difference method. This method calculates the temperature value at a future time based on a calculated time increment and the current temperature value.

3. PARAMETRIC SYSTEM IDENTIFICATION (PSI)

The algorithm shown at the preceding sections does not consider experimental data in the formulation, such as that one from flight or wind tunnel test. Essentially, the input of the program is defined by geometrical and physical properties of body, flight envelope, and local pressure, which can be estimated by Newton method or CFD steady simulations. Another question is related to the thermally thin surface assumption. From this assumption it is adopted a 1D formulation during the estimative of final temperature of surface, i.e., Eq. (5) is solved for each node of the grid in an uncoupled way. It is a good approach for a typical short range missile, since the time of flight is less than 9 seconds, but it fails in the estimation of surface temperature of long range missiles (or rockets). The objective of the PSI method is to include the information from the experimental data in the mathematical formulation of aerodynamic heating in order to estimate the average heat transfer coefficient. Basically, the purpose of method is to try to match calculated final temperatures with known experimental temperature data and use the value found for to estimate the heat flux. Essentially, the parametric system identification (PSI) consists in to include an optimizer that could adjust the unknown h thermal coefficients until a best-fit approximation in a least squares sense to experimental temperature data is obtained. Parameter identification has several advantages over the other modeling choice as, for example, Computational Fluid Dynamics (CFD). Creating a mathematical model of the missile using CFD unsteady analysis is a critical job due to the runtime and the number of configurations of analysis (high/low altitude, three-dimensional, turbulent, compressible supersonic and subsonic flow). PSI ignores these complexities and focuses on the final result. This makes PSI not only simpler, but the information that comes out of the PSI model can easily be used in improving the design. Another interesting aspect of this methodology is that PSI also handles nonlinear conditions such as the final effects at the aerodynamic heating due to asymmetric vortices originated from the flight at high angle of attack.

The PSI procedure adopted in this work follows the reasoning line: firstly, physical, geometric, steady pressure coefficient and experimental data is read in from a data file. The optimizer function calls a user written function called TEMPERATURE that calculates the temperature-time history (ordinary differential equation) using the current parameters supplied by optimizer. Once the temperature-time history is calculated, an error function is returned to optimizer function based on the differences between predicted and experimental temperature-time histories. The optimizer then adjusts the unknown parameters (heat transfer coefficient) and the process is repeated until certain convergence criteria are met. It is used the optimizer *fmincon*, function from MATLAB, which uses a Levenberg-Marquardt method and an active set strategy to minimize an error in a least-squares sense subject to simple constraint placed on the variables by the user. The output of analysis is the average heat transfer coefficient parameter.

4. NUMERICAL IMPLEMENTATION

The steps involved in the process of analyses are described bellow:

1- Input:
- Initial conditions in term of temperature;
- Geometry of missile indicating the laminar and turbulent regions (grid);
- Physical properties of missile (thickness, material);
- Pressure distribution on the missile (Newton method and CFD results);
- Flight profile from the flight dynamics;
- Temporal variation of experimental temperature (Texp) at the grid

Marching Time …

2- Calculate the Local Conditions:
- Stagnation pressure based on the flight profile (Eqs.9-11)
- Stagnation temperature (Eq. 1)
- Pressure Coefficient (Eq.13-15) or CFD simulation results
- Static Pressure (Eq.12)
- Mach number (Eq.7)
- Static Temperature (Eq.8)
- Density (Gas perfect equation)
- Specific heat (Eq.16)
- Thermal conductivity (Eq.17)
- Viscosity (Eq. 18)
- Prandtl number (Eq. 19)
- Recovery factor (Eq. 3)

3- Calculate the adiabatic wall temperature (Eq.6)
4- Calculate the convective heat transfer coefficient (Eqs.20-24)
5- Determine the wall equilibrium temperature (T_S) from the differential equation (5) applied to the all points of the grid (uncoupled way). In this first approach, it is not considers the radiation term.
6- Go to the step 2;

End of Marching Time.

