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SUBSTANTIATION OF THE PROTECTION SYSTEM'S TECHNICAL OUTLINE FOR THE AEROSPACE OBJECTS

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Summary. *During their operational time, spacecraft are exposed to numerous factors, which are specific to the space environment. Spacecraft designing is a complex scientific and technical problem, which solution necessarily requires taking into account the possible effect of these factors on the structural elements and systems of the vehicle, including protective means in its concept and ensuring its functioning in the expected operational conditions. This paper presents a review of the main space environment factors, which affects the spacecraft, defines global trends in the protection systems' development and substantiation of the perspective protection system's technical configuration.*

Key words: *Environment factors, thermal flow, load, protection system, insulation, isolation.*

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1. Problem statement. The relevance of the issue of substantiating the choice of spacecraft's components and equipping the vehicle with protection systems is determined by the conditions and missions of the flight. Operational conditions of the spacecraft are characterized by the impact of the specific factors of space environment (FSE), which largely determine the requirements for the spacecraft's structural characteristics. Spacecraft's life cycle includes, in addition to the stage of direct work in the space environment, also stages of construction, preparation for launch and the launch itself, which definitely have their own characteristics and factors that can affect the design of the vehicle. But it is the stage of work in the space environment that is determined by conditions radically different from conditions on Earth, which require deep research and special attention during the process of designing the spacecraft, its systems and units.

The technical outline of the spacecraft and its systems, in addition to load conditions, is greatly influenced by a number of conflicting requirements (Figure 1). These requirements may be approximately divided into general, typical for most aerospace vehicles, and special, associated with distinctions of the spacecraft's functioning.

Finding a successful solution for these scientific, technical and technological problems involves the analysis and consideration of the available results of FSE research, modern experience in the aerospace engineering field, as well as performing a new research in the direction of finding a rational architecture of the spacecraft protection systems' technical configuration, modeling their work, substantiation of the best options.

2. Analysis of the known researches' results. Experience of thermal design and research of modern methods of spacecraft's thermal protection is shown in studies [1–6, 17].

Experience in studying of space radiation impact and radiation protection is presented in sources [7–14, 17]. Studies [11, 14] also adverts to radiation impact on human's health, which is extremely important in the context of manned spacecrafts' designing.

Designing of micro-meteoroids and orbital debris (MMOD) protection for the spacecraft is discussed in details in the study [15]. Study [16] also analyses risk of high-speed MMOD impact for the near-Earth manned and unmanned spacecraft. MMOD impact risk assessments, which is necessary for ensuring protection of the spacecraft's systems and its crew, were developed, planned and performed for the difficult missions beyond low near-Earth orbit and in the interplanetary space. This study also presents results of development of the ESABASE2/Debris software, ESA standard, as a tool for the MMOD impact analysis.

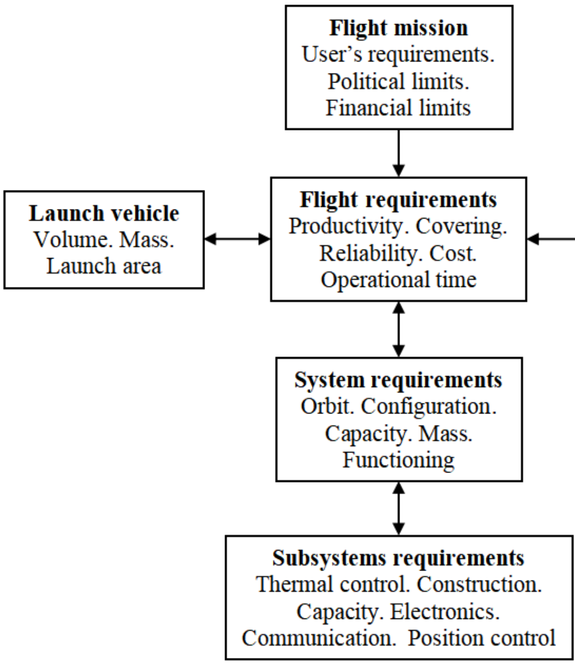


Figure 1. Scheme of the requirements for the spacecraft

3. Goal of this article is to inform about the results of the analysis of main FSE, which influence the spacecraft's construction and systems, modern global trends of the development of FSE protection methods for the spacecraft, substantiation of a rational architecture and the main requirements for the aerospace systems' protection. Specified technical outline can be seen both in modernization and in the development of new designs for perspective aerospace vehicles.

