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Transient thermal analysis of Vega launcher structures

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Abstract

A transient thermal analysis is carried out to verify the base cover thermal protection system of Vega 2nd stage Solid Rocket Motor (SRM) and the flange coupling of the inter-stage 2/3. The analysis is performed with a finite element code. The work has developed suitable numerical Fortran subroutines to assign radiation and convection boundary conditions. The thermal behaviour of the structures is presented.

Keywords: Thermal protection; Finite element analysis; Solid Rocket Motor; Radiation and convection

1. Introduction

The materials employed in the aerospace industry must withstand very large temperatures and thermal fluxes. The re-entry capsules suffer a strong increase of temperature due to the friction drag that, if not limited by special devices, could disintegrate the vehicle. The transient thermal response of materials exposed to high temperature environments is a key issue in designing the internal thermal protections of Solid Rocket Motors (SRM) or the heat shield of re-entry vehicles. Moreover, the inside components of SRM are invested by the huge thermal flux of the combustion gas which could destroy the mechanical structure and melt the metals of the system. Some internal mechanical systems (i.e. hydraulic jacks) have to be thermally protected too. Furthermore, the account of thermo-mechanical phenomena in composite materials at high temperatures is of great importance, as discussed by Dimitrienko [1]. In conclusion, dimensioning the structures is important for the safety of the launcher, but the design has to be optimized in order to reduce the mass of the structure.

Traditionally, SRM thermal analysis employs a lumped parameters approach based finite differences scheme of the governing heat transfer equation. Recently, finite element analysis gained widespread popularity for structural analysis. The shift to finite elements for a thermal analysis was possible by vast improvements in computational hardware, as the CAD import, easy mesh generation and intrinsic link with the physical geometry of the structures.

The purpose of the present paper is to study the thermal and structural design and to dimension two internal thermal protection systems of the new European launcher Vega, i.e. base cover and flange coupling. Vega has been designed as a single-body launcher with three solid propulsion stages (P80, Zefiro 23 and Zefiro 9), an additional liquid propulsion upper module (Avum) and a fairing for payload protection. Vega will be able to launch satellites for a wide range of missions and applications with payload masses ranging between 300 and 2500 kg.

The structures investigated are the base cover of the second-stage Zefiro 23 and the inter-stage connecting the second and third stage. Zefiro 23 motor employs a lightweight carbon-epoxy case, with a low density thermal insulation,

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Nomenclature

Latin

c_p	specific heat, J/kg K
T	temperature, K
t	time, s
x, y, z	Cartesian coordinates, m

Greeks

κ	thermal conductivity, W/(m K)
ρ	density, kg/m^3

Subscripts

aero	aero-thermal heat flux
c	convective heat flux
e	infrared radiation heat flux
flight	total heat flux
i	internal
r	incident radiation heat flux
s	surface
w	wall

Acronyms

ETP	external thermal protection
LV	launcher vehicle
SRM	Solid Rocket Motor
TP	thermal protection

charged with glass micro spheres and a moving nozzle, based on flexible-joint technology. The first airframe is aluminum shells, with integrated stiffeners inside, while the inter-stage 2/3 is a conical structure to match the different stage diameters.

The goal of this paper is to use a finite element software to verify the thermal protections of the structures taking into account the boundary conditions. The governing equations and the boundary conditions of the problem are introduced first, then the numerical analysis, the numerical results with the discussion and the conclusions are presented.

2. Governing equations

The general three-dimensional heat transfer equation, in Cartesian coordinates, is the following:

$$\rho c_p \frac{\partial T}{\partial t} = \frac{\partial}{\partial x}\left(\kappa \frac{\partial T}{\partial x}\right) + \frac{\partial}{\partial y}\left(\kappa \frac{\partial T}{\partial y}\right) + \frac{\partial}{\partial z}\left(\kappa \frac{\partial T}{\partial z}\right) + g \qquad (1)$$

The base cover thermal protection system and the flange coupling of Vega second stage are three-dimensional axial symmetric structures which can be studied as two dimensional by investigating a transversal cross section. Furthermore, the axial length of the base cover is much higher than the curvature radius of the cross section allowing to investigate the problem of the base cover as one dimensional, where the coordinate x has the origin on the external surface and is perpendicular to the surface towards the internal side. The local heat fluxes are set perpendicularly to the surface.

The following boundary conditions are assumed:

$$-\kappa \frac{\partial T}{\partial x} = f(T_s, t) \quad \text{on the external surface for } t > 0 \qquad (2)$$

$$T = T_i \quad \text{on } x \geqslant 0 \text{ for } t = 0 \qquad (3)$$

In Eq. (1) ρ is the density, c_p the specific heat, k the thermal conductivity and g the heat generation. In Eq. (2) $f(T_s, t)$ is a function of time and of the surface temperature T_s at $x = 0$.

The following physical properties are assumed for the two materials investigated:

– Silicon rubber, $k = 0.13$ W/(m K), $c_p = 1300$ J/(kg K), $\rho = 670$ kg/m^3.
– Aluminum, $k = 120$ W/(m K), $c_p = 800$ J/(kg K), $\rho = 2700$ kg/m^3.

The commercial finite element software MSC Marc, MacNeal-Schwendler [3], is employed to solve the equations. The MSC software consists of two programs, Marc and Mentat, which work together to generate geometric information, to define the structure, to analyse the model and to depict graphically the results.

Radiation, convection and aero-thermal heat fluxes on the surfaces of the Vega Launcher Vehicle (LV), generally called thermal loads, are taken into account according to Fabrizi et al. [2]. The thermal loads, generally time and space dependent, are related to the LV external surface location (axial and radial) and to the instantaneous flight phase. They are the followings:

1. Aero-thermal heat flux (q_{aero}) is due to aerodynamic heating effects. The relative intensity depends on the external air flow along the LV trajectory, the LV incidence and swing angle and the LV external shape geometrical characteristics (i.e. cone angle of heat shield fairing, protuberance dimensions, stage length and others).
2. Incident radiation heat flux (q_r) is due to the solar radiation and the solid propellant motor exhaust jet radiation. The relative intensity depends mainly on the thermal fluid dynamics of the combustion gas along LV trajectory and the geometrical characteristics of the motor nozzles.
3. Heat flux jet impingement (q_c) is due to the interaction between the external air flow and the solid propellant motor exhaust jet impingement.
4. Infrared radiation heat flux (q_e) is emitted by the LV external surface.