7- Compare the last experimental temperature at the node A with the results from the *Marching time*, which is, $T_{S=node(A)}$;

8- If $Cost = Abs\left(T_{S=A} - T_{S=A}^{Exp}\right) > 10^{-04}$, adopt another value for h using a optimization algorithm and go to the step 2 (*Marching Time*);

9- The method converge when the cost function is smaller than 10^{-04}.

10- Output

The wall equilibrium temperature (T_S);

The average heat transfer coefficient h;

5. CONFIGURATION OF SIMULATION

5.1. Flight Profile

In aircraft close-in-combat scenarios the ability to engage targets in the rear hemisphere, is a significant advantage. Super-agility in missiles refers to this capability. Following a successful missile launch and separation, dynamic pressures are often too low for aerodynamic controls to make a quick turn. When the propulsion system ignites, vectoring the thrust (or using reaction jets) can provide this capability, and as the velocity increases, the aerodynamic surfaces become more effective. For the missile to possess super-agility (high-angle-of- attack capability) some form of alternate control is needed. Figure 3 illustrates the maneuvering of an agile missile from launch to endgame, indicating a high-angle-of-attack (AOA) maneuvering capability provided by either thrust vector control (TVC) or reaction control system (RCS) thrusters (Wise and Roy, 1998). Based on the Fig. 3, it was defined the flight profile shown in Fig. 4 for the test case.

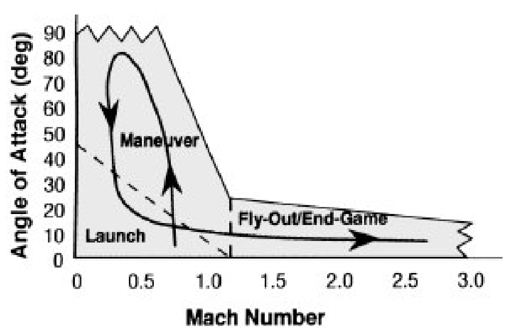


Figure 3. Agile missile flight envelope (Wise and Roy, 1998)

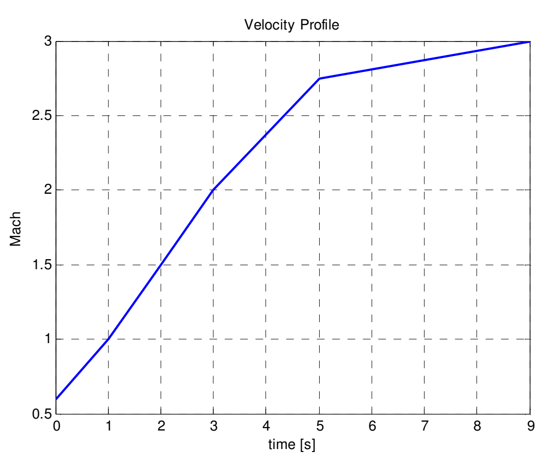


Figure 4. Flight Profile – Test Case (H= 0 m)

5.2. CFD Simulations

Computational Fluid Dynamics (CFD) simulations were carried out by NSC2KE software (Mohammadi, 1994), a free code developed to solve 2D and axisymmetric flow. The principal options of the software are $k-\varepsilon$ turbulence model and non-structured/structured grid. It was adopted axisymmetric flow (zero degree AOA) since it is the most critical condition in terms of aerodynamic heating. Table 1 shows the conditions of simulation and Fig. 5 present the grid chosen from the refinement test of grid. Table 2 and Fig.6 illustrate the principal results. It is possible to conclude that the results presented here in terms of shock position and pressure coefficients are satisfactory and can be used in the estimative of heat transfer coefficient. Finally, it is important to stress here that the pressure coefficient for Mach number greater than 3 was obtained from Newton method modified (Eqs 12-14).

Table 1. Simulation Conditions ($\alpha = 0°$).

