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Advancement of Design Technologies for Small Remote Earth's Sensing Satellites

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Abstract: One of the key goals of satellite design is reduction of system's mass while maintaining high reliability (probability of faultless operation). Every technological decision made is based on the goal of the project, available production and technological base, material and time resourses and designer's experience. A large number of iterations of design work at enterprises could be often attributed to the established features (traditions) of the organization of work of designers and technologists. There is a demand for advancement of satellite design methods. Development of every satellite component should be carried out within a set of shared models of structural loading, thermal conditions assembly model, unified functional model and a unified model for the formulation and execution of design requirements. This approach to design is theoretically justified. Also, this approach is published in a more systematic way and is compared with other approaches.

Key words: Small satellite, remote sensing of the Earth, design, construction integration, complexing, method, reliability

INTRODUCTION

Small satellites are often used for flight testing of the newest technology. They are complex systems and have practically the same features as the full-size spacecraft.

One of the main research trends of satellite design is reduction of system's mass while maintaining high reliability (probability of faultless operation).

One of the approaches towards mass reduction is deep integration of onboard equipment. The concept of modular design is not very common amongst Russian space enterprises. Some methodical approaches have been developed, however, they are seldom used in practice.

There is a demand for development of methodical approaches and design techniques for small satellites that would take into account deep integration of on board equipment principles. This conclusion is obtained on the basis of the positive effect shown there from the application of such new approaches (Safronov *et al*. 2018).

Formulation of the integration problem: The integration problem is formulated as follows: let a certain range of external and external factors X be given. Every factor is attributed with a charachteristic vector $x_j = \{x_j1, x_j2,\ldots, x_je\}$. A strategy A must be defined, that would include the minimum number of elements (centers) of the solution set Y and ensuring the satellite's response to any external and internal factors from the set X while: satellite's onboard equipment p can realize every external factor $x \in X$. The number of acceptable reactions a_{ir^i} to any internal or external factor should be maximal but no less than two. The number of strategy's elements p should be maximal. The vector function of relative losses in the implementation of the strategy on the set X reaches its minimum value when min φ $(x_i, Y, a_{iq}{}^{i})$:

$$q^i = 1, \ldots, R_i$$
$$i = 1, \ldots, n$$

The algorithm for the problem of integration of on-board equipment can be reduced to determining the vector of the basic composition $\bar{p}$, finding the optimal set of permissible reactions for each external or internal factor for any state vector of the satellite and analyzing the vector function of the relative losses.

MATERIALS AND METHODS

Integration criteria: Based on the analysis of existing technical solutions, a high level of integration of onboard systems leads to:

- Decrease of satellite's weight
- Increase of systems reliability and reliability iof satellite as a unit

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Onboard equipment integration can influence:

- Mass of the satellite
- The stability of the installation sites of the sensing elements of the attitude control system
- Power consumption of the attitude control system (through reducing the main moments of inertia of the satellite)
- Energy consumption of the thermal control system (optimization of the temperature conditions of the onboard modules)
- Reliability of successful performance of the i-th onboard task P_i

Thus, the criteria for onboard equipmen integration are:

$$\begin{aligned} & m_{\text{МКА}} \rightarrow \min, \\ & N_{\text{nomp} \_\text{СуД}} \rightarrow \min, \\ & N_{\text{nomp}\_\text{COTP}} \rightarrow \min, \\ & P_i \rightarrow \max \end{aligned}$$

Satellite's mass model taking into account integration of onboard equipment: The first-approximation mass model of a non-integrated satellite equipment has the following form:

$$m_{se\ i} = k_{m\ se\ i} \cdot m_{ss}$$

Where:

$k_{m\ se\ I}$ = Mass coefficient of the i-th satellite equipment

m_{ss} = The average mass of a modern satellite with a close target vector or mass of the prototype

Ratio of the mass of functional components i-th system m_{fci} to the mass of its body m_{ki} for modern satellites is about:

$$1 \leq (k_k = m_{fc}/m_{bi}) \leq 1.92$$

This value is obtained by analyzing the materials of some papers (Fortescue *et al.* 2011; Gao *et al.* 2018). By Gao *et al.* (2018) presents some analytical information on the equipment of modern satellites. The results of the analysis of onboard equipment mass characteristics of observation satellites are presented in other study (Fortescue *et al.* 2011).

The weighted average is $k_{ê}$ = 1.24. Wherein the mass m_{ki} includes the body itself with fixtures, the elements of metallization. The m_{fci} includes the mass of electric device, printed circuit boards amplifiers, power supplies, sensors, connectors, wires, buses, wiring, etc.

