TECHNICAL NOTE

THERMO STRUCTURAL INTEGRITY OF A CLOSURE ASSEMBLY FOR THE BASE REGION OF A RE-ENTRY MODULE

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Abstract

Base heating is an important phenomenon in the design of re-entry capsules. The nature of reverse flow in the base region is largely decided by the conditions prevailing at the end of the fore body. For configuration like the re-entry module, the reverse flow is found to be supersonic near the base. It is therefore necessary to protect the base region as well as the internal components from base heating.

Suitable Thermal Protection System (TPS) of minimum mass that will protect the base thermally is to be designed for the base heating levels during re-entry. Low density ablative is selected as the TPS material.

The base region of the re-entry module consists of a closure assembly made up of structural elements which are thermally protected. The structural elements and TPS have different thermo-physical properties. It is necessary to qualify the closure assembly for re-entry loads. However, qualification of closure assembly is a challenging job as there is no facility in the world which can simulate all the time varying flight parameters simultaneously. Hence to study the integrity of the closure assembly during re-entry, detailed temperature mapping of closure assembly was carried out.

Thermal mapping of the closure assembly was carried out considering both axisymmetric and three-dimensional model using finite element package and the results are compared. Maximum temperatures attained at different locations of the assembly are discussed. Temperature distribution of the closure assembly at critical instants of flight was estimated. Thermo-structural analysis carried out showed the adequacy of the design, which was subsequently demonstrated by a successful re-entry flight.

Nomenclature

C_p	= specific heat at constant pressure (J/kg/K
k	= thermal conductivity (W/mK)
n	= direction normal to the surface
N_i	= shape factor
h	= convective heat transfer coefficient (W/m^{2}K)
q	= heat flux (W/m^2)
T	= temperature (K)
t	= time (sec)
ρ	= density (kg/m^3)
ε	= emissivity
σ	= stefan boltzmann constant (W/m^2/K^4)
α	= thermal diffusivity = $\dfrac{k}{\rho\, C_p}\ (m^2/s)$
M	= Mach number

Superscript

e	= element
s	= surface
r	= no. of nodes of an element

Subscript

x, y, z	= cartesian direction
s	= surface

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f = fluid
o = initial
∞ = infinity

Introduction

During the re-entry of the module at 28800km/hr, the base region is exposed to convective heating due to reverse flow of gases. Base heating is an important phenomenon in the design of re-entry capsules. For slender bodies, literature data indicates that heat flux in the base region is 6 % of the flare-end value [1]. However for blunt re-entry modules heat flux values are to be evaluated using CFD approaches.

Annular area at the base of the re-entry module is closed using a closure assembly as shown in Fig.1 and is made up of Carbon Fiber Reinforced Plastic (CFRP) composite panels. The composite panels are provided with Silica based Low Density Ablative (LDA) as TPS for the base region of the re-entry module.

It is essential to qualify the closure assembly for re-entry loads. However, qualification of closure assembly is a challenging job as there is no facility in the world, which can simulate all the time varying flight parameters simultaneously. Hence to study the integrity of the closure assembly during re-entry and accept the closure assembly for flight, detailed temperature mapping of closure assembly was carried out.

An axisymmetric model is sufficient for thermal mapping of annular closure assembly due to symmetry of the structure. However, to see the effect of supporting brackets, a 60 degree sector is modeled using three-dimensional finite elements. The thermal analysis of the axisymmetric and three-dimensional model of annular closure assembly was carried out using NISA [2] software and the results are compared. Maximum temperatures attained at different locations of the assembly are discussed and thermo-structural studies were carried out at critical instants of time.

Fig.1 Base Configuration of Re-entry Module

Governing Equations

The heat conduction equation derived from the law of conservation of energy is obtained by considering the heat flow to the interior of the body.