4. Research results

All of the FSE may be approximately divided into following groups:

I) specific thermal conditions – temperature range of ±150°C in the space environment and over 1600°C during the atmospheric reentry;

II) high vacuum impact – pressure value in the space is near $p = 10^{-8}...10^{-12}$ Pa;

III) electromagnetic and corpuscular radiation from the Sun (corpuscular, ultraviolet and X-ray radiation), from the Earth (Van Allen radiation belts) and from the outer space (cosmic rays);

IV) micro-meteoroid and orbital debris impact.

FSE may cause a negative impact to the durability and operating capabilities of the spacecraft's systems and structural elements, so providing their durability under space environment conditions is necessary for the effective long-time operation.

In the spacecraft's designing it is necessary to take into account the possible effects of these factors on its systems and elements, to ensure, in particular: spacecraft's appropriate thermal design, maintenance of the thermal balance during operation; spacecraft's proper protection from the MMOD impact; spacecraft's structural materials' resistance to the high vacuum influence, electromagnetic and corpuscular radiation, and other specific conditions of the space environment.

4.1. Thermal protection systems. Thermal operational conditions of the spacecraft are characterized by the heat exchange only in the form of thermal radiation, and are defined by the following heat flows: 1) direct solar radiation; 2) solar radiation reflected from Earth (albedo radiation); 3) thermal energy radiated from Earth (planetary radiation); 4) thermal radiation from the spacecraft to deep space.

Also, aerodynamic heating is essential during the atmospheric reentry of a reusable spacecraft. Spacecraft's external surface temperature in the space is between ±150°C, and it may outreaches 1600°C during the atmospheric reentry. Operational temperature range of spacecraft's elements is the following: between ±100°C for the majority of constructional elements; between ±50°C for electric drives and radio equipment; internal temperature of electronic devices – below 70°C; temperature of crucial elements – 20±1°C.

Thus, to preserve the functionality of spacecraft's systems, it is necessary to maintain the thermal balance – the balance between the heat flows entering and leaving the spacecraft.

Thermal balance equation may be expressed as:

$$\sum m_i c_i dT_i = (Q_{ext} + Q_{int})\, dt - Q_{rad} dt \text{ [W]},$$

where m_i, c_i, T_i are known as, respectively, mass (kg), specific heat capacity (J/kg·K) and temperature (K) of i-th constructional element of the spacecraft; Q_{ext} is the sum of external thermal flows (W); Q_{int} is the internal thermal flow of the spacecraft (W); Q_{rad} is the radiation from the spacecraft to deep space (W).

The sum of external thermal flows Q_{ext} may be expressed as:

$$Q_{ext} = A_s(Q_{sol} + Q_{al}) + \varepsilon_w Q_{pl} + Q_{mol} + Q_{rec} \text{ [W]},$$

where Q_{sol}, Q_{al} are the flows of direct solar radiation and albedo radiation, respectively (W); Q_{pl} is the planetary radiation (W); Q_{mol} is the thermal emission caused by molecular impacts with the spacecraft (W); Q_{rec} is the possible thermal emission caused by atomic recombination on the radiator surface (W); A_s is the coefficient of solar radiation absorption by the external surface of the spacecraft; ε_w is the blackness degree of the spacecraft's external surface.

If the specific thermal flows from external sources q_i (W/m^2) are known, then respective element Q_i of the external thermal flows' sum depends on spacecraft's midsection area directional to the respective thermal flow S_{Mi} (m^2) as:

$$Q_i = q_i S_{Mi} \text{ [W]}.$$

Internal thermal flow of the spacecraft Q_{int} depends on its' elements and systems operational program, their amount and the energy consumption capacity N (W). For the calculations, the maximum average heat flow $Q_{int\,max}$ from n spacecraft's devices may be expressed as:

$$Q_{int\,max} = \sum_{i=1}^{n} N_i \text{ [W]}.$$

If the external surfaces of the spacecraft are covered with a heat-insulating coating, it may be assumed that heat removal occurs only through the panels of the thermal regulation system (Figure 2).

In that case, thermal flow from the spacecraft to deep space Q_{rad} may be expressed as:

$$Q_{rad} = Q_{int\,max} + Q_{ext} \text{ [W]}.$$

To keep the temperature of spacecraft and its elements within the operational range, the active and passive thermal protection systems are being used.