The m_{fci} mass is a function of class of utilized electric devices K_{ued}, the degree of redundancy r, optimality of the electrical circuit (composition of elements, $\bar{D} = (D_1, D_2, ..., D_m)$, length of communication lines l) functional purpose of the device F_i and in general depends on the level of professionalism of the designer and financial and organizational limitations U_i:

$$M_{fci} = m_{fci}(K_{ued}, r, \bar{D}, l, F_i, U_i)$$

For more detailed calculations for the non-integrated equipment of the earth remote sensing satellite, another method is useful (Kirilin *et al.*, 2015) which is intended for solution of high dimension design problems can be used. Also, mathematical models being in use in aerospace industry are presented there.

Multi-function device weight m_{mfd}, obtained as a result of the integration of several devices in the first approximation can be approximately determined as the sum of the masses of its components $m_{fci\,|I = 1, ..., n}$, designed separately and its structure:

$$m_{mfd} = \sum_{i=1}^{n} m_{fci} \cdot (1+1.24) = 2.24 = \sum_{i=1}^{n} m_{fci}$$

The mass of the multifunctional device is updated based on the results of weighing the prototype manufactured, after acceptance, control and complex vibrodynamic, static, thermovacuum tests and electromagnetic compatibility tests.

The mass of the on-Board Cable Network (BCN) when assembling modules in packets is smaller than for unintegrated onboard equipmant. Weight of the BCN is a function of the layout vector of the i-th onboard system $\bar{H}_i = (H_{i1}, H_{i2}, ..., H_{ie})|_{e = 1, ..., w}$ and type of information exchange of the i-th interface $T_{\text{иО}i}$:

$$M_{bcn} = m_{bcn}(\bar{H}_i, T_{iei})$$

The degree of unification of interfaces, defined as the ratio of BCN's mass with the unified type of information exchange $m^{uie}{}_{bcn}$, to the overall mass of the BCN is expressed by the interface unification factor $k_{\text{уи}}$:

$$M_{bcn} = k_{ui} = m^{uie}_{bcn}/m_{bcn}$$

Digital model of a remote Earth's sensing satellite based on ontegrated onboard equipment: A satellite platform fitted with an optical-electric payload has been chosen as a prototype. Masses of the device, the payload and the structure of the satellite are 124.38, 27.77 and 23.5 kg, respectively.

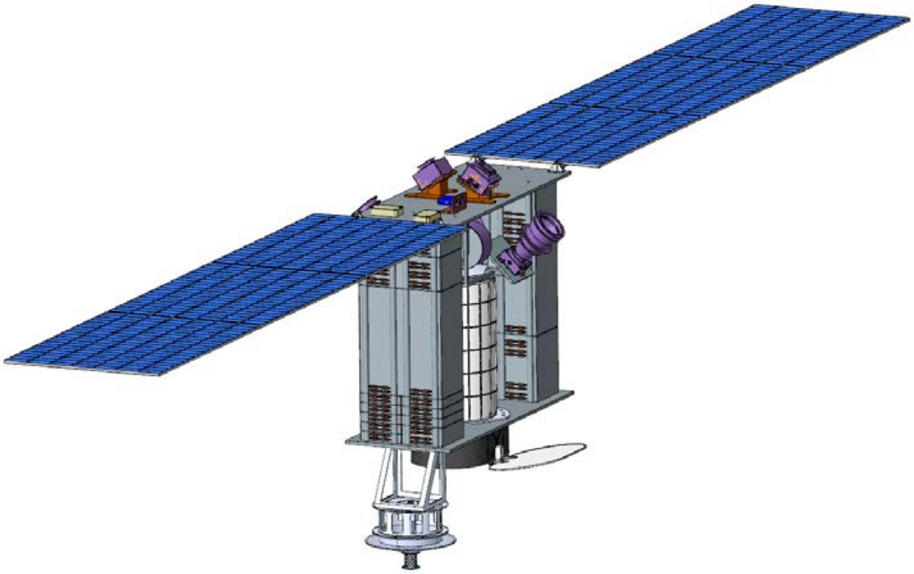

Fig. 1: General view of the designed satellite

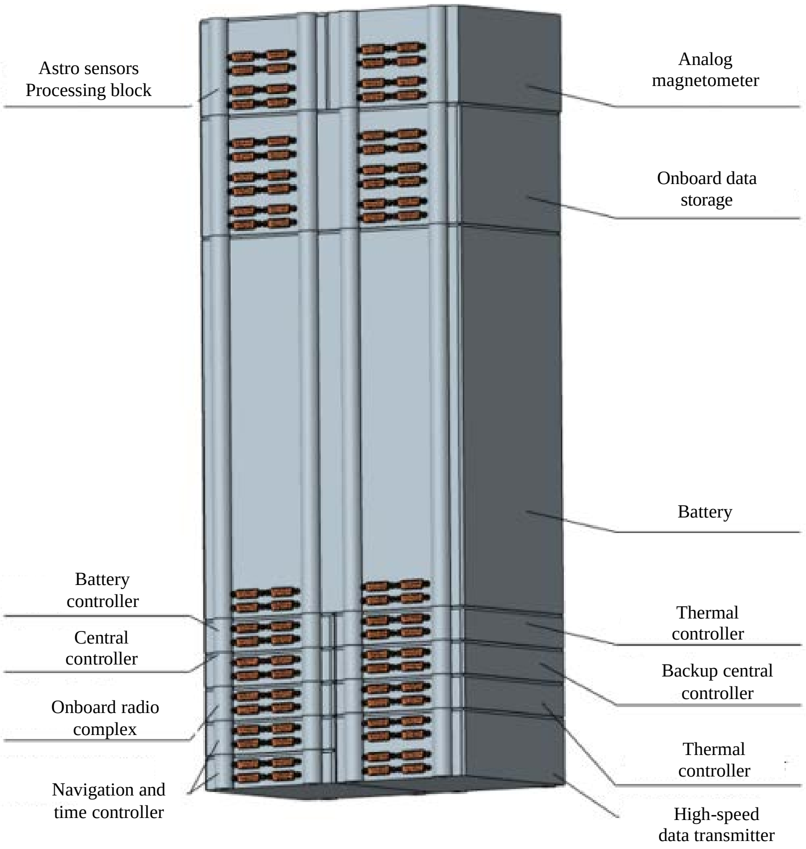

Fig. 2: A set of the onboard equipment

The satellite, designed with the use of the developed principles of integration has an open architecture, the bearing elements are the bodies of onboard equipment blocks and two horizontal honeycomb panels (Fig. 1). The satellite shares the same payload with the prototype standardized equipment modules are organized into two sets (Fig. 2) which are located on either side of the paylaod. The propulsion system is designed using the

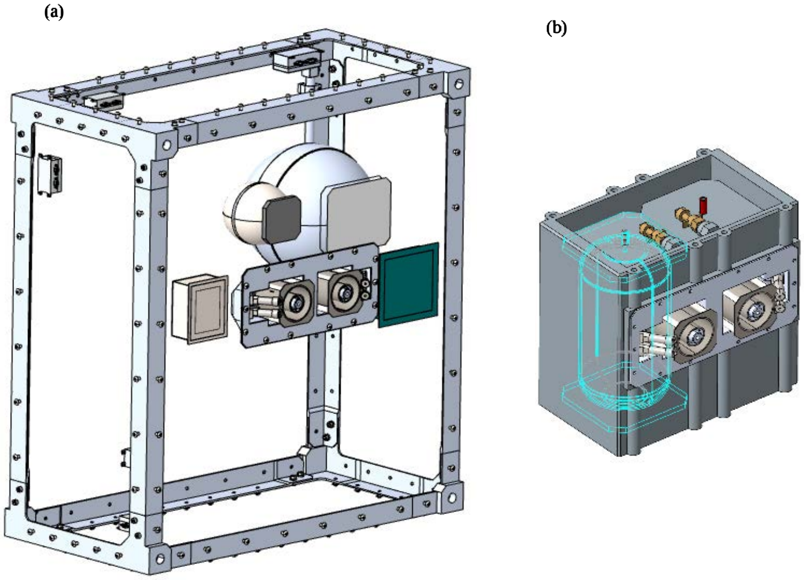

Fig. 3: a, b) Propulsion system assembly

results of the analysis (Krejci and Lozano, 2018) which describes the principles of operation of propulsion systems, taking into account the features of small size, weight, power plant and other operational limitations of small satellites. The propulsion system is made in the form of a standardized module. The weight of fuel and operation modes is selected based on simulation according to the methodology (Volotsuev *et al*., 2017). Proposed there the orbit correction algorithms assume step-by-step verification of the perturbation force by direct measurement and calculation of control parameters by approximate analytical relations. Such approach is quite applicable in this research. The xenon supply unit and the fuel