

Original Research Article

The First Mission of Iranian Spacecraft: Heat Shield Design

ABSTRACT

In the present paper, the Iranian spacecraft is considered, and is studied from an aerothermodynamic viewpoint to design the proper thermal protection system, for its first mission: the sub-orbital flight. The design procedure is based on the decoupling of the aerodynamic heating phenomena and the solid phase heating process. Due to the sub-orbital mission of this spacecraft as a first flight, an elastomeric material is used as an unconventional heat shield to protect the blunt body reentry vehicle. The full Navier-Stokes equations are used for heat-flux estimation during flight, and the one-dimensional equations for an ablative materials are used to design the thermal protection system, according to the structural criteria.

Keywords: Ablative material, Blunt body, Design, Iranian spacecraft, Reentry, Sub-orbital flight, Thermal protection system.

1. INTRODUCTION

In order to fulfill the Iran's space program, the first Iranian spacecraft is under design and construction, as shown in Fig. 1. One of the major problems to be considered in high speed flights is temperature rise due to an aerodynamic heating. The aerodynamic heat flux is sometimes so high that the thermal protection systems (TPS) such as ablative coatings must be used to protect the inner systems against the thermal failures. The TPS undoubtedly is one of the necessary systems for success in space missions.

Most widespread ablative heat shields are composite materials that have two components: resins and reinforced fibers. Resin is decomposed by heating and produces the pyrolysis gas. If the aerodynamic heating continues, the composite material completely converts to the char. The produced pyrolysis gases enter to the boundary layer flow, and during this travel, from the depth of the material to the surface, they absorb some heats; in addition, at the boundary layer, they reduce the convective heating. Some mechanisms like chemical reactions between pyrolysis gases and boundary layer gases cause the surface regression by consuming the formed char layer on the surface of the material [1-6].



Fig. 1. The Iranian spacecraft

The surface's temperature-time history must be predicted numerically to design the proper TPS. This requires the coupled solution of the flow field and the solid phase governing equations. Such studies are so much time-consuming processes due to the differences in order of magnitudes of the flight time scale and the required time steps of the numerical simulations, and therefore the better methods should be used instead. This is done here by decoupling of the flow field and the solid phase governing equations.

To do this, the flow field corresponding to the point on the trajectory is simulated using the constant temperature wall boundary condition, to calculate the heat flux in that point. Repeating this leads to the estimation of the cold-wall heat flux time history during the flight. Then the solid phase should be simulated using the predicted heat flux to calculate the temperature history within the shell. It is important here that the actual heat flux should be calculated instantaneously using the surface temperature. Increasing the number of trajectory points in this simulation process improves the accuracy of the results [7-13]. The first mission of the Iranian spacecraft is a sub-orbital flight. Although the final main mission will be the orbital flight with an actual high speed reentry, but the aerodynamic heating of the first mission (sub-orbital) is not comparable with the main mission. Therefore, the TPS design of the first mission is done here based on the lowest time and cost of development process, and the chlorosulfonated polyethylene synthetic rubber is chosen as the heat shield. The full Navier-Stokes equations are used to compute the heat flux during the reentry of the blunt body and then, the one-dimensional equation in the solid phase is used to determine the proper thickness of heat shield according to the structural criteria.

2. GOVERNING EQUATIONS AND NUMERICAL PROCEDURE

Full Navier-stokes equations governing the mass, momentum, and energy of the compressible flow are used to compute the heat flux. The conservative forms of the governing equations in an axisymmetric mode are presented here:

$$\frac{\partial U}{\partial t} + \frac{\partial(F + F_v)}{\partial x} + \frac{\partial(G + G_v)}{\partial y} + \frac{G + G' + G_v + G'_v}{y} = 0 \quad (1)$$

Where

$$\begin{aligned} U &= \begin{bmatrix} \rho \\ \rho u \\ \rho v \\ \rho e \end{bmatrix} & F &= \begin{bmatrix} \rho u \\ \rho uu + p \\ \rho uv \\ \rho uh \end{bmatrix} & G &= \begin{bmatrix} \rho v \\ \rho vu \\ \rho vv + p \\ \rho vh \end{bmatrix} & G' &= \begin{bmatrix} 0 \\ 0 \\ -p \\ 0 \end{bmatrix} \\ F_v &= \begin{bmatrix} 0 \\ -\tau_{xx} \\ -\tau_{xy} \\ q_x - u\tau_{xx} - v\tau_{xy} \end{bmatrix} & G_v &= \begin{bmatrix} 0 \\ -\tau_{yx} \\ -\tau_{yy} \\ q_y - u\tau_{yx} - v\tau_{yy} \end{bmatrix} & G'_v &= \begin{bmatrix} 0 \\ 0 \\ \tau_{\theta\theta} \\ 0 \end{bmatrix} \end{aligned} \quad (2)$$

In which the stress terms are defined as:

$$\begin{aligned} \tau_{xx} &= \mu \left(2 \frac{\partial u}{\partial x} - \frac{2}{3} \nabla \cdot V \right) & \tau_{xy} &= \mu \left(\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \right) \\ \tau_{yy} &= \mu \left(2 \frac{\partial v}{\partial y} - \frac{2}{3} \nabla \cdot V \right) & \tau_{\theta\theta} &= \mu \left(2 \frac{v}{y} - \frac{2}{3} \nabla \cdot V \right) \end{aligned} \quad (3)$$

Here

$$\begin{aligned} e &= c_v T + (uu + vv)/2 \\ q &= -k \nabla T \end{aligned} \quad (4)$$

It should be noticed again that the computed heat flux should be improved during the heat shield analysis [14], and the convective heat flux is just considered here. The commercial software for simulating the compressible flows is used in this analysis to compute the heat flux.