In Cartesian coordinate system, three dimensional transient heat conduction equation is given by

$$\frac{\partial}{\partial x}\left(k_x \frac{\partial T}{\partial x}\right) + \frac{\partial}{\partial y}\left(k_y \frac{\partial T}{\partial y}\right) + \frac{\partial}{\partial z}\left(k_z \frac{\partial T}{\partial z}\right) = \rho C_p \frac{\partial T}{\partial t} \qquad (1)$$

where

$$T = T\,(x\,,\,y,\,z\,,\,t\,) \qquad (2)$$

Initial temperature of the structure is taken as To. On external surfaces, convective heating as well as radiative loss from the surface is considered which is given by :

$$-k\frac{\partial T}{\partial n} = q^s\,(t) - \sigma\varepsilon\,(T_s)^4 \qquad (3)$$

where $q^s\,(t)$ is obtained from CFD analysis.

This equation can be rewritten as

$$-k\frac{\partial T}{\partial n} = h\,(\,x\,,\,t\,)\,(\,T_s\,(\,t\,) - T_f\,(t)) - \sigma\varepsilon\,(T_s)^4 \qquad (4)$$

It is to be noted that convective heat transfer coefficient, h, varies with time and location. Fluid temperature also varies with time. On internal surfaces, adiabatic condition is considered.

Perfect contact is assumed between various TPS materials and base structure which means continuity of temperature and heatflux at the interfaces and this will result in estimating the interior temperatures marginally on the higher side.

Numerical Modeling

In finite element formulation, temperature distribution over an element 'e' is assumed as

$$T^{e}(x, y, t) = {}^{r}\sum_{i=1} N_{i}(x, y)\, T_{i}(t) \tag{5}$$

Substituting in variational form and minimizing with respect to all nodes one gets

$$[K](T)^{e} + [C]^{e}(\partial T / \partial t)^{e} = (F)^{e} \tag{6}$$

The elements are assembled in to global matrices and the final system of equation is obtained as

$$[K](T) + [C](\partial T / \partial t) = (F) \tag{7}$$

where [K] is the conductivity matrix, [C] is the capacitance matrix and (F) is the load vector. In the present formulation 20 noded hexahedron is used for three-dimensional modeling and 4 noded linear quadrilateral elements is used for axi-symmetric finite element modeling.

The finite element software package (NISA Package) uses the wave front technique for the solution of the overall finite element equilibrium equations, which are in the form of simultaneous linearised algebraic equations. The wave front technique uses the Guass elimination method for the solution of simultaneous linear equations.

Finite Element Model

The closure assembly closes the annular gap at the base region of re-entry module between the floatation container and internal structure and thermally protects the component from the re-entry heating. Axisymmetric and three dimensional finite element models of the annular closure assembly are shown in Fig.2 and 3 respectively.

Fig.2 Axisymmetric Finite Element Model of Closure Assembly

Thermal Environments

Detailed CFD simulations using UNS2D was carried out simulating the base geometry. The code solves the Navier-Stokes equations using finite volume method. The code uses some of the state-of-the-art numerical techniques like upwinding with Van Leer Flux vector splitting and implicit time integration for getting accurate results in a reasonable amount of computational time. CFD simulations were carried out for different flight conditions. It is seen that for a flow speed of M=20 at an altitude of 50 km, heatflux as well as the air temperature reaches maximum value in the base region. Fig.4 shows the Mach number palette around the body at peak heating condition. Reverse flow Mach Number is of the order of 2 at this condition. Fig. 5 shows the corresponding gas temperature palette for the computational domain. Maximum temperature in the base is 4300K ($\frac{T}{T_{\infty}} = 15.94$). For this condition, base

Fig.3 Three Dimensional Finite Element Model of Closure Assembly

Fig.4 Mach Number Palette at Peak Heating Condition heatflux is 5% of the corresponding fore body stagnation point value. Based on CFD studies carried out different flight conditions, base heating levels were estimated as function of forward stagnation point heatflux and is used for thermal analysis.