Flight Regime	Mach	Reynolds/D (10^5)	Experimental Data
Subsonic	0,20	4,00	-
	0,40	4,00	-
	0,60	4,90	Hsieh (1976) e Azevedo (1989)
Transonic	0,85	4,00	Hsieh (1976)
	1,00	4,40	Hsieh (1976)
Supersonic	1,20	4,60	-

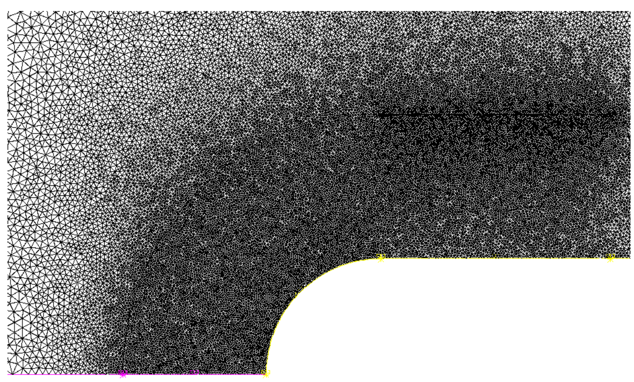


Figure 5. Geometry– Test Case

Table 2. Shock position comparison: Numerical (NSC2KE) and Experimental (Hsieh, 1976).

Mach	Numerical Shock Position (x/R)	Actual Shock Position (x/R)
1,2	-1,27	-1,27
1,5	-0,62	-0,60