The governing equations for the ablative material analysis are the conservation of mass and energy.

Conservation of mass:

$$\left. \frac{\partial \rho}{\partial t} \right|_{x_s} = \frac{\partial \dot{m}_{g_x}''}{\partial x_s} \bigg|_t + \dot{S} \left. \frac{\partial \rho}{\partial x_s} \right|_t \quad (5)$$

Conservation of energy:

$$\rho c_p \frac{\partial T}{\partial t} = \frac{\partial}{\partial x_s} \left(k_x \frac{\partial T}{\partial x_s} \right) + (h_g - \bar{h}) \left. \frac{\partial \rho}{\partial t} \right|_x + \dot{S} \rho c_p \frac{\partial T}{\partial x_s} + \dot{m}_{g_x}'' \frac{\partial h_g}{\partial x_s} \quad (6)$$

Where,

$$\rho = \chi_v \rho_v + (1 - \chi_v) \rho_c \quad (7)$$

$$\begin{aligned} \bar{h} &= \left[\frac{\rho_v H_v - \rho_c H_c}{\rho_v - \rho_c} \right] & H_v &= \int_0^T C_{p_v} dT \\ \rho c_p &= \rho_v c_{p_v} \chi_v + (1 - \chi_v) \rho_c c_{p_c} & H_c &= \int_0^T C_{p_c} dT \end{aligned} \quad (8)$$

χ_v is a parameter which is used to present the properties, especially density of cells that are decomposed partially. If χ_v equals 1, the cell is not decomposed and is virgin. In addition, if the χ_v equals 0, the cell is completely decomposed and is char. For the states between these two extremes, virgin and char, χ_v can have the amount between 0 and 1.

The above equation is a complete set of equations to simulate thermal behavior of ablative materials. These equations are discretized by finite difference method. The energy balance at ablative material surface is:

$$\dot{q}_{in} = -k \left. \frac{\partial T}{\partial x} \right|_{at \ surface} + \dot{m}_{g_x}'' h_{g_x} + \rho_c \dot{S} \Delta H_c \quad (9)$$

More details can be found in Ref. 1. There are about 20 thermo-chemical properties which must be known to predict the ablative material behavior under the imposed heat flux. The measurements of these required data are so complicated, and even they couldn't be found easily in the literature. So, some of them should be guessed according to the data of the similar materials. Some data which are used in this computation are presented in Table 1.

Table 1: Thermo-chemical data of the ablative material

PARAMETER	VALUE
CONDUCTIVITY	0.15
INITIAL DENSITY	380.
HEAT CAPACITY	1500.
SUBLIMATION TEMPERATURE	500.
LATENT HEAT	4.E5

3. RESULTS AND DISCUSSION

The predicted trajectory of the sub-orbital flight of this spacecraft is shown in Fig. 2, and the variation of the predicted flight Mach number is presented in Fig. 3.

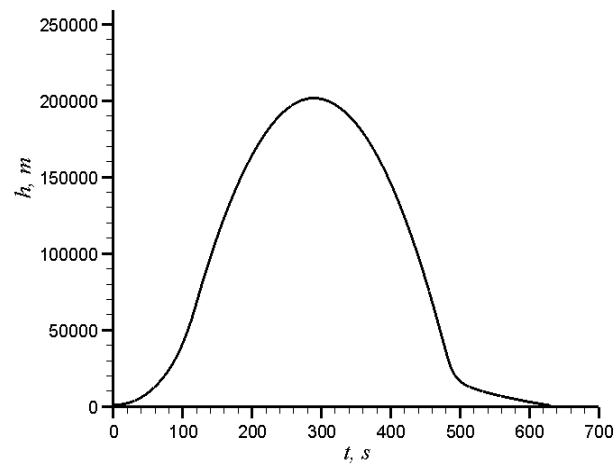


Fig. 2. The sub-orbital trajectory

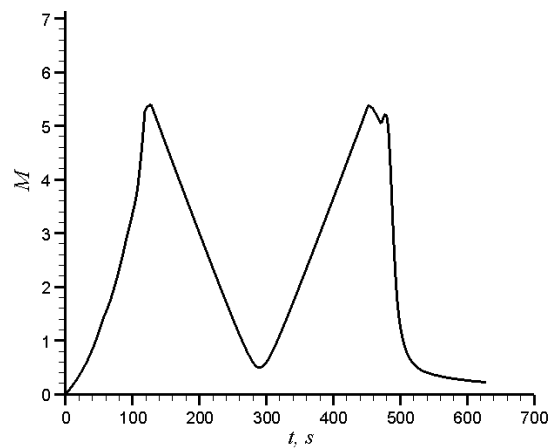


Fig. 3. The variation of flight Mach number versus time

Using this trajectory, and according to the spacecraft geometry, the flow field is simulated around it during the reentry flight, in some time-intervals. As an example, the Mach number contours are illustrated in Fig. 4 for $t=474$ s.

