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Carbon/carbon high thickness shell for advanced space vehicles

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ABSTRACT

The aim of this paper is to study a shell structure for space applications. The aim of the structure is to be reusable and thick. This first study will focus on thermal environment. Starting from a defined geometry, a prototype will be manufactured. Thermo-structural behavior of the structure will be analyzed by numerical analysis and tests. Thermal properties, such as thermal conductivity and heat capacity, will be studied by the use of the inverse method. A robust numerical approach, such as inverse method, is one of the best for this problem as many parameters concur for the determination of properties. Such approach permits to perform the parametric and structural identification of the model. These procedures are presented including both experimental investigation and methodical-numerical aspects. Special test equipment and the regularizing algorithm for solving the ill-posed inverse heat conduction problem are described. In the frame of thermal properties determination, a verification and prediction of thermocouple error will be performed. Developed in the frame of this work the experimental and theoretical methodology for complex data acquisition of unsteady thermal state of the prototype of shell structure is proposed.

1. Introduction

High-temperature carbon composite materials, such as carbon-carbon, have several advantages over metals (titanium, steel, copper) in terms of specific strength, rigidity, maximum operating temperature, thermal erosion resistance, etc. They are widely used as heat-shielding coatings for spacecraft, especially in conditions of vacuum, inert medium, minimized impact of combustion products of fuel, since under the given conditions they are able to increase with increasing temperature: the effective enthalpy of the erosion failure.

The idea to use carbon-carbon composite materials as a thermal protection coatings is not a novel. For example, a reinforced carbon-carbon thermal protection system were used on the Space Shuttle Orbiter vehicle's nose cap and wing leading edge regions where temperatures reach 1538 °C [1].

The paper [2] describes the preliminary design, development, and initial testing of prototype carbon-carbon components of the thermal protection system of the NASA STARPROBE spacecraft. The thermal protection system is designed to limit the spacecraft scientific instrument package to a maximum temperature of 55C during the time of closest approach to the Earth's Sun when the peak radiative heating rate reaches 393 W/cm^2. The thermal protection system is comprised of a carbon-carbon thin shell primary shield and two carbon-carbon sandwich construction secondary shields which block reradiation from the primary shield to the spacecraft. These shields are joined and stiffened by carbon-carbon structural members and attached to the spacecraft with eight thin-walled carbon-carbon struts.

In the [3] it has been shown, that for Mars exploration mission the nonablative lightweight thermal protection system consisted of a carbon-carbon (C—C) composite skin, insulator tiles, and a honeycomb sandwich panel could be used.

The effect of thermal deformation on the strength of models of thermal protective coating at temperatures above 1000 °C was studied in the work [4]. The coating is of shell type and made of a carbon-carbon composite with varying winding angle. The physical and mechanical characteristics of the composite were determined experimentally in simulated real conditions. The stress–strain state of a thermal protective cylindrical shell of length 340 mm, mid-surface radius 90 mm, and thickness 10 mm subject to heating is solved using the three-dimensional theory of elastic-

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Nomenclature

Latin symbols

$c(T)$	Heat capacity
$f_m(\tau)$	temperature measurements
$\overline{g^s}$	Vector of gradient minimization method at current iteration
J	Leas-square minimized functional
$\bar{J}_p'^{(s)}$	Gradient of the functional J at current iteration
M	Number of thermocouples
$\bar{p}$	Vector of unknown parameters
$q_1(\tau)$	Heat flux at the heated boundary
$q_2(\tau)$	Heat flux at the inner boundary
$q_{w_i}(\tau)$	Heat flux at selected points at the surface
$T(\tau, x)$	Temperature
$T_0(x)$	Initial temperature
$T_{w_i}(\tau)$	Temperature at selected points at the surface
x	Spatial variable
$x_1, x_2, \ldots, x_m$	Coordinate of thermocouples

Greek symbols

α_1, α_2, β_1, β_2	Coefficients determined kind of boundary conditions (1 for left, 2 for right)
$\overline{\gamma^s}$	Step descent
$\Delta T(x, \tau)$	Increment of temperature
δ_f	Integral error of temperature measurements
$\lambda(T)$	Thermal conductivity
ρ_c	Density after thermokinetic process
$\sigma_m(\tau)$	Measurement variance
τ	Time
τ_m	Final time
$\psi(x, \tau)$	Adjoint variable for temperature
$\theta(x, \tau)$	Increment of density

ity. The results obtained allow determining the stress state of a thermal protective shell-type coating depending on the type and winding angle of the carbon reinforcement within the temperature range from 20 to 1200 °C.

Carbon-carbon is lightweight, retains its strength at high temperatures, has high tailorable thermal conductivity, and exhibits low wear from room temperature to high temperatures. These characteristics make the carbon-carbon composites attractive candidates as advanced thermal system materials. Primarily, the composites are employed in the aerospace industry thereby capitalizing on their auspicious thermal capabilities. Due to their excellent mechanical, thermal, wear, and frictional properties, the carbon-carbon composites are great candidates in today's brake industries in aviation and some automotive industries. Applications requiring thermal management or system elements needing high temperature stability, including rocket nozzles and exit cones, also benefit from the desirable carbon-carbon composite qualities [5].

There are multiple options for dealing with the severe thermal environments encountered during hypersonic flight. Passive, semi-passive, and actively cooled approaches can be utilized. Among passive technology Space Shuttle Orbiter elevons are the best example of reusable technology which does not use liquid coolant for thermal control, in fact, the use of liquid coolants is efficient but there is an increase of weights and failure risks.

Hot structures are also more durable than current tile and blanket TPS, are easier to inspect, and require less maintenance and repair [6].

The use of laminated composite shells as hot structures in many engineering applications has been expanding rapidly in the past four decades due to their higher strength and stiffness to weight ratios when compared to most metallic materials. Composite shells now constitute a large percentage of recent aerospace or submarine structures. When shell structure is applied to space, the most challenging structure are the ones used for the wing leading edge, vehicle nose of vehicles and windward connections. The extreme temperatures on these components brought the technology to develop ceramic materials able to withstand the harsh reentry environment conditions. In this preliminary study a C/C shell has been manufactured and investigated in order to understand its behavior at high temperatures.

2. Material and methods

The C/C shell was manufactured on the basis of the experience obtained by previous studies [7]. A 6 K twill textile produced by Microtex was used in order to constitute the 3D preform. The preform was then infiltrated with the Isothermal Chemical Vapour Deposition method. This method allows the deposit of quite a variety of matrices and the processing of any complex shape. The multi-directional fiber preform is fixed in the furnace with a tooling or build up with a temporary binding agent, which is removed in a first pyrolysis step. The ceramic matrix is obtained by the decomposition of gaseous species within the open porosity of the preform. The precursor used was methane CH_4.

The preform was uniformly heated and some infiltration cycles were applied on the sample in order to intensify it.

The shell was then refined by grinding operation before performing the tests for the thermal properties determination by the use of the inverse method.

Three small samples were taken from the shell in order to determine the thermal expansion coefficient. This was done by the use of a LINSEIS push rod dilatometer, L75H (Fig. 1) operating in a vacuum environment at Sapienza University of Rome. The test followed the ASTM C1470.

The problem of carbon-carbon material characterization (in fact estimating the thermal properties by inverse heat conduction problems (IHCP) technique [8–10]) was considered on the above mentioned shell sample. The cylindrical specimen of C—C material was submitted for thermal tests as a half-pipe (Fig. 2). The geometrical parameters, weight and size of specimens are presented in

Fig. 1. Push rod facility for thermal expansion coefficient determination.