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Thermal regimes of space composite structures. Part I

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Abstract. Composite materials have a unique combination of mechanical and thermal characteristics. This allows you to create lightweight, durable, heat-resistant design. In the article the common features for all space composite structures were highlighted. The article deals with the thermophysical problems arising in the design of space composite structures associated with strict requirements for their weight and functional efficiency. The feasibility of using combined heat transfer models to improve design accuracy has been shown.

1 Composition materials for aerospace applications

1.1 Brief history of composite materials

Composite materials (CM) consist of two or more components with a definite interface [1]. Industrial production of CM dates back to the late 1930s. The aviation industry has been one of the main consumers of CM ever since. Starting from the mid-1950s, CM technology has been developing in rocket and space technology parallel to and independent of the aviation industry. This "branching" was caused by fundamentally different operating requirements for the thermal regimes of aircraft, ballistic missiles and spacecraft. In rocket and space technology, the operating level of thermal loads, temperatures, heating / cooling rates and temperature gradients is fundamentally different from that of aviation. A reentry vehicle is typically subjected to a heat flux with $50 \cdot 10^5$ W/m^2 density its heat shield withstands temperature gradients exceeding 1000 degrees/mm. An orbiter is exposed to the temperature changes ranging from plus 150°C on the illuminated portion of the orbit to minus 150°C on the shadow portion all of these is still out-of-limit for the conventional aircraft [2]. It should be noted, however, that the development of hypersonic aircraft in the near future may partially alleviate this difference.

First-generation CM were based on a polymer matrix: impregnated timber, glass fiber reinforced plastics (GFRP) and asbestos-filled plastics. Impregnated timber, a combination of birch timber lamina and phenolic resin, was used in power structures of military aircraft, in air rudders of R-7 missiles [3, 4]. There were attempts to use it for liquid fuel tanks for of ballistic missiles [5]. GFRP comprising glass fabrics, tapes or yarns and epoxy or phenolic resin was used in radio-transparent fairings for airplanes and missiles, solid rocket motor bodies, and launch pod containers [6, 7]. Asbestos-reinforced plastic was used in the first

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ablative thermal protection for the warheads of ballistic missiles and space vehicles that helped overcome the “thermal barrier” [8, 9].

The second-generation of CM includes carbon fiber reinforced plastics (CFRP) and organic fiber reinforced plastics (OFRP). CFRP proved an excellent material for any power structure that required high strength, stiffness and linear dimensions stability: wing skins and body parts, antenna reflectors etc. [10, 11]. OFRP took time to be introduced into aerospace technology and were used for solid rocket motor bodies and meteoroid protection screens [12].

Third-generation CM are metal composites and carbon or ceramic matrix composites. Metal matrix composites with boron fibers reinforcement were employed in a number of load-bearing structures, such as trusses, frames and landing gear [13]. The nose cones and leading edges in the Space Shuttle and Buran were made of carbon-matrix and carbon fiber reinforced CM. Additional surface and in-depth oxidation protection guaranteed operation in 100 flight cycles: up to 1650ºC maximum surface temperatures for up to 20 minutes per flight [14, 15]. Current research in ceramic matrix and carbon/ceramic fiber composites has shown that these materials can work at temperatures exceeding 2000° C in high-speed erosion and chemically active gas flows [16, 17].

1.2. General approaches to spacecraft thermal design

Spacecraft thermal design is a multistage, iterative process [18]. When choosing design solutions that will enable the specified thermal regime for a new spacecraft, engineers must account for the design requirements, operating conditions, available material and production resources. It is customary to consult the knowledge data bases containing information on previous research, on characteristics of structural, thermal protection and thermal insulating materials, coatings, liquid and gaseous media, and aerodynamic information etc. These databases can contain information on mathematical models, thermal physics, aerodynamics, radio physics, optics, problem solving methods and software. Apart from computational and theoretical data, they contain information on composite materials manufacturing technologies, laboratory testing methods of materials and coatings, ground and flight testing methods of structures, test equipment, instruments and sensors.

Verification of mathematical models is embedded into the design process. This is a procedure for structural and /or parametric identification based on inverse heat transfer problems using experimental data. These inverse problems are stated in the extreme form, where the unknown quantities are determined as a result of minimization of the residual functional between theoretical and experimental data for the similar spatial points of temperature measurements in the entire experiment. Reliable mathematical models enable selection of optimal design solutions.

2 Earth orbit cleanup

2.1 Transformable space debris capture device

Space debris is becoming a major problem for the practical activity in the outer space. There are a number of ways to clean up the near-Earth space from debris [19, 20]. One of the devices for cleaning the near-Earth orbits can be debris capture vehicles. To manufacture these spacecraft, transformable load-bearing composite elements can be used in combination with multilayer hybrid fabrics to withstand collision with space debris elements.

To date there are no recycling methods for the expired space structures, so the only option is to dispose of them by deorbiting into the atmosphere or transferring them into graveyard orbits away from main operational orbits. In any case, this will require special capture devices: nets, harpoons or catchers that will play an important though intermediate role in the cleanup process.

