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H. Holovina,
A. Boopathiraja

RESEARCH OF THERMAL CONDITIONS ON THE SURFACE OF ROCKET LAUNCHER PAYLOAD FAIRING DURING THE FLIGHT

Introduction

Due to the high flight velocities rocket launcher can experience intensive aerodynamic heating. Peak of aerodynamic heating usually occurs in stagnation point regions, such as on a blunt nose section. So biggest thermal loads act on rocket fairing and this cause severe risk to its structure and payload [13]. The main objective of this article is to determine the thermal protection for payload fairing (PF) that can maintain required internal temperature over the mission.

Thermal control techniques may be broadly grouped within two classes, passive and active, with the former preferred when possible because of simplicity, reliability, and cost. Passive control includes the use of cooling fins, special paints or coatings, insulating blankets, heat pipes, and tailoring of the geometric design to achieve both an acceptable global energy balance and local thermal properties [3, 4, 12].

The design of thermal protection is based on the principle that the thermal energy transmitted from the boundary layer must be either, absorbed, rejected, or expended. Three basic approaches to launch vehicle thermal control have evolved: heat sinking, radiative cooling, and ablative shielding.

The heat sink technique, as the name implies, uses a large mass of material with a high melting point and high heat capacity to absorb the entry heat load [3, 7].

The principle of radiative cooling is to allow the outer skin of the vehicle to become, literally, red hot due to the convectively transferred heat from the flow field around the vehicle. Blackbody radiation, primarily in the infrared portion of the spectrum, then transports energy from the vehicle to the surrounding atmosphere. Convective heating to the vehicle is proportional to the temperature difference between the fluid and the wall, whereas the energy radiated away is in proportion to the difference in the fourth powers of the fluid and wall temperatures. The net result is that thermal equilibrium can be reached at a relatively modest skin temperature provided that the rate of heating is kept low enough to maintain near-equilibrium conditions. Radiative cooling obviously requires excellent insulation between the intensely hot outer shell and the internal vehicle payload and structure. This is exactly the purpose of the shuttle tiles, the main element of the shuttle thermal protection system. Essentially a porous matrix of silica (quartz) fibers, these tiles have such low thermal conductivity that they can literally be held in the hand on one side and heated with a blowtorch on the other [3, 4].

Ablative cooling is also typically the least massive approach to heat protection. These advantages accrue at the expense of vehicle (or at least heat shield) reusability, which is a pronounced benefit of the other techniques. Ablative cooling occurs when the heat shield material, commonly a fiberglass resin matrix, sublimes under the heat load acting on rocket launcher. When the sublimed material is swept away in the flow field, the vehicle is cooled. This process can produce well over 10^7 J/kg of effective energy removal. Ablative cooling has been the method of choice for most existed launcher fairings in nowadays [3, 8, 12].

Materials for Thermal Insulation

The insulating capability of a material is measured with thermal conductivity (λ). Low thermal conductivity is equivalent to high insulating capability (R-value). In thermal engineering, other important properties of insulating materials are product density (ρ), surface emissivity (ε) and specific heat capacity (c). It is important to note that the factors influencing performance may vary over time as material ages or environmental conditions change.

Protection of a structure in a very high temperature environment may be accomplished with ease through the use of a new class of engineering materials. These thermally protective materials are known as “ablators” or “ablative material”. They are applied to the exterior of a load bearing structure and thereby isolate it from the hyper thermal environment. The structure is thus maintained near its initial temperature, at which it exhibits optimum strength characteristics [9, 12].

There are two categories of ablative materials: melting and non-melting types. In melting type (thermoplastics), the liquid is removed immediately after formation and newer surface is exposed; hence, it is not efficient. In non-melting category, again there are two types: high-temperature ablators (HTA) and low-temperature ablators (LTA). The examples of HTA are carbon–carbon (C–C) and carbon/silicon carbide ceramic matrix composites. Three-dimensional C–C composites are used in nose tip and leading edges. Because of their excellent strength retention with increase in temperature, the material remains in place and blocks the heat for longer duration and, subsequently, gets removed by oxidation at higher temperatures. In the case of LTAs, mechanical ablation precedes chemical ablation because their strength reduction with temperature is appreciable. LTAs are made of char-forming plastics (thermosets), which provide multiple levels of protection [5, 7].

After analyses of heat-insulating materials used in aerospace vehicles the aerogel was chosen as the newest and perspective insulation. With a density only three times the density of air, aerogel is the lightest solid material known. It is also called “frozen smoke” because of its appearance. It composed of up to 99.98% air by volume. Aerogels are a diverse class of “amazing materials” with properties unlike anything else. A typical thermal properties of silica aerogel are given below:

Table 1 – Properties of Aerogel [6]

Melting point	>1200 ℃
Density	2 to 350 kg/m^3
Thermal conductivity	0.015 W/mK
Specific Heat Capacity	840 J/kg/K
Thermal diffusivity	12 E^{-8} m^2/s

Design of thermal insulation

The design of thermal insulation was made on the example of fairing of Ukrainian rocket launcher “Cyclone-4”. The initial data for design has adopted the following:

1) geometric parameters of the payload fairing,
2) flight statistical data of altitude, velocity and time (standard flight parameters for considered class of rockets were taken from [1]),
3) mechanical parameters of chosen insulative material (aerogel – table 1).