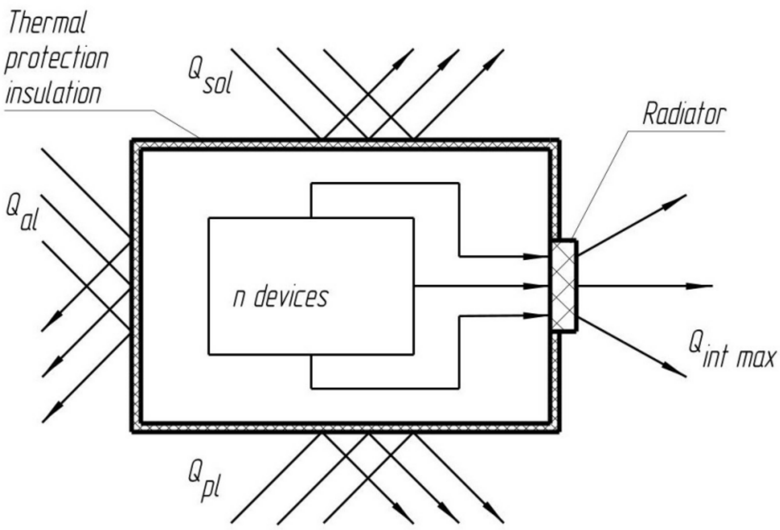


Figure 2. Model of the thermal flows and thermal protection systems of the spacecraft

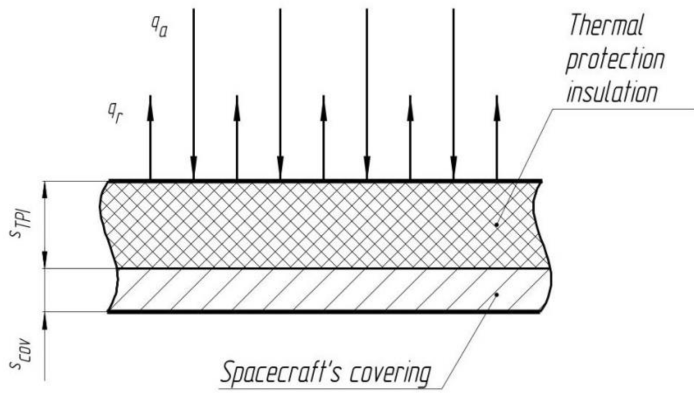
4.1.1. Passive thermal protection systems. Passive thermal protection systems are designed to provide more controllable heat input and output for the spacecraft. These systems include:

1) multi-layer insulation (Figure 3) – construction of heat-reflecting thin sheets, which mostly don't have their own thermal emission, and interlayers between them;

2) thermal protection coatings, which are designed differently to work properly under different thermal conditions;

3) ablative thermal shields, which are being used to protect the reusable spacecraft from the aerodynamic heating during the atmospheric reentry.

a)



b)

Figure 3. Photo of the multi-layer insulation (a)
and calculation scheme of the thermal protection insulation (b)
Source (a): John Rossie of Aerospace Educational Development Program (AEDP), CC BY-SA 2.5 license

Different types of thermal protection insulation are installed on the different external areas of the spacecraft depending on the temperature that this element experiences during the space flight and atmospheric reentry.

The principle of operation of an ablative heat shields is that the outer layer of the insulation takes the thermal load and is blown away by the gas flow. That creates cooler boundary layer around the surface and reduces heat transfer directly to the spacecraft. The ablative coating may be thick enough to work for heat absorption and emission without damaging, which make it partly reusable.

Based on the research, the main requirements of heat-resistant materials for perspective aerospace vehicles have been formulated, which are the following:

1) high heat resistance;
2) the ability to dissipate a large part of the convective heat flow;
3) minimal thermal conductivity coefficient;
4) minimal thermal expansion coefficient;
5) minimal weight;
6) resistance to external climate influence.

4.1.2. Active thermal protection systems. Active thermal protection systems, or thermal regulation systems, provide forced heat exchange between the spacecraft and its environment through emission of excessive thermal energy. These systems contains:

1) thermal plates with heat-separating devices;
2) heat pipes (Figure 4), which act as heat transfer elements;
3) surface electric heaters for maintaining devices' temperature;
4) heating radiators, which transfers thermal energy from heat transfer elements and also maintaining devices' temperature;
5) cooling radiators, which is necessary for heat dissipation into the outer space — they are being installed on the part of the device that will receive the least amount of sunlight during operation, with the normal direction into the open space.

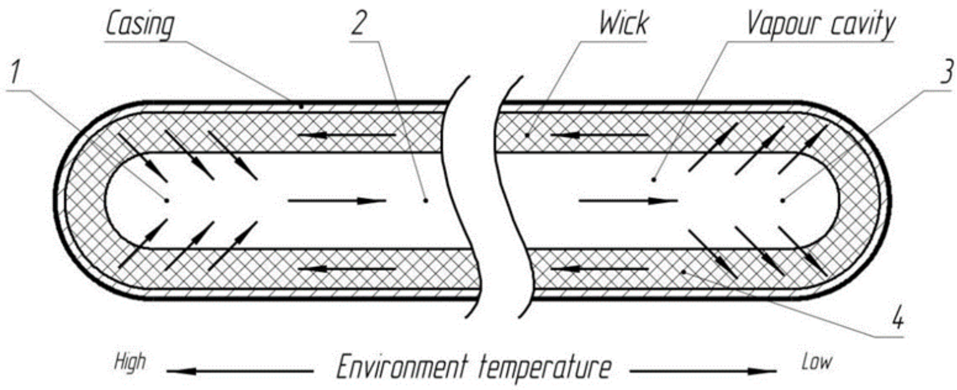


Figure 4. Work scheme of the heat transfer element – heat pipe:
1) evaporating of the working fluid, 2) migrating of the vapour to lower temperature end, 3) condensing of the vapour with releasing of thermal energy, 4) flowing of the working fluid to the higher temperature end

In order to reduce mass of the spacecraft $m \rightarrow min$ its devices and aggregates may be installed on multifunctional body panels supported by heat pipes. These panels take external loads and loads from the devices, provide heat transfer from devices and heat dissipation into the outer space, and also may work as vibration dampers and anti-meteoroid screens.

In addition to the passive and active thermal protection systems, other thermal design methods are being used, such as optimal layout of the spacecraft's elements depending on the expected heat distribution and selecting materials with proper absorption coefficients and blackness degrees in the configuration which provides the most appropriate estimated thermal balance for the spacecraft.

In general, solutions for the spacecraft's thermal design must be connected with other design solutions for that spacecraft. Therefore, the solution to the problem of ensuring the spacecraft's systems thermal operational conditions has a significant impact on the choice of other constructive solutions during its overall design.

4.2. Radiation protection systems. During the spacecraft's operational time its components require protection from electromagnetic and corpuscular radiation, which is divided into:

1) corpuscular solar radiation;
2) electromagnetic solar radiation (ultraviolet and X-ray gamma radiation);
3) radiation of the Earth's Van Allen belts (internal – proton radiation, external – electron radiation);
4) cosmic rays.

The background radiation in space isn't homogeneous. Potential radiation exposure during the spacecraft's operational time depends on the trajectory of the spacecraft, its position relative to Van Allen belts in particular, time of its operation and solar activity during that period. In general, flux of solar radiation energy, depending on distance between the Sun and the spacecraft, may be expressed as:

$$q(r) = \frac{q_0}{r^2} \left[\frac{\mathrm{W}}{\mathrm{cm}^2}\right],$$

where q_0 = 0,14 W/cm^2 is the solar constant, equals the radiation flux received on the distance of one astronomical unit from the Sun; r – distance to the Sun in astronomical units.

In the spacecraft's structural materials and systems, radiation mostly impacts their optical properties (transparency of the optical systems' elements), mechanical properties of plastics and elastic materials, electrical properties of conductive materials.

In order to protect the spacecraft and its crew from radiation influence, the following methods are being used:

1) passive radiation protection – using the spacecraft's covering materials, which stop the penetration of radiation, and protective screens for radiation-sensitive optical elements;
2) active radiation protection – dissipation of radiation using electromagnetic fields;
3) medical reduce of the negative impact on the crew's health;
4) reduce of the radiation damage on the spacecraft by optimization of its trajectory and technical characteristics to accomplish the flight mission faster.

Passive radiation protection materials are characterized by the ability to disperse directed radiation and the stopping power F of the material, which may be expressed as:

$$F = \frac{dE}{dx}\ [\mathrm{MeV/cm}]; \quad F \sim N Z_T; \quad \frac{F}{\rho} \sim \frac{Z_T}{A_T},$$

where E is the particle energy (MeV); x is depth of the material penetration by the particle (cm); N is the number of molecules per unit volume; Z_T is the atomic number of the material; ρ is the density of the material (g/cm^2); A_T is the atomic mass of the material.

The following concepts exist in the field of active radiation protection:

- electrostatic shielding – use of a powerful electric field to reflect radiation particles;
- plasma shielding – use of a magnetic field to catch charged particles, which creates a plasma that induces an electric field to reflects incoming radiation;
- magnetic shielding – use of a powerful magnetic field to reflect radiation particles.

4.3. MMOD protection systems. Micro-meteoroids and orbital debris may cause an erosion on the spacecraft's covering and its devices, optical elements in particular.

Damage of the spacecraft's structural elements (an example is shown on Figure 5, [15]), caused by MMOD, values by crater radius as:

$$r = K(mV)^{1/3} \text{ [m]},$$

where *r* is the crater radius after MMOD impact (m); *m* is the mass of the micro-meteoroid (g); *V* is the micro-meteoroid's movement velocity (m/s); *K* is the coefficient depending on the micro-meteoroid and covering material, which may be approximated for calculations as *K = 1*.

Spacecraft's covering endures the MMOD collision, if its thickness exceeds micro-meteoroid's radius *r* by 1,5…2 times, but reinforcing the covering enough to protect it from MMOD impact causes unreasonable weight increase.

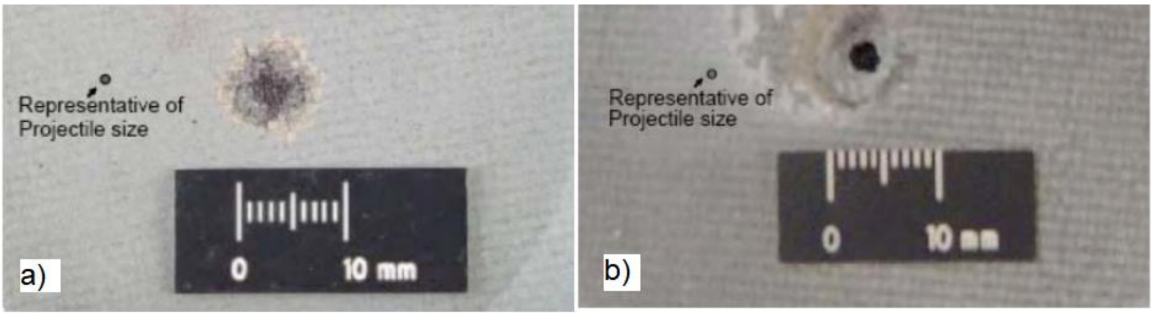


Figure 5. Spacecraft's structural elements impact damage due to 0,8 mm aluminum projectile at 7,1 km/s (normal impact): a) damage before arc-jet test; b) damage after arc-jet test simulating reentry conditions indicating burn-through

Instead, the following alternatives are possible:

1) using protective anti-meteoroids screens to catch MMOD and slow them down or break them;

2) facing the vulnerable equipment opposite to spacecraft's flight direction to reduce risks of the MMOD impact;

3) doubling spacecraft's equipment and systems, which may be damaged by MMOD, to provide stable operation even in case of the main system disabling.

In addition, for the perspective spacecrafts is advisable to take into account the micro-meteoroid impact detection system (Figure 6) [15].

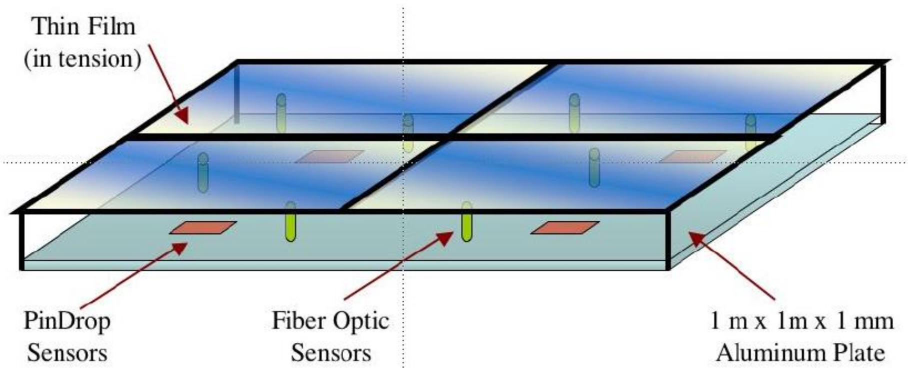


Figure 6. Micro-meteoroid impact detection system

4.4. High vacuum impact protection. For the space environment, which is highly rarefied, typical pressure value is near $p = 10^{-8}...10^{-12}$ Pa. High vacuum is defined as the environment with the pressure below $p = 10^{-6}$ Pa. Such environment is characterized by the mean free path comparable with the spacecraft's linear dimensions, and low return coefficient Z, which is defined as ratio of the number of particles returning to the surface after evaporating to the general number of evaporated particles. Accordingly, the main factor of the high vacuum impact on the spacecraft's construction consists in the increased mass loss, caused by evaporation, sublimation and desorption of structural materials, insulations and lubricants. This effect leads to the surface evaporation of the spacecraft's materials.

Intensity of the surface evaporation depends on temperature and physical properties of the materials (Table 1) [17].

Speed of surface evaporation S is measured in thickness units per time units and may be expressed as:

$$S = 0{,}2466 \cdot 10^{6} \frac{p}{\rho} \sqrt{\frac{\mu}{T}} \ \text{[cm/year]},$$

where p is the vapour pressure of the surface material (Pa); ρ is the density of the surface material (kg/m^3); μ is the molecular weight of the material's gas phase; T is the environment temperature (K).

Table 1

Speed of the surface evaporation

Material (chemical element)	Speed of the surface evaporation at the specified temperature (in °C)		
	0,1 μm/year	10 μm/year	1 mm/year
Cd	38	77	122
Zn	71	127	177
Mg	110	171	233
Au	660	800	950
Ti	920	1070	1250
Mo	1380	1630	1900
W	1870	2150	2480

Calculated evaluation of the mass loss Δm_i caused by high vacuum impact is only approximate, because intensity and specifics of the surface evaporation in the real conditions may change depending on various factors (such as cleanness and quality of the surface material, surface voltage).

In general, speed of surface evaporation for metal amounts to $S = 10^{-4}\ldots10^{-3}$ cm/year.

If thickness of surface evaporation reaches $\Delta\delta = 10^{-4}$ mm, the material changes its optical properties. Evaporation of about 0.5...1 mm affects the durability of the construction and may cause damage to the spacecraft's sections hermetic state. Surface evaporation is mostly actual for materials of the spacecraft's elements, which have high operational temperature and are in direct contact with the external environment.

Materials such as molybdenum, graphite and tungsten are the most resistant to surface evaporation, when cadmium and zinc are the least resistant. High vacuum impact for the resistant materials manifests mostly in the surface cleanness change, which may cause breach of the proper thermal conditions. In order to reduce the evaporation, external surfaces are coated with protective films (oxide coatings, phosphate coatings).

For the plastic materials speed of the surface evaporation is measured as a percentage of the mass per unit of time, since the evaporation of plastics, unlike metals, occurs in all of the material's volume. Acceptable mass loss for plastics should not exceed 1...2% per year. Evaporation deteriorates their durability properties, changes the value of electrical resistance and optical properties. In general, plastic surfaces, which is supposed to contact with the rarefied environment, are coated with the layer of metal, because plastics are noticeably less resistant to the high vacuum impact.

Vacuum evaporation also occurs with surface contamination, which results in the spacecraft's external surfaces becoming cleaner. At the same time in the space environment, due to its low oxidation potential, the formation of protective oxide films on the surfaces is difficult. This leads to increased adhesion and cohesion of surfaces, which may cause blockage and «cold welding» in contact pairs.

Due to the fact that heat transfer in a vacuum is practically possible only by radiation, heat exchange is difficult even between surfaces adjacent to each other – this is caused by small vacuum gaps on the contact surfaces because of their roughness. Vacuum also leads to the development of undesirable electrophysical phenomena (electrical discharges, breakdowns, leakage currents).

Based on research of the environmental factors' impact on the spacecraft and review of modern global trends [1–7], stages and procedures of the further research are stated. Further research involves the substantiation of the choice of solution options for the reusable spacecraft's protection systems, the choice of design and locations of the passive thermal protection system elements (thermal protection insulation of the spacecraft).

For the calculation substantiation of the choice of structural parameters for the spacecraft's perspective protection system, four main stages of research are proposed.

1. Definition of the calculated operating conditions of the spacecraft's main components (as the objects of protection), which provide their reusability. Analysis of load conditions of spacecraft's prototypes, parameters and characteristics of typical flights. Formulation of list of calculation load cases. Model of the operational (loading) process of the spacecraft's elements and systems during one cycle «launch» – «reentry» is approximately divided into four stages (Table 2), and risk of the negative impact of relevant factors during flight or waiting time.

In addition, accounting results of defect detection of aerospace systems' products after real operation, identifying damage mechanisms [16], is advisable.

Table 2

Stages of operation process of the spacecraft's elements and systems during one cycle «launch» – «reentry»

№	Stage	Factors, which influence the spacecraft
1	Orbital launch	Vibration, impact and linear loads, aerodynamic heating, acoustic noise
2	Orbital flight	Space vacuum, high and low (up to cryogenic) temperatures, electromagnetic radiation (from thermal to hard X-rays), corpuscular flows, MMOD impact, weightlessness
3	Atmospheric reentry and landing	Overloads, impacts and incraesed temperatures aerodynamic heating)
4	Storage and transportation	Biological factors, moisture, temperature, sea water, sea fog, vibrations, impacts, linear accelerations

2. Based on analysis of prototypes' design and their operational conditions, performing of a finite element analysis for models of the passive thermal protection system components and objects of protection. Choosing of physic-mechanical properties of the structural materials and geometrical parameters of the construction. Research of the possibilities to ensure the specifies thermal conditions, using models of spacecraft's protection systems elements, with methods of computer modeling. An example of calculation model for the spacecraft's thermal protection system element [5] is presented on Figure 7.

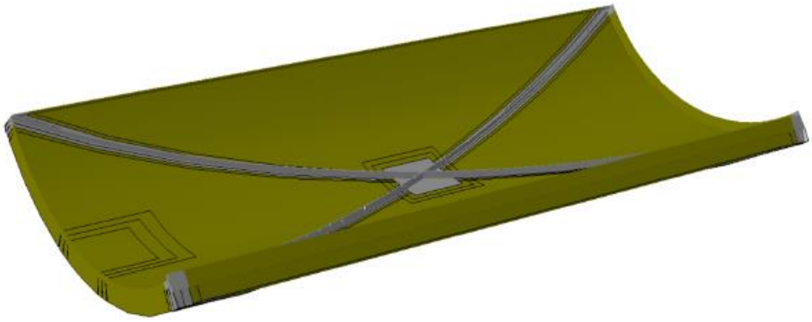


Figure 7. Calculation model of the protection system's element – composite sandwich structure carrying the ablative insulation

3. Studying the characteristics of the models of the spacecraft's protection systems' elements. Research variants of models of the passive thermal protection system for the spacecraft, their operating characteristics with the aim of determining the best variant, using computer modeling [19]. The best variant of the model should meet the requirements of providing the necessary thermal operational conditions, thermal field in structures $T_i(x, y, z, \tau) \rightarrow min$, required number of load cycles $N_i \rightarrow max$, minimal mass $m_i \rightarrow min$ and minimal damage $P_i \rightarrow min$ by various ways in the expected operating conditions during the flight time τ [20].

Proposed algorithm of the thermal conditions calculation for the elements of the spacecraft with ablative thermal protection insulation is shown on Figure 8.

