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A new design proposal and thermal analysis of subassemblies inside a satellite

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Abstract. On this study the authors aim to validate the use of CFRP composites in a subassembly of a satellite as a replacement for currently used aluminum. Considering the high costs of sending a mass unit of payload into orbit, the mass should be minimized. Therefore, the application of composite materials in electronics housing structures was proposed.

For this, a new design is proposed and a comparative thermal analysis of the two boxes is made. Present study focusses on a subassembly's surfaces and materials of a satellite structure subject to thermal variations in order to numerricaly validate the new design. A satellite electronic housing was subject to a three-dimensional finite element analysis. The satellite's thermal loading in an orbit of sun-tracking mode are partially transfered to the electronic housing inside. A model is developed and the temperature distribution in the satellite subassembly predicted using empirical available data. Based on the numerical results conclusion and discutions are drawn.

Keywords: Transient heat transfer, carbon fiber-reinforced plastics, satellites, aluminum.

Introduction

One of the most important factors of spacecraft design is mass reduction. The launch cost is determined overwhelming by the mass factor, and in the same time the launch cost has a major relevance for the mission cost. For a Low Earth Orbit (LOW) satellite, launch cost is 5000 USD [1]. From the reduction in mass (and size) of a component, it can be presumed that a less supporting structure is needed. Therefore, even a comparatively small reduction in mass of one component of the satellite can result in a cascade reduction of mass and cost.

This study proposes to replace an existing electronic Advanced Data and Power Management System (ADPMS) housing for both in terms of box configuration and in terms of materials used. Box reference was implemented on PROBA2 satellite that was launched by the European Space Agency in November 2, 2009.

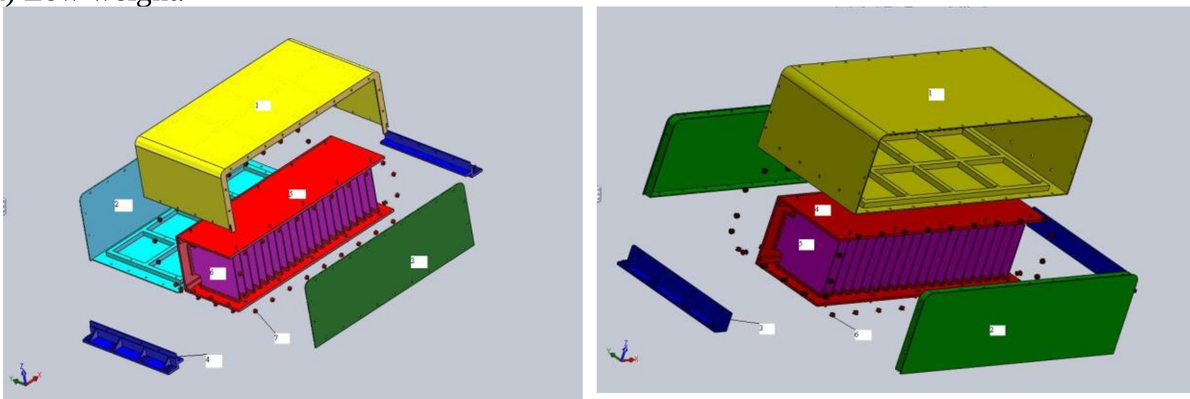
For a component to be viable in space, it must meet several conditions, the most important being [2,3]:

- Reaction to extreme temperatures;
- Protection against radiation;
- Vibration behavior;
- Structural strength.

The objective of this study is to validate the proposed geometry made of CFRP in terms of temperature variation behavior. For this, comparative studies between proposed and the existing geometry was made, using the finite element method.

It was considered a box made of fewer components generate a lot of advantages that are directly reflected in the total cost of the mission (see Fig 1)[4]. These are:

a) Fewer molds needed for the components;
b) Fewer connections;
c) Rigidity;
d) Low weight.



a. b.