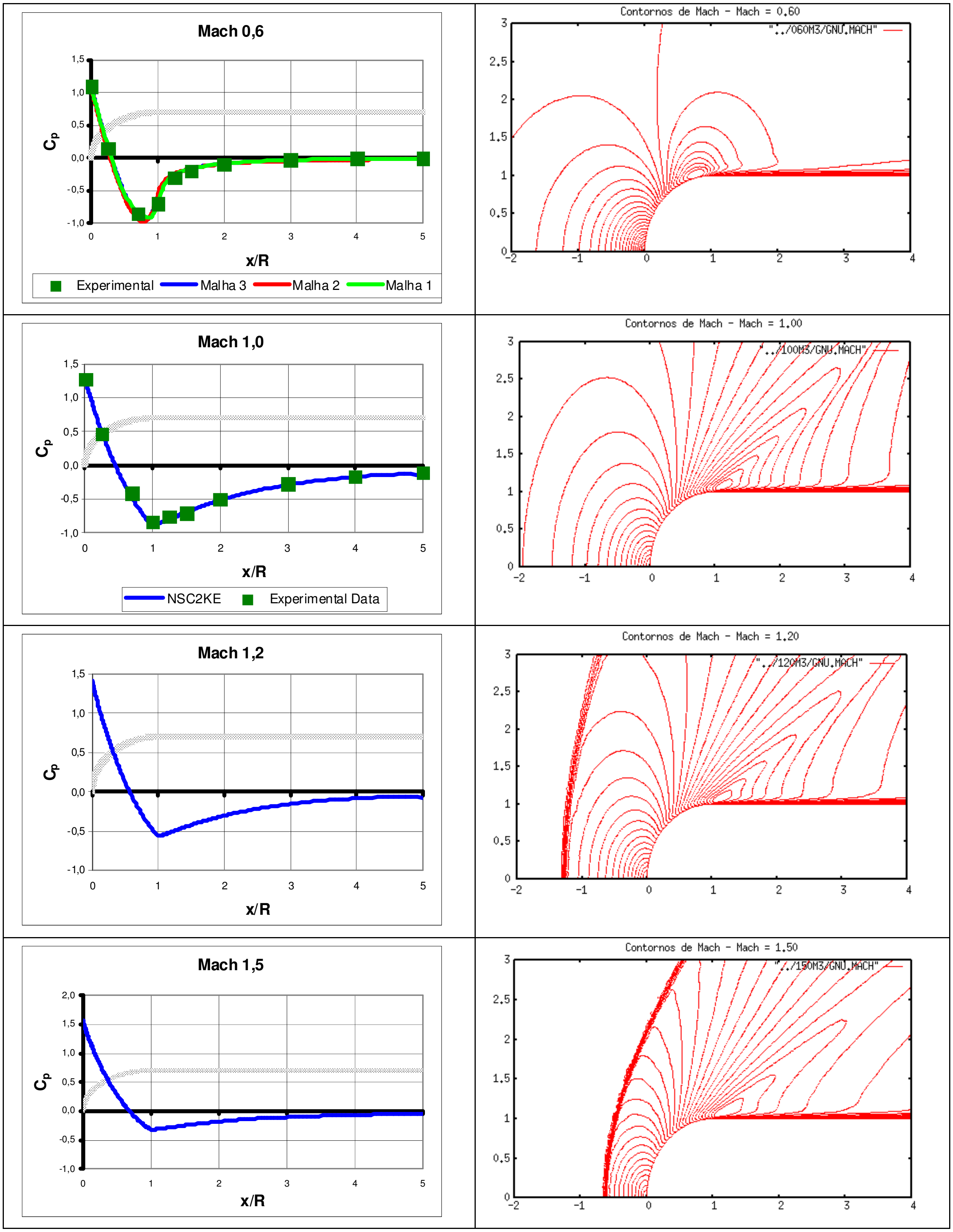


Figure 6. CFD Results

5.3. Case Study

Unfortunately, the most complete set of flight or wind tunnel data is classified at the present time. Consequently, a comparison of the theory with the experimental test is given separately in the form of a confidential addendum to this work. However, it is presented a simple case, adiabatic flow, in order to illustrate the applicability of method. The adiabatic case is significant since the boundary layer will act as an insulator to the missile and prevent heat conduction out of the missile for all wall temperatures below the adiabatic wall temperature. For wall temperatures below the adiabatic temperature, heat energy will be conducted out of the boundary layer. If radiation were not considered then the adiabatic wall temperature would be the same as the recovery temperature. Figure 7 shows the possible temperature profiles within the boundary layer (ESDU item 69011). For the adiabatic case, $\delta T/\delta y=0$, the skin temperature will be equal to the recovery temperature.

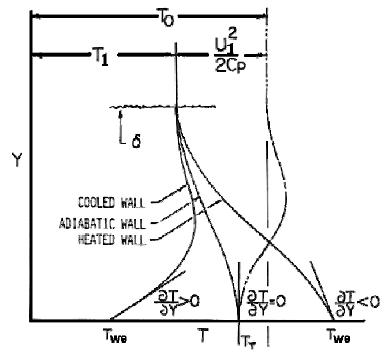


Figure 7. Boundary-layer temperature distribution.

In order to evaluate the methodology of identification, it is solved the inverse problem: it is adopted the final temperature at each point of the surface grid derived from the differential equation model, Eq. (5), as an input for the method. From now, this data will be considered as “Experimental Temperature”. From these data, it is possible to run the PSI method and identify the optimum parameter of the system, which is average convection heat transfer coefficient *h*. It is important to stress here that the methodology adopted to determine the “experimental temperature” uses heat transfer coefficient (*h*) as a function of time. The PSI method will obtain the average heat transfer coefficient. In order to test the robustness of the methodology, it is included a disturbance on the reference temperature (input) of the system too. This disturbance has been included in the final temperature on the body. Firstly is calculated a random number from the uniform probabilistic distribution inside the range [-1, 1], *unifrnd(-1,1,1)*. From the Monte Carlo method (5000 iterations), applied to each node, it is estimated the new reference temperature (TT) by adding up the real value of temperature (T) to the disturbance times the random number above mentioned, Eq. (26). Finally, the reference temperature that will be used in the PSI procedure is the average temperature for each node, Eq. (27)