Temperature distribution along the thickness of the solid elements can be determined by solving the differential equation of Fourier [2]

$$\frac{\partial T}{\partial \tau} = a_t \frac{\partial^2 T}{\partial^2 y}, \tag{1}$$

where, τ – time;
y – coordinate along the thickness;
$a_t = \frac{\lambda}{c\rho}$ – thermal diffusivity;
λ – thermal conductivity coefficient;
c – specific heat capacity;
ρ – density.

Boundary condition:
– at the internal surface

$$\left(\frac{\partial T}{\partial y}\right)_{in} = 0, \tag{2}$$

– at the external surface

$$\left(\frac{\partial T}{\partial y}\right)_{ex} = \frac{h}{\lambda}\left(T_r - T_{ex}\right). \tag{3}$$

Solving the Fourier equation with given boundary conditions can be obtained [2]

$$\tau = \frac{\delta^2}{a_t}\varphi, \tag{4}$$

where, δ – thickness of insulation

φ – Fourier's similarity criterion.

Equation (4) can be used for determining the required thickness of thermal insulation under non-stationary thermal process.

Fourier's similarity criterion depends on the ratio of temperature increment between internal fairing volume and external surface of insulation:

$$\frac{\Delta T_{in}}{\Delta T_{av}},$$

where, ΔT_{in} – temperature inside PF;

ΔT_{av} – temperature on the external surface of insulation

$$\Delta T_{av} = \frac{\Delta T_1 \cdot t_1 + \Delta T_2 \cdot t_2 + \cdots}{t_1 + t_2 \cdots}. \tag{5}$$

The two methods of determination of external temperature of thermal insulation surface: analytical and flow simulation in finite element method package, were carried out.

For analytical determination of temperature on the external surface of thermal insulation was used mathematical relation of radiative heat transfer [2]

$$\frac{h}{\varepsilon}\left(T_r - T_s\right) - \sigma T_s^4 = 0, \tag{6}$$

where, T_s – temperature on the surface;

$h = 2700\rho_\infty M_\infty$ – heat transfer coefficient,

ε= 1 – emissivity,

$T_r = T_\infty\left(1 + 0.18M_\infty^2\right)$ – reduction temperature,

σ – Stefan-Boltzmann constant.

The table 2 contains calculated according to the equations (6) and (5) surface temperature and average temperature respectively.

Among the series of average temperatures in the above table, the maximum one is should be selected for the further calculation of thickness. But here the 2nd highest value at 31 km is taken to compare the results with flow simulation because used for simulation finite element package works only with low altitudes (up to 30 km).

Table 2 – Calculation of average temperature

Time increment	Mach №	Surface temperature	Incremental temperature	Average temperature
(sec)		(K)	(K)	(K)
0	0.00	288.15	0	
40	0.73	306.33	18.18	18.18
20.6	1.48	339.66	33.33	23.33
20	2.49	444.42	104.76	43.54
20	3.91	573	128.58	60.44
7.04	4.39	765.48	192.48	69.07
22.96	5.42	651.64	-113.84	36.92
20	6.77	536.37	-115.26	16.70
13.8	7.86	468.60	-67.77	9.61
16.2	8.99	451.00	-17.60	7.17
15	10.22	391.58	-59.42	2.06

Fairing is designed to ensure that the temperature of the inner surface of the fairing structure does not exceed 10°C during the flight, so we can calculate required ratio of temperatures for determination of Fourier's similarity criterion:

$$\frac{\Delta T_{in}}{\Delta T_{av}} = \frac{10}{60.4} = 0.165$$

According to the recommendations in [2], for temperature ratio 0.165 can be determined value of Fourier's similarity criterion $\varphi = 0.184$.

From equation 4, thickness of aerogel thermal insulation:

$$\delta = \sqrt{\frac{\tau \cdot a_t}{\varphi}} = \sqrt{\frac{\tau \cdot \lambda}{c_o \cdot \rho \cdot \varphi}} = \sqrt{\frac{100.6 \cdot 0.015}{800 \cdot 150 \cdot 0.184}} = 9 \text{ mm}, \tag{7}$$

After determination of insulation thickness it is necessary to verify reliability condition:

$$\frac{h\delta}{\lambda} > 10 \quad \Rightarrow \quad \frac{314 \cdot 0.0152 \cdot 3.91 \cdot 0.009}{0.015} = 11.19 > 10$$

So condition is satisfied. And theoretical calculations provided thickness of thermal insulation equal to 9 mm.

In nowadays for considered class of rockets due to the big cost of full scale experiments for determination of temperature distribution during the flight commonly used numerical simulation in different software packages [10, 11]. In this research for flow simulation also was used software package that implements finite element analysis. Created three dimensional geometrical model of PF is shown in figure 1.