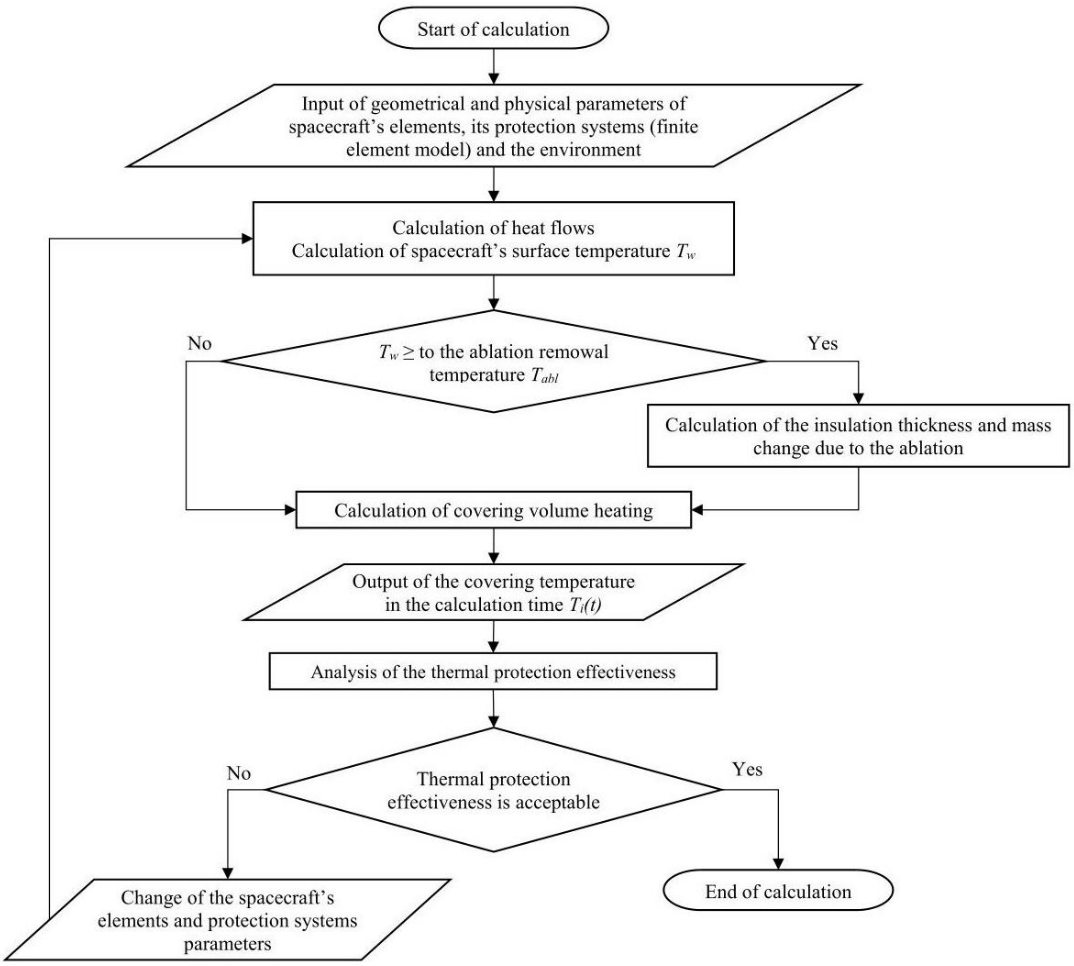


Figure 8. Calculation algorithm for the thermal protection system's elements

4. Definition of the most profitable options for the design parameters of spacecraft's protection systems, from the aspect of resource ensuring and reusability of the main components (objects of protection).

After analyzing the simulation results, choosing the best design solution for the passive thermal protection system of the reusable spacecraft is expected.

Presented stages are the main. If necessary, additional calculations of stable and unstable thermal states of the spacecraft's component are performed, taking into account influence of the different environmental factors, physical fields, heating or cooling on the mechanical condition and damage of the materials.

Conclusions. Based on FSE analysis, technical configuration and the main requirements for the spacecraft's protection systems, which may be used both in the modernization of the standard spacecrafts' protection methods and in the creation of modern high-tech protection systems. Equipping modern and perspective spacecrafts with high-tech protection systems is a key element of both ensuring a significant increase in the technical level from a general scientific point of view, and improving durability parameters and the possibility of safe reusability.

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ОБҐРУНТУВАННЯ ТЕХНІЧНОГО ОБРИСУ СИСТЕМ ЗАХИСТУ АЕРОКОСМІЧНИХ СИСТЕМ

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Резюме. *Космічні апарати під час експлуатації піддаються впливу чисельних факторів, специфічних для космічного середовища. Проєктування космічного апарату – складна науково-технічна задача, в ході розв'язування якої необхідно враховувати можливу руйнівну дію факторів навколишнього середовища на конструктивні елементи та системи апарату. В ході проєктування у концепцію апарату мають бути закладені засоби його захисту від дії середовищних факторів, а проєктування захисних систем повинно бути тісно ув'язаним із іншими конструктивними рішеннями для даного апарату з метою забезпечення його функціонування в очікуваних експлуатаційних умовах. Оглянуто основні фактори космічного середовища, що впливають на космічний апарат. Серед таких факторів виділено специфічні температурні умови, вплив глибокого вакууму, радіаційний вплив та дію мікрометеорних часток. Зазначаються найбільш суттєві ризики, зумовлені впливом даних факторів на конструкцію й системи космічного апарату. В якості таких ризиків визначено порушення теплового режиму космічного апарату, поверхневий знос конструктивних матеріалів та мастил, зміна властивостей матеріалів під дією радіаційного випромінювання та вплив на здоров'я екіпажу, ерозійний вплив мікрометеорних часток. Наведено основні формули для кількісного* *оцінювання впливу зазначених факторів космічного середовища. Визначено світові тенденції розвитку систем захисту космічних апаратів від зазначених факторів. Розглянуто різновиди пасивних та активних систем теплового захисту, радіаційного захисту, способи зниження впливу глибокого вакууму та ерозійної дії мікрометеорних часток. Обґрунтовано технічний обрис перспективних засобів захисту. Сформовано основні вимоги до засобів теплового захисту, запропоновано основні етапи досліджень систем захисту космічних апаратів, у контексті яких цикл функціонування багаторазового космічного апарату «пуск» – «повернення» поділяється на чотири стадії з певним характерним набором середовищних факторів. Запропоновано алгоритм розрахунку системи пасивного теплового захисту для космічного апарату багаторазового використання під час атмосферного спуску.*

Ключові слова: *фактори середовища, тепловий потік, навантаження, система захисту, покриття, ізоляція.*

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