Fig.1 a) Components for existing electronic housing on Proba 2 (1- hat section; 2- base and rear panel; 3- front panel; 4- mounting rails; 5- wedge locks; 6-electronic plates; 7- aluminum rivets)
b) Components for the CFRP proposed electronic housing (1- upper and lower panels; 2- lateral panel; 3- mounting rails; 4- wedge locks; 5-electronic plates; 6- aluminum rivets)

After the design of all components was accomplished the weight of each box was evaluated, and the results are in Table 1.

Table 1 Mass of subansambly after design

Model	***Mass [kg]***
Existing one	7.32
Proposed CFRP	6.01

Finite elements simulations

As shown in [4, 7-10] a classical equipment in a satellite is expected to be subject to a temperature variation between -10^{o}C and +40^{o}C. Hence, in the finite elements simulations the following assumptions were made:
a) There are not taken into account issues that arise at the interface between the contact box panels;
b) The existing box is made of aluminum alloy (see Table 2);
c) The proposed box is made of composite panel K1100/Wolfram/M46J (see Table 2). The 1.73 mm wall thickness is divided from the outside as follows: K1100 -0.815 mm; Wolfram - 0.1 mm; M46J - 0.815 mm (see Fig .2)
d) K1100 and M46J composite materials were considered **homogeneous and orthotropic**;
e) Electronic components, including outlets that are installed are composite (FR4, see Table 2)[6];
f) Satellite box’s side supports have all degrees of freedom suppressed on their underside.
g) Both boxes are subject to a +40^{o}C/-10^{o}C/+40^{o}C temperature variation.

Table 2 Materials used in finite elements analysis of composite boxes

Material	E_x [MPa]	E_y [MPa]	υ_{xy}	υ_{yz}	ρ [kg/m^3]	k [W/mK]
K1100	931000	70000	0.85	0.3	2200	1000
M46J	265000	71000	0.87	0.3	1590	84
Aluminium alloy	71000	-	0.33	-	2770	205
Glass/epoxy FR4	24000	21000	0.136	0.118	1850	0.81
Wolfram	411000	-	0.28	-	19250	173