$$TT\,(node,iteration) = T(node) + \Delta T * unifrnd(-1,1,1) \tag{26}$$

$$T\_New\,(node) = mean\,(TT(node,iteration)\,) \tag{27}$$

6. RESULTS

Four simulations were performed in order to evaluate the robustness of method. The first one deals with the identification of *h* parameter using the pressure derived from the CFD method. The principal objective with this analysis was to choose the best optimizer for the software. In this context, it was not used the radiation term neither noise. Figure 8(a) shows these results in a nondimensional context. The temperature was normalized by the stagnation conditions. The reasonable question to ask first is whether or not the solver can satisfies the restrictions imposed to the problem. From the Fig.8(b) it is evident that the solver chosen was right. The second test uses the temperature data with disturbance in the process of identification. This case is more representative with regard to the experimental data available in the real life. Figure 9 shows the good concordance found among the experimental and numerical results. It is important to stress here that the convergence was very fast and the average difference between the heat transfer coefficient estimated and the experimental one was less than 15%. In the third test, it is tested the methodology of identification using the same parameters defined at the first test but, in this time, it is included the radiation term in the mathematical model. The influence of term T_S^4 is illustrated at the Fig. 10. There is certainly non reason to expected great difference between the simulation illustrated in the Figs. 9 and 10, since the time flight set up in these simulations was very small (9 s). Finally, the last simulation is to evaluate the robustness of the method. It is included noise and radiation term. The flight time was redefined for twenty seconds (t_{final}= 20 s). Figure 11 shows the results of simulation.

In this case it is possible to note the influence of radiation at the final surface temperature. In fact, long time of flight implies great surface temperature. In summary, for all practical purposes, the distinguished characteristics of the method devised in this report show that the algorithm can be used in the conceptual design phase for analysis of aerodynamic heating of rockets and missiles.

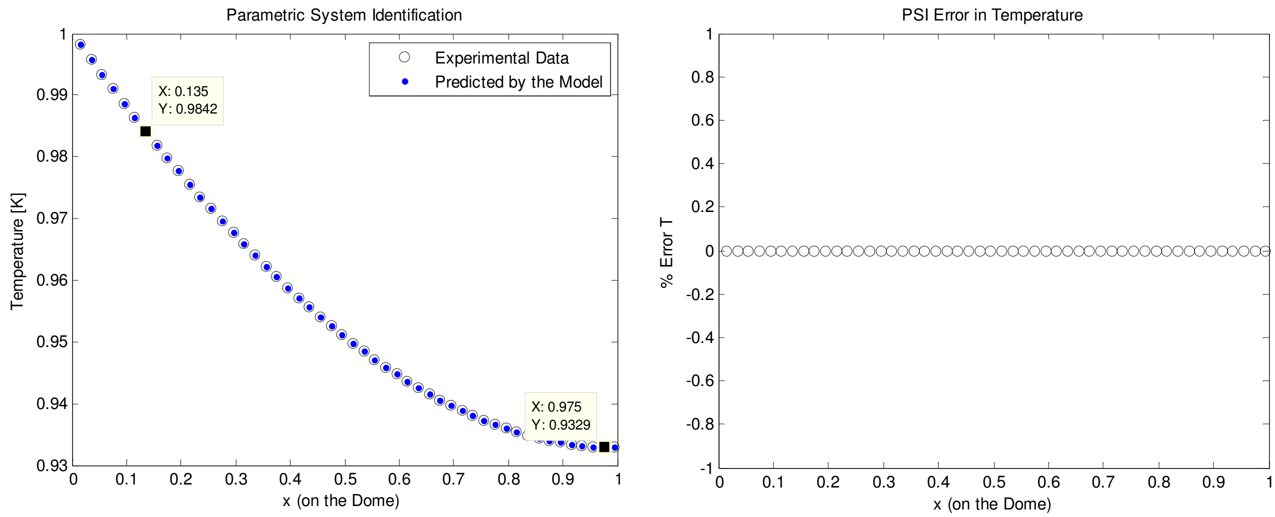


Figure 8. Optimizer *fmincon*.