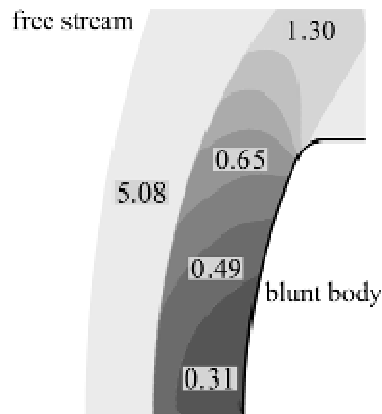


Fig. 4. Mach number contours at $t=474$ s

Using appropriate clustered grid near the body, the heat flux variation is estimated during the reentry flight as shown in Fig. 5. It should be noticed that the maximum heat flux of this sub-orbital flight is about one order of magnitude smaller than the high speed reentry from the orbital flight. In addition, this heat flux is the stagnation point's flux, which quickly decreases along the surface, but it is considered for the TPS design as a worst condition. The structural criterion for the TPS design is that the maximum temperature of the metal shell of the blunt surface must not increase more than 70°C during flight. A backup structure is chosen aluminum with the thickness of 3 mm.

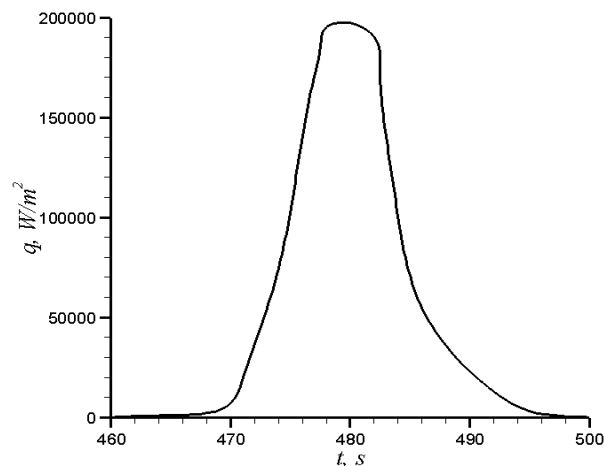


Fig. 5. Cold-wall heat-flux variation versus flight time

Using the ablative material simulation software, and according to this criterion, the proper thickness of the heat shield is computed by trial and error. The results show that the hypalon's thickness must be about 1 cm to protect the subsystems and satisfy the structural criteria.

Variation of TPS regression during reentry is shown in Fig. 6. The results show that about 70 % of the heat shield will be gasified and recessed due to the imposed heat flux.

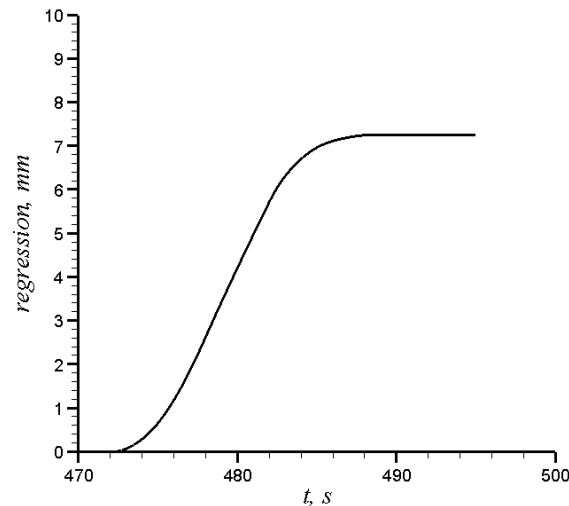


Fig. 6. TPS regression during reentry

4. CONCLUSION

The thermal protection system for the sub-orbital flight of the Iranian spacecraft is designed using numerical analysis. Due to the lower imposed heat flux compared to the high speed reentry, the unconventional heat shield is selected to be used there, which is chlorosulfonated polyethylene synthetic rubber. The results show that the maximum heat flux is about one order of magnitude smaller than the reentry flight from the orbit, and the thickness of the TPS is calculated 1 cm to protect the spacecraft's subsystems and satisfy the structural criteria. Beside the thermal properties of this material, the manufacturing feature of hypalon makes it proper to be used in this mission. However, the performance of this heat shield must be examined before flight by using the experimental techniques. It should be noticed that for the main orbital mission, the conventional heat shield like the carbon-phenolic must be used due to the higher heat fluxes.

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