Figure 6 shows the heat flux history on annular closure as well as on the interface ring. Heat flux along the sides of the interface ring is computed and variation is given in Fig.7. The re-entry heatflux is computed from a re-entry altitude of 120 km till touch down. For re-entry from low earth orbits, radiation heating from the high temperature gas is negligible compared to aerodynamic heating. However the radiative heatflux is also considered. Radiative heatflux is computed based on the correlation [3]. Maximum radiative heatflux is 0.3 W/cm^2.

Fig.5 Temperature Palette at Peak Heating Condition

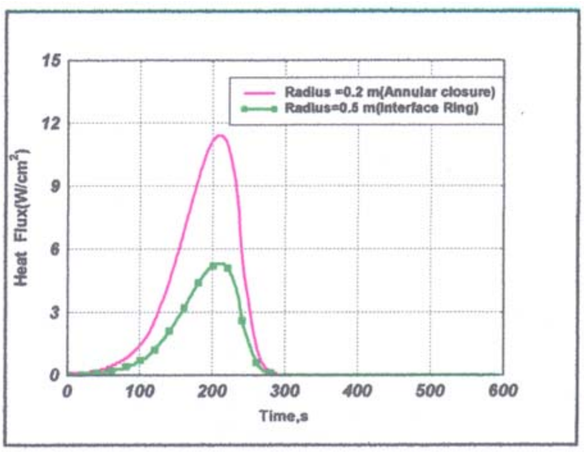

Fig.6 Cold Wall Convective Heat Flux History on Annular Closure and on Interface Ring

Thermal Response Analysis

Axi-symmetric and three-dimensional thermal analysis for re-entry module closure assembly is carried out using finite element package (NISA). The initial temperature of the structure at re-entry altitude (120 Km) is taken as 303 K. The annular area at the base of the re-entry capsule is closed using annular closure system and is made of composite panels. The Thrust cylinder made of aluminium alloy honey comb structure. The composite panels are provided with Low-density ablative (LDA) as thermal protection system for the re-entry capsules base region. The gap between the payload interface ring and annular deck is filled using flexible insulation. Thermo physical properties (ρ, C_p and k) of various materials for varying temperature and pressure conditions are considered in the analysis as shown in Table-1. Thermal response analysis is carried out for duration of 600s from 120 km till touch down.

Results and Discussions

Axisymmetric and three-dimensional thermal response analysis was carried out and the results were compared. Fig.8 shows the comparison of temperature histories between the two models for a typical location [F in Fig.2]. Fig.9 shows the comparison of temperature histories between the two models for a location at the base end of interface ring [A in Fig.2]. It is seen from the above figures that the results from the two models fairly agree with each other. It is also seen from Fig.9 that the maximum temperature at the base end of the ring is 539K at 236 seconds. Thermal protection at the base end of inter-

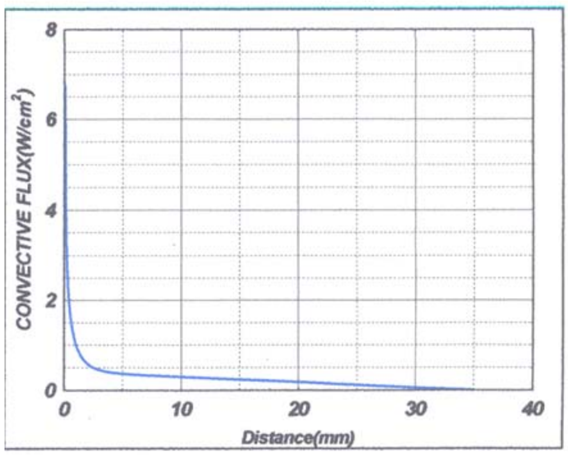

Fig.7 Cold Wall Convective Heat Flux History along the Sides of the Interface Ring