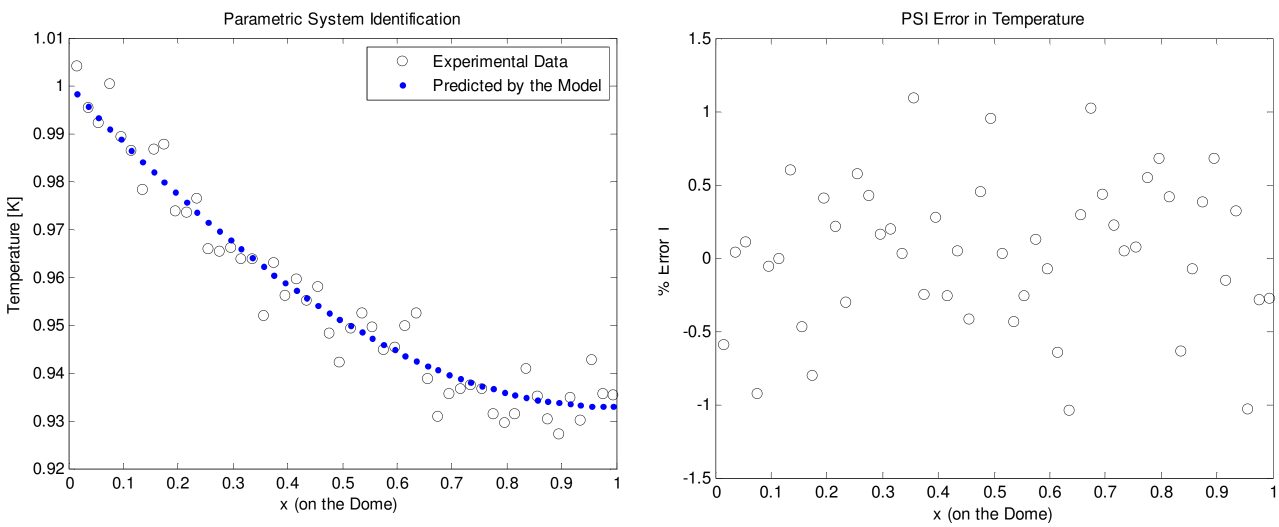


Figure 9. PSI method with noise.

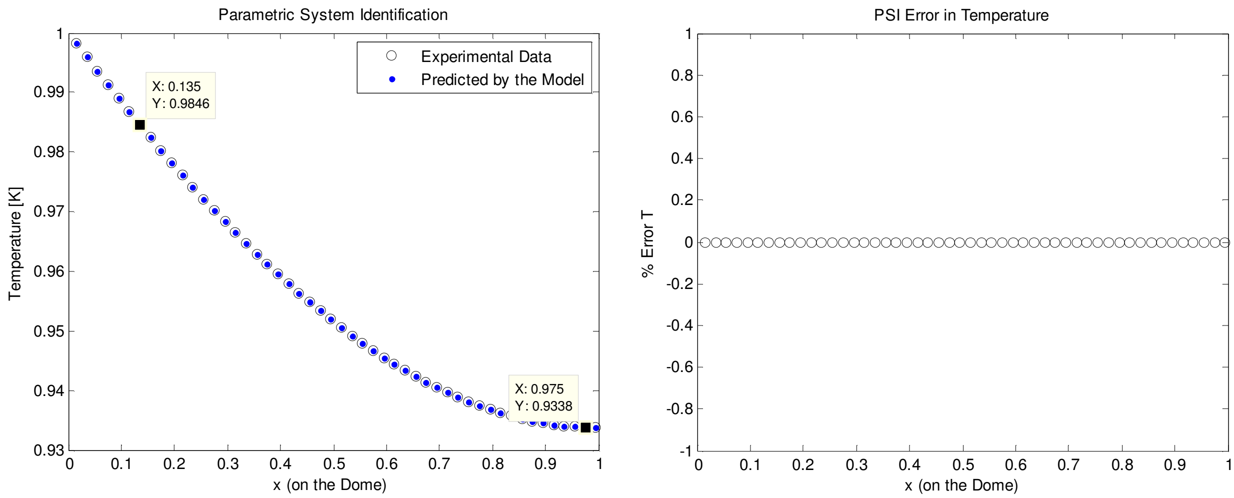
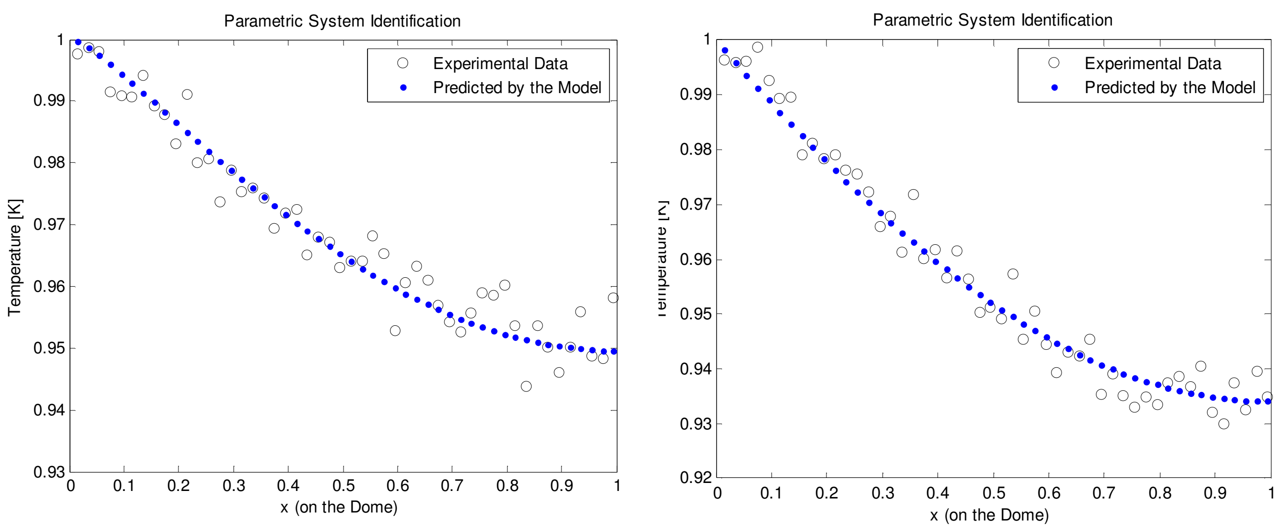