One of the catcher options can be a transformable tetrahedral pyramid with 6.3, 5.0 and 5.0 m sides and 6.3x6.3 m^2 base. The open base is directed funnel-like towards the flight, capturing the space debris elements (Fig. 1). Four telescopic rods (6 m total length, 200x100 mm^2 base cross-section) are used to deploy the catcher from the its stow position.

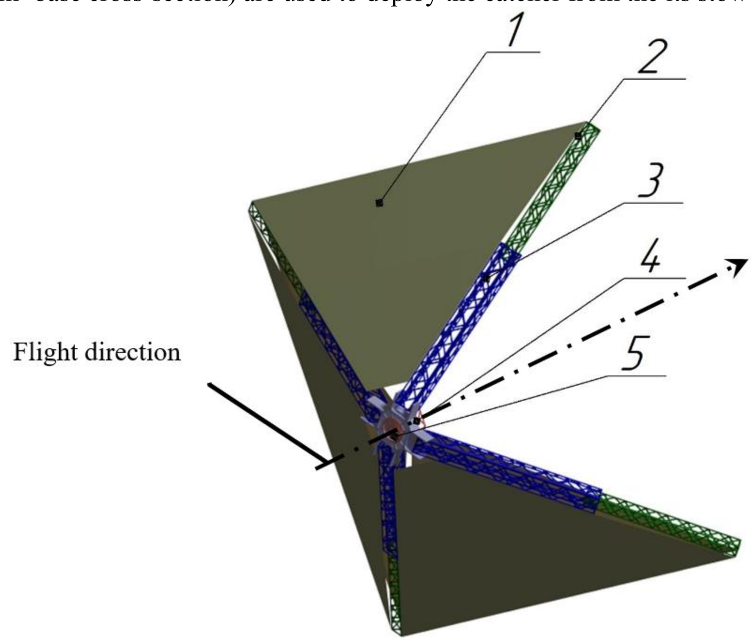

Fig. 1. Space debris catcher, one of the flaps not shown: 1 – debris catcher flap; 2, 3 – webbed load-bearing element; 4 – base; 5 – mount.

The walls of the catcher should be strong enough to withstand the impacts of space debris. The flaps are proposed to be 200 mm thick consisting of two separate screens. The top layer of each screen contains a layer of screen-vacuum thermal insulation (SVTI-I type), 20 mm thick, coated with metallized aramid fabric (product identification number 5397-92). The screen-vacuum thermal insulation consists of 20 μm thick polyimide film layers separated by a fiberglass scrim. The second layer of the screen serving as debris protection is a 4 mm thick fabric stack comprising 20 layers of CBM fabric (product identification number 8601), 190 g/m^2 linear density, plain weave, 15 warp threads by 15 weft threads, 58 tex. The back layer of the screen is another layer of screen-vacuum thermal insulation, 20 mm thick. The distance between the screens is 150 mm.

2.2 Modelling thermal state of a space debris catcher

When in the near-earth space, the catcher will be subjected to harmonical fluctuations in temperature. To understand the level of the walls' stability, the thermal regime of the trap must be determined with and without impact.
The aim of the work is to investigate the temperature state of multilayer fabric stacks when exposed to the heat radiation from the Sun and Earth.

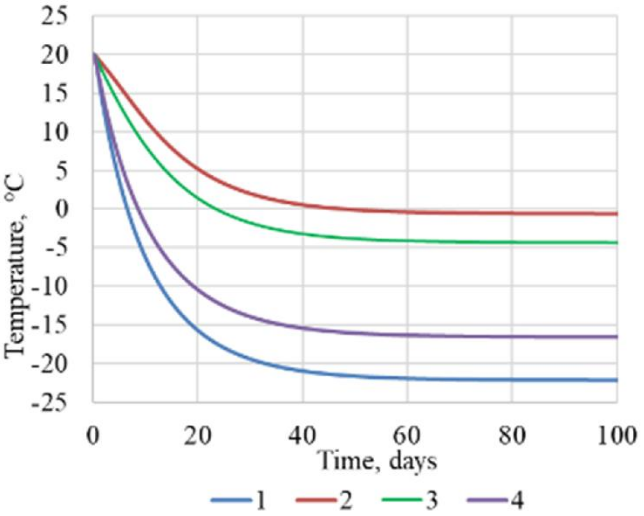

Fig. 2. Temperature of debris protection aramid fabric: 1 – first layer in the upper (relative to Earth) part of the catcher; 2 – second layer in the upper part; 3 – first fabric layer in the lower (relative to Earth) part of the catcher; 4 – second layer in the lower part.

The thermal regime modeling did account for the screen-vacuum insulation compacting under load or the elastic shell deployment elements potentially acting as "thermal bridges". Both the screen-vacuum insulation and debris protection layers were considered as a continuous material free from internal microstructure. The thermal physical characteristics of the materials were assumed to be independent of temperature.

With regard to the thermal loads, it was assumed that the direction of the catcher geostationary flight coincided with its symmetry axis, and the transverse axis was directed at the Earth. Under these conditions the catcher will be unevenly illuminated by the Sun, which will cause significant temperature changes along its surface. It was established that the outer surface temperature can vary from minus 150°C to plus 120°C, which is outside the recommended temperature range for the aramid fabric. However, the double-sided screen-vacuum thermal insulation makes it possible to reduce the temperature fluctuations to the acceptable values (Fig. 2).

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