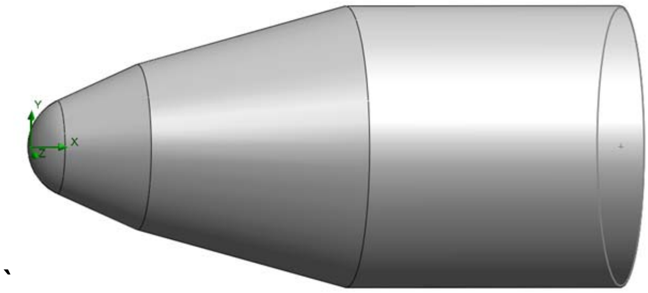

Figure 1 – Geometrical model of insulative skin

Computation domain over the model was created with instantaneous properties of atmosphere and velocity of vehicle according to the standard flight parameters for considered class of rockets [1]. A laminar and turbulent property of air is also varying based on the velocity of vehicle with different altitude.

Based on the FEM model of the insulative shell, the solver performed static analyze over model with surface temperature as main goal. The temperature distribution over the model for altitude 31 km is shown in the figure 2.

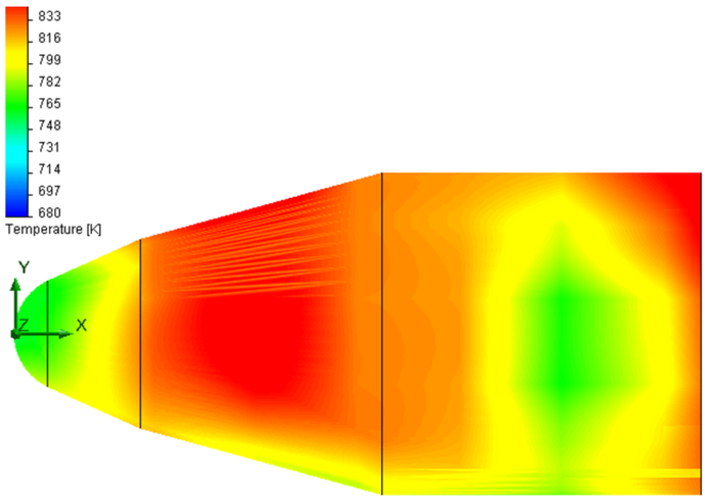

Figure 2 – Distribution of temperature over the model at altitude of 31 km

Based on the series of the flow simulation over the model with various altitudes, the temperature over the surface of PF was obtained as shown in the table 3.

Table 3 – Surface temperature of fairing over different altitude

Altitude (meters)	Surface temperature (K)
0	288.15
4105.3	285.87
10250	307.55
19080	454.92
31090	822.71

The ratio of temperature increment between the elemental altitudes is calculated based on the thermal load from the flow simulation results in table 4, by using the eqn. (5).

Table 4 – Calculation of average temperature

Altitude	Time increment	Mach number	Surface temperature	Incremental temperature	Average temperature
(meters)	(sec)		(K)	(K)	(K)
0	0	0.00	288.15	0.00	
4105.3	40	0.73	285.87	-2.28	-2.28
10250	20.6	1.48	307.55	21.68	5.86
19080	20	2.49	454.92	147.37	40.98
31090	20	3.91	822.71	367.79	105.95

According to the maximal average temperature from table 4 and required temperature of fairing inner surface can be calculated ratio of temperatures for determination of Fourier's similarity criterion [2]:

$$\frac{\Delta T_{in}}{\Delta T_{av}} = \frac{10}{105.95} = 0.096 \;\Rightarrow\; \varphi = 0.136$$

From equation (4) with determined Fourier's similarity criterion, thickness of aerogel thermal insulation is

$$\delta = \sqrt{\frac{\tau \cdot a_t}{\varphi}} = \sqrt{\frac{\tau \cdot \lambda}{c_o \cdot \rho \cdot \varphi}} = \sqrt{\frac{100.6 \cdot 0.015}{800 \cdot 150 \cdot 0.136}} = 10 \text{ mm}$$

Condition for reliable insulation thickness

$$\frac{h\delta}{\lambda} > 10 \quad \Rightarrow \quad \frac{314 \times 0.0152 \times 3.91 \times 0.01}{0.015} = 12.44 > 10 .$$

With the achieved thickness of insulation from experimental calculation, the condition ($\frac{h\delta}{\lambda} > 10$) is also satisfied. Hence insulation can withstand in the elevated temperature and it can maintain the internal temperature as required.

Conclusion

The following conclusions can be obtained on the basis of the conducted research:

1. Thermal load acting over the surface of payload fairing was calculated theoretically.
2. Based on the thermal loads, required insulation material was selected and the required thickness was also calculated.

3. Due to the limitation in the flow simulation packages, the experimental calculations were done up to 30 km only and this values are compared with the analytical calculations.

Results achieved from the theoretical and experimental calculations are almost likes similar in values. So these results are verified and it can provide the necessary strength to withstand in elevated temperature and also it can maintain the temperature inside the payload fairing as required.

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Рецензент: канд. техн. наук., доц. А.В. Кондратьев, Национальный аэрокосмический университет им. Н.Е. Жуковского «ХАИ», г. Харьков.