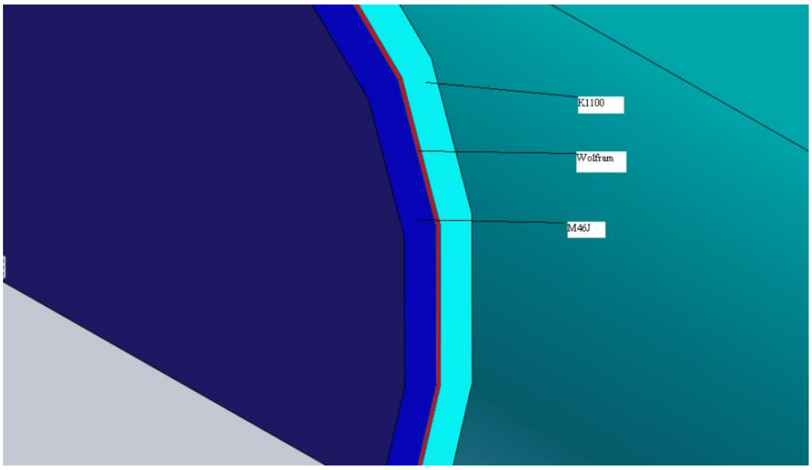


Fig. 2 The arrangement of material layers in composite panel

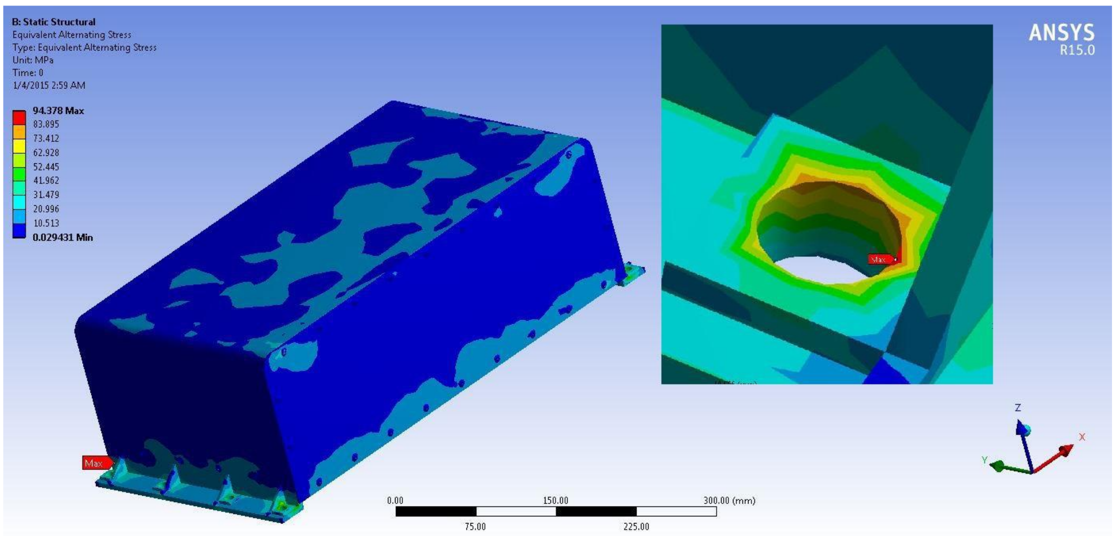


Fig.3 Stresses obtained for existing aluminum box- equivalent stress von Mises [MPa]

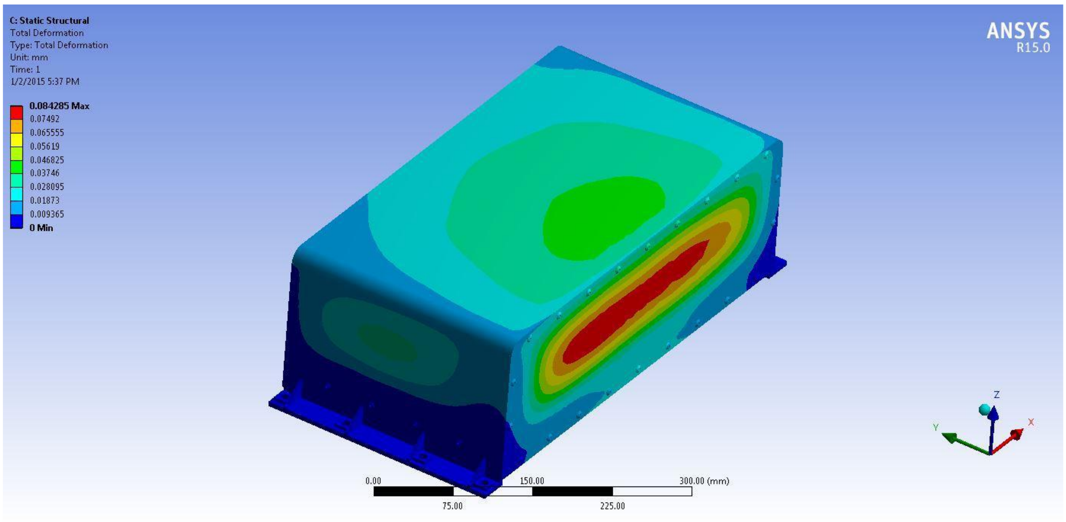


Fig.4 Displacements obtained for existing aluminum box [mm]