face ring is not possible due to mission constraints. Since the temperature of the interface ring is higher than the limit of 423K, the mounting brackets and pyro systems, which were mounted on the interface ring were shifted up where the temperature levels are well within the limits. Thermo-structural analysis was carried out for this condition considering thermal and pressure loads. Stress in the interface ring exceeded the allowable limit for less than 1mm depth. Hence, the small region may yield locally. The maximum temperature at the interface [E in Fig. 2] of bracket and the interface ring is 413K at 275 seconds. Figs. 10 to 12 shows the temperature history at various interfaces. It is seen in all the cases that the temperature limit of 423K is not violated till the end of flight. Temperature distribution based on the three dimensional analysis at 212 second is shown in Fig.13. This is one of the critical events in flight identified for thermo-structural analysis as the surface temperature of LDA reaches as maximum of 1188K.

Table-1 : Thermo-Physical Properties of Materials

Material	Density (kg/m^3)	Specific Heat (J/kg K)	Thermal Conductivity (W/mK)	Emissivity
Al alloy	2800	963	121	0.68
Thrust cylinder	100	920.0	k_y, k_z = 3.85 k_x = 0.70	-
CFRP Honey comb	255	920.0	k_y = 0.81 k_x = 2.43	-
Flexible Insulation	270	921.0-1130.0	0.027 - 0.034	0.64
Low density ablative	380	1290-1690	0.129 - 0.264	0.68

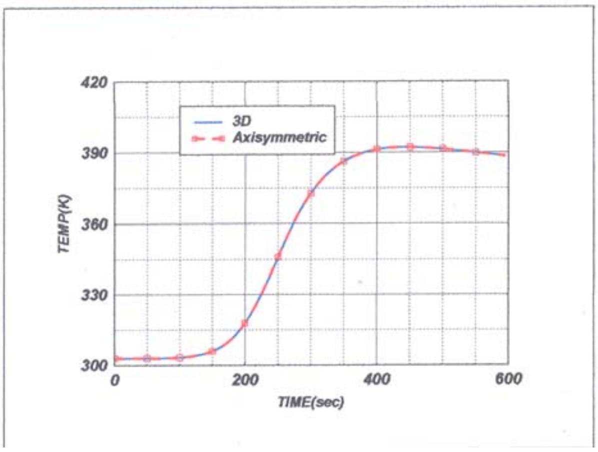

Fig.8 Temperature Histories at the Interface of Interface Ring and Thrust Cylinder

Compressive stresses in LDA exceed the strength for a depth of 2 mm. This can cause local bulging in LDA. It is to be noted that available virgin LDA material is 19 mm thick and stresses are well within the limits. Stresses in backup structure are within the allowable limits.

Conclusions

Thermal environments of the closure assembly in the base region of the module have been evaluated using CFD techniques. Base heating levels were estimated for the re-entry mission based on the ratio derived from CFD

