EFFECT OF AERODYNAMIC HEATING ON SEPARATION MECHANISM USED IN ROCKETS AND NEED OF THERMAL PROTECTION FOR SUCH STRUCTURES

B. Sundar*, T.V. Radhakrishnan*, George Joseph* and A.K. Verma*

Abstract

During the ascent flight of a rocket, when the vehicle passes through the dense atmosphere, the outer skin of the structure is subjected to aerodynamic heating loads. Due to this heating load, the structure is heated and the strength of the structure deteriorates rapidly due to temperature rise of the structure. The structure needs to be thermally protected to restrict the temperature rise in order to maintain its structural integrity. However, in certain components of rockets, like in separation mechanism that is heated due to aerodynamic heating loads, the structure also needs thermal protection to minimize the large temperature gradients which if not controlled can result in thermal expansion and jamming of the separation system. Therefore it is very important to thermally protect such structures from aerodynamic heating to minimize the temperature gradients across the structure and avoid a likely mission failure.

In this paper, a theoretical study is carried out to study the separation mechanism of a rocket which is subjected to aerodynamic heating. Methodology to estimate the aerodynamic heating using the actual flight trajectory of the rocket is also detailed. A 1-D approach is used to estimate the temperature rise of the joint using finite difference schemes. The joint is also idealised using a 2-D finite element mesh to carry out the thermal response analysis of the mechanism. Comparison of the results of the analysis shows that 2-D approach gives a realistic picture of the temperature distribution across the separation mechanism compared to 1-D approach.

Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
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<tbody>
<tr>
<td>A &amp; B</td>
<td>defined by Equation (1)</td>
</tr>
<tr>
<td>$C_f$</td>
<td>local skin friction coefficient</td>
</tr>
<tr>
<td>$C_h$</td>
<td>Stanton numbers</td>
</tr>
<tr>
<td>$F_1, F_2$</td>
<td>defined by Equation (3)</td>
</tr>
<tr>
<td>$h$</td>
<td>convective heat transfer coefficient</td>
</tr>
<tr>
<td>$H$</td>
<td>enthalpy</td>
</tr>
<tr>
<td>$M$</td>
<td>Mach number</td>
</tr>
<tr>
<td>$Pr$</td>
<td>Prandtl number</td>
</tr>
<tr>
<td>$q$</td>
<td>heat flux</td>
</tr>
<tr>
<td>$r$</td>
<td>recovery factor</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>$S$</td>
<td>distance along body measured from stagnation point</td>
</tr>
<tr>
<td>$T$</td>
<td>temperature</td>
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Greek Symbols

<table>
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<th>Symbol</th>
<th>Description</th>
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<tbody>
<tr>
<td>$\gamma$</td>
<td>specific heat ratio</td>
</tr>
<tr>
<td>$\gamma_{af}$</td>
<td>Reynolds analogy factor</td>
</tr>
<tr>
<td>$\lambda_{tr}$</td>
<td>Intermittency factor</td>
</tr>
<tr>
<td>$\rho$</td>
<td>mass density</td>
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Subscripts

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<th>Symbol</th>
<th>Description</th>
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<tr>
<td>e</td>
<td>evaluated at edge of the boundary layer</td>
</tr>
<tr>
<td>rec</td>
<td>evaluated at recovery conditions</td>
</tr>
<tr>
<td>w</td>
<td>evaluated at wall conditions</td>
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Introduction

Several experiments are carried out to study the physics of the upper atmosphere. To carry out such experiments, rockets of size of 100mm, 200mm and 560mm in diameter are usually employed. Such rockets carry small payloads to carry out the experiments in the upper atmosphere. The duration of rockets in atmosphere is of the order of few minutes. The altitude attained is of the order of 450 km.

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km during such short duration. The payloads are enclosed by fairings to protect them from adverse heating conditions during flight.

During the atmospheric flight the rocket structure is subjected to aerodynamic heating. These rockets like launch vehicles get the necessary thrust in two or three stages. The spent stages are separated from the on going stage using a separation mechanism. The payload is housed in the nose cone region of the rocket. When the rocket reaches the predetermined altitude where the scientific experiments are planned, command for separation is given and the nose cone is ejected using a separation mechanism.

The command for separation is given at a predetermined time. If the separation event does not occur at predetermined time then the study of certain physical phenomena is not possible. This constitutes a serious anomaly and calls for detailed investigation.

In this paper, a case study of a rocket has been taken where separation of the nosecone got delayed and the event happened at a later time. The reasons for the anomaly could be many and the effect of thermal expansion of the separation mechanism could be one of them. Fig.1 shows the separation system geometry used in the rocket.

The release is achieved through a ball lock release mechanism. A pre-compressed spring is placed between the nose cone fairing and the bipod place over the upper end of the sustainer, which provides the necessary energy for separation. The bipod is held pressed to the continuing stage by the compressed spring until the spring gets relaxed completely and then it is pulled along with the nose cone. The springs are activated by pyros which fire in the opposite direction thereby releasing the ball into the socket.

After the releasing of the ball, top ring which is a part of the nose cone slides over the bottom ring and finally gets ejected.

The paper describes in detail the thermal environments during the ascent phase of the rocket and the theoretical modeling of the separation mechanism using Finite Element Method (FEM) and Finite Difference Schemes. The temperature distribution across the separation mechanism is estimated and the results are compared and analysed. The paper also highlights the need for providing thermal protection to such structures which may not require thermal protection from strength consideration instead may require to reduce the temperature gradient across such structure which can expand unevenly leading to jamming of the structures.

![Fig. 1 Separation system geometry](image-url)
Computation of Aerodynamic Heating Rates on the Rocket Body

Theoretically, convective heat transfer can be calculated by solving the Navier-Stokes equations which completely describe the flow field over a body using conservation law of mass, momentum and energy. These equations are extremely complex and their numerical solution requires huge computer resources. Alternatively when the viscous-inviscid interactions (like flow separation, shock interference) are not strong it is convenient and economical to obtain inviscid flow field data from wind tunnel tests or Euler codes and use these as inputs in a boundary layer solver.

A large number of codes are available in-house which can give heat transfer rates on the body for different flow conditions. However, these codes cannot be used for routine design computations since they require enormous computer time. Moreover, aerodynamic heating analysis involves computation of convective heat flux on a body along trajectory and thermal response of the structure to this heat flux. These two problems are coupled since the wall temperature affects the flow and vice-versa. The use of boundary layer codes for trajectory wise analysis thus becomes highly time consuming and costly. Therefore for design purpose it is generally sufficient to use simple and reasonably accurate engineering methods for estimating the heat transfer rates over the rocket body in different flow regimes.

To this end, a software code ‘One Dimensional Thermal Analysis Package’ (OTAP) has been developed at Vikram Sarabhai Space Center (VSSC) for the computation of heat transfer rates over the body and its thermal response along a given trajectory. The design code ‘OTAP’ has been developed as an integrated package by patching together applicable theories for various regions like stagnation regions and aft body and for different flow conditions like laminar and turbulent flows in continuum and free molecular flow regime. For ascent phase of the flight of a rocket, however, heating in the continuum flow regime is important as far as the thermal design/analysis is concerned.

Heating rates on Cylinder

For the computation of convective heating rates on cylindrical after bodies of rocket, the design code ‘OTAP’ makes use of correlations of heating rates on flat plate. From among the various correlations methods available in the literature, the method proposed by Van Driest given in Ref.[1] is selected. Codes to solve the turbulent boundary layer equations modeled by Cebeci and Smith have been also developed. The results of these codes have been compared with those obtained from the correlations.

Turbulent Flow

Heat transfer rates are obtained for compressible turbulent flow over a flat plate by using Reynolds’s Analogy along with a prediction for skin friction. The skin friction, $C_h$, is computed as

$$\frac{0.242 (\sin^{-1} \alpha + \sin^{-1} \delta)}{A \sqrt{\frac{C_f T_w}{T_e}}} = 0.41 + \log (Re_f, C_f)$$

$$- (n + 0.5) \log \left( \frac{T_w}{T_e} \right)$$

where

$$\alpha = \frac{2A^2 - B}{B^2 + 4A^2} ; \quad \delta = \frac{B}{\sqrt{B^2 + 4A^2}}$$

$$A^2 = \frac{(\gamma - 1) M_e^2}{2T_w/T_e} ; \quad B = \frac{T_w}{T_e} - 1$$

(1)

Stanton number, $C_h$ is given by $C_h = \gamma df C_f/2.0$. The heat flux, $q$, is given by $q = C_h \rho_c u_c (H_{rec} - H_w)$ Where, $H_{rec}$ the enthalpy at recovery temperature is given by

$$H_{rec} = H_e + 0.5 r u_e^2$$

(2)

Laminar Flow

For laminar, the Stanton number, $C_h$, is computed as given in Ref.2.

$$C_h = \left( F_1 + F_2 (T_w/T_e) \right) Pr^{0.5}$$

(3)

The recovery factor is taken as $Pr^{0.5}$. 
Boundary Layer Transition

Heat transfer in the transition phase is calculated using the weighted average between laminar and turbulent heating. The weighting factor being used is the Intermittency factor, which is calculated following Chen and Thyson as given in Ref. [3]. Heat transfer rates in the transition zone is calculated as

\[ q_{\text{trans}} = \lambda_n q_{\text{turbulent}} + (1 - \lambda_n) q_{\text{laminar}} \]  

Validation of Engineering Methods

In order to check the validity of engineering methods for the prediction of heat transfer rates, thermal response analysis was carried out for the typical flight trajectory using design code ‘OTAP’ and the predicted skin temperature data compared with flight measured data on the cylindrical region [4 and 5]. Fig.2 shows the typical flight trajectory of a rocket as a function of time. Fig.3 shows the comparison of the measured temperature history made on the rocket body along with predicted temperature history on a cylindrical location for the flight trajectory. The comparison is fairly good. A fairly good agreement seen between the theoretical predictions and flight measured data has shown the adequacy of design code ‘OTAP’ as useful design tool for thermal design/analysis of a rocket.

Using the design code ‘OTAP’, the cold wall heat flux history on the cylindrical portion of the rocket body housing the separation mechanism is estimated considering the actual flight trajectory and is given in Fig.4.

Theoretical Modeling

Two approaches were followed to study the thermal response of the separation mechanism subjected to aerodynamic heating. Temperature history at different locations of the separation mechanism is estimated using one dimensional fully implicit finite difference scheme. A finite element method is also employed to thermally map the temperature field across the separation mechanism.

The governing equations employed and the solution procedure employed in the two approaches is detailed below.

Time wise variation of the heat flux is seen in both the approaches. Radiative loss to the ambient from the structure is also considered. The initial temperature of the structure is taken as 313K. The analysis is carried out for a duration of 218 seconds. The results obtained from the two approaches are compared and discussed in the subsequent sections.

Finite Difference Model

For thermal design and analysis of sounding rockets it is usually sufficient to solve the heat conduction equation in one dimension.

For materials whose thermal properties are dependent on temperature and with non-linear boundary conditions, one has to resort to numerical methods. Software package ‘OTAP’ developed in house is used for the solution of one-dimensional problems. This software package uses fully implicit finite difference scheme.
The elements are assembled into global matrices and the final system of equation is obtained as

\[ [K] (T) + [C] (\partial T/\partial t) = (F) \]  \hspace{1cm} (7)

Present code uses 4 noded linear quadrilateral elements for finite element modeling. Fig 5 shows the 2-D finite element model of the separation mechanism.

The finite element software package (NISA Package) employs direct integration method \([6]\). There are various schemes in direct integration method to integrate the above equations directly. Out of these, fully implicit scheme is chosen because it allows for larger time steps with less oscillations and also the numerical approach is closer to the physical phenomenon \([7\ and \ 8]\).

**Results and Discussion**

The analysis was carried out using one dimensional finite difference scheme and two-dimensional finite element method (FEM) and the results are reported in \([9]\). Fig 6 shows the temperature history of separation plane at two locations A and B as shown in Fig 5 using 1-D finite difference scheme. The temperature history at location A is higher than predicted at location B. Location B is the region of the criticality where the tongue and groove joint is formed. It is seen that there is no temperature gradient across the thickness at location B. However in 1-D model, the effect of conduction due to the presence of adjacent structures cannot be modeled and hence a 2-D approach using FEM is adopted which can take care of axial conduction.

Figure 7 and 8 show the temperature distribution in the separation system at 53 seconds and 218 seconds respectively. The temperature distributions at these particular instances of time are considered because the event of separation was planned at 53 seconds but it occurred at 218 seconds. It is seen that maximum temperature near the joints at 53 seconds is 407K. It is to be noted that there is an axial gradient of 63K existing across the mechanism. At 218 seconds the maximum temperature near the joints is 391.4K and axial temperature gradient is 5K. It can also be observed that joint stabilizes to a uniform temperature at 218 seconds compared to 53 seconds where large axial temperature gradients exists.

It is also to be noted that the primary structural material used for the separation mechanism is steel which can withstand temperature up to 573K without any substantial loss of strength. Temperature history computed using 1-D
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Fig. 5 2-D finite element model of the nose cone separation system

Fig. 6 Temperature history on the separation system using one-dimensional analysis

Fig. 7 Temperature distribution in the separation system at 53s

Fig. 8 Temperature distribution in the separation system at 217s
approach for a minimum skin thickness of 2.5mm (Location A) shows that the maximum temperature reached is 474K which does not exceed the temperature constraint of 573K and hence no thermal protection is required for the structure. Adjacent structures shows lower temperature rise because of the large thermal mass available (Location B). This is the approach adopted for thermal design of rocket structures where minimum skin thickness is considered and the thermal response of the structure is evaluated. However, in this particular case study it is clearly seen that 1-D approach used for design is not appropriate as it does not take care of the temperature gradients existing in the structure due to axial conduction. These temperature gradients can possibly cause jamming of the joints and hence thermal protection is required to reduce the temperature gradients.

Expansion of the separation system can occur due to temperature gradients, namely in radial and longitudinal directions. The expansion in the radial direction can only result in increase in the clearance between the joints aiding smooth separation. However, the expansion in the longitudinal direction can possibly jam the separation mechanism due to restriction of movement in that direction. This is the case at 53 seconds when the temperature gradients are high. However, at 218 seconds when the axial temperature gradients are lower, the structure could have regained its original configuration thereby leading to clean separation.

Conclusions

Thermal environments of the separation mechanism during flight is evaluated and the temperature distribution across the mechanism using 1-D finite difference scheme and 2-D finite element method was estimated. 1-D approach is not sufficient to give a correct picture of the temperature distribution of the separation system. 2-D approach using FEM indicates that large temperature gradient exists at the scheduled time of separation and the gradient vanish at a later time leading to clean separation. Hence it is clearly seen that the even though the structure temperature may not exceed the operating temperature limits, it may still require thermal protection to reduce the temperature gradients existing across the structure. In such cases where the separation mechanism are involved it is important that 2-D approach is adopted to carry out the thermal design of the structure and provide adequate thermal protection on such structures to reduce the temperature gradients and to ensure smooth separation of the joints.

References