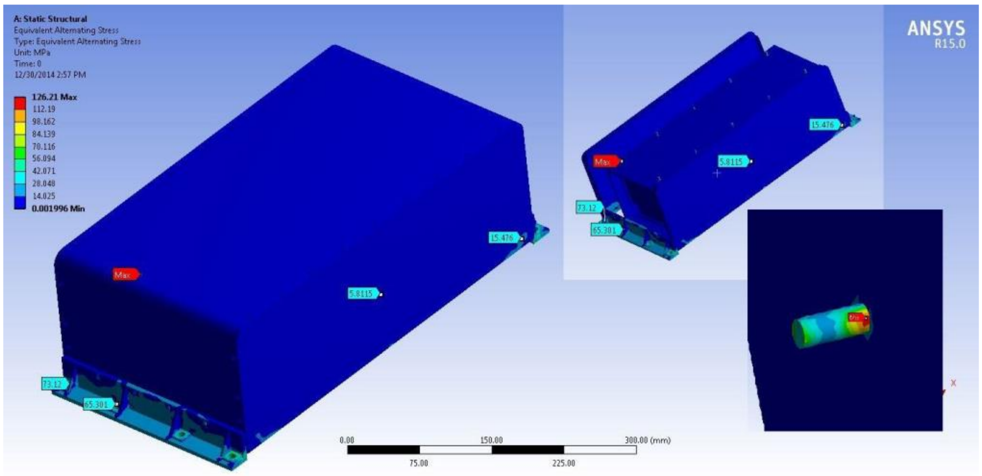


Fig.5 Stresses obtained for proposed CFRP electronic housing- equivalent stress von Mises [MPa]

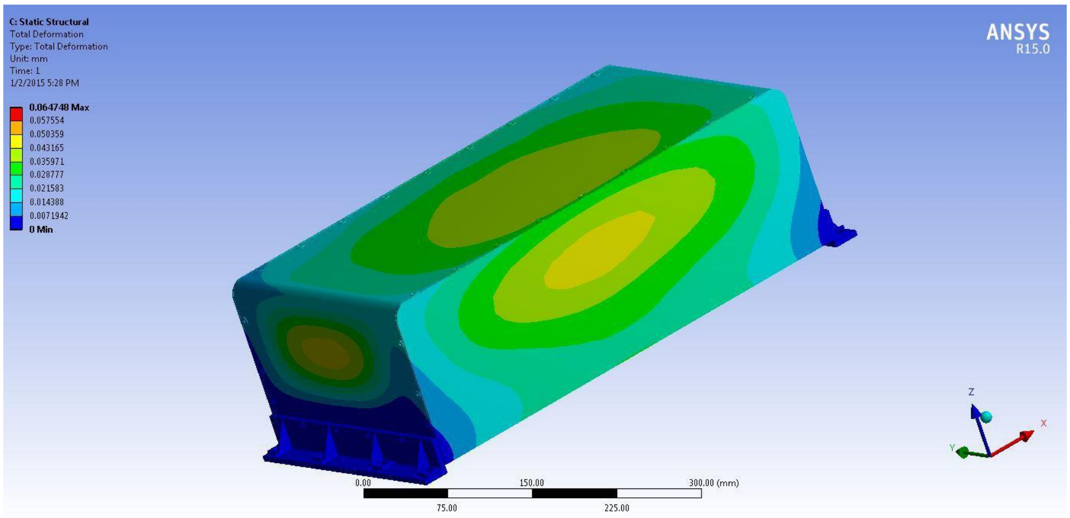


Fig.6 Displacements obtained for proposed CFRP electronic housing [mm]

Conclusions

As it was to be expected the aluminum housing withstands the harsh conditions of space. The maximum values of stresses and strains for aluminum electronic housing are far below those imposed by the manufacturer. Although in the case of composite box tensions grow by 34% its integrity is not jeopardized. Regarding deformations are insignificant differences between the studied models and obtained values not threaten structural integrity [3].

After modeling it was concluded that the proposed model of composite electronic housing reaches a weight of 6.015 kg this value represents about 80% of aluminum's box weight, so a significant reduction in mass is possible.

Further weight reductions are possible, giving up the riveting panels and introducing the seams between the common elements. It can also reduce the wall thickness to a minimum until they reach the maximum stresses and strains imposed.

It also notes that the proposed solution successfully meets all the requirements for space in terms of thermal variation stress.

Of course, to fully validate the model for space applications are required comprehensive analyzes. Furthermore, all analyzes should be coupled with experimental validation of the proposed model.

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