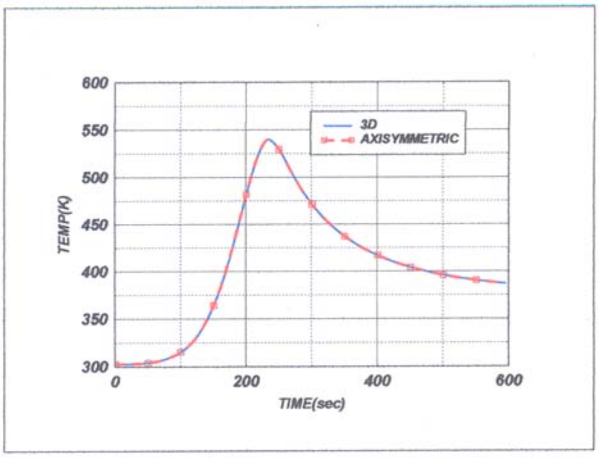

Fig.9 Temperature Histories at the Base End of the Interface Ring

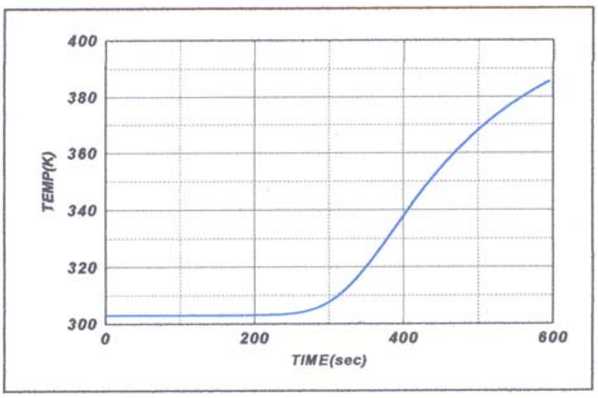

Fig.10 Temperature History at the Interface of LDA and Backup Structure

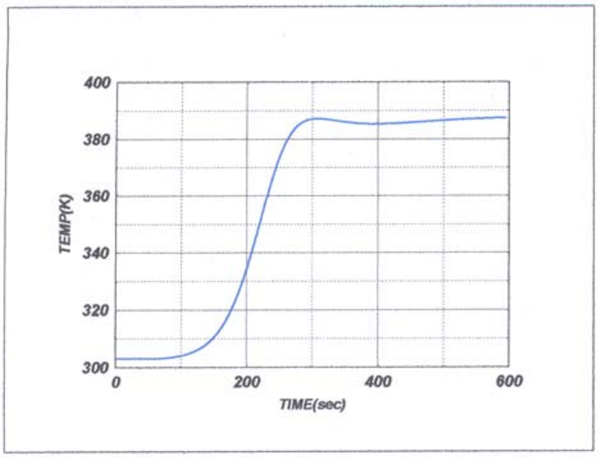

Fig.11 Temperature History at the Interface of Bracket and Backup Structure

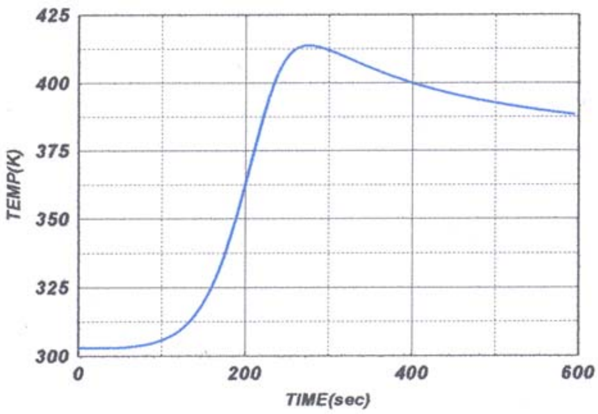

Fig.12 Temperature History at the Interface of Bracket and Interface Ring

Fig.13 Three Dimensional Temperature Distribution of LDA and CFRP at 212s

results. Thermal analysis was carried out considering three-dimensional model of closure assembly and results are compared with axisymmetric model. Maximum temperatures at various interfaces of the structures are within the specified temperature limits proving the adequacy of thermal design.

To study the structural integrity of the closure assembly, detailed temperature distribution of the closure assembly was estimated simulating the flight thermal environments. Since the temperature of the interface ring is higher than the limit of 423K, the mounting brackets and pyro systems, which were mounted on the interface ring were shifted up where the temperature levels are well within the limits. Three dimensional temperature distribution is used for thermo-structural analysis.

Thermo-structural analysis [4] was carried out for this condition considering thermal and pressure loads. Stress in the interface ring is found to exceed the allowable limit for the outer surface and up to 1 mm depth. Thus a small region is expected to exhibit local yielding. Stresses are within limits for the backup structure. Compressive stresses in LDA exceed the strength for a thickness of 2 mm beyond which the stresses are within limits.

Thermo-structural analysis carried out showed that the adequacy of the design which was subsequently demonstrated by a successful re-entry flight.

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