Figure 10. PSI method with Radiation term.



(a) With radiation

(b) Without Radiation

Figure 11. PSI method with noise and radiation (20 seconds).

7. CONCLUSIONS

This report described the methodology which can be adopted to estimate the aerodynamic heating in missile/rocket aerodynamic configurations in a conceptual design level. A MATLAB computer code has been developed to calculate the transient missile aerodynamic heating parameters utilizing basic flight parameters such as altitude, Mach number, and angle of attack (or pressure coefficients). The insulated skin temperature of a vehicle surface on either the fuselage (axisymmetric body) or wing (two-dimensional body) was computed from the basic heat balance relationship throughout the entire spectrum (subsonic, transonic and supersonic) of flight. This calculation method employed a simple finite difference procedure which considers radiation, forced convection, and non-reactive chemistry. The surface temperatures were estimated from the surface pressure distribution. A modified Newtonian flow model and CFD method for estimative of pressure have been used. A semi-empirical method based on the Parametric System

Identification (PSI) has been used to estimate the convective heat transfer coefficient and, in consequence, the heat flux. In order to test the algorithm, a set of results of surface temperature were defined as "experimental data". From an optimization process it was possible to estimate the surface temperature and heat parameters when noise and radiation influence are considered. The code was developed as a tool to enhance the conceptual analysis of aerodynamic heating of high speed missiles and rockets. In order to improve the methodology, it is recommended the following implementations:

i. Estimate, from another CFD software (for example FLUENT), the surface pressure AND temperature distribution. It was important to calibrate the method;
ii. Include internal elements, with different energetic characteristics (heat source, material, area, ...) in order to evaluate the use of different thermal resistance with regard to the robustness of the method ;
iii. Applies the methodology for different configurations, which are: body, body+wing, body+wing+fin, body+strakes+fin, and so on, and validated with experimental data;
iv. Include the influence of thickness. It will be possible to analyze the convergence of optimization method when it is considered the coupling among the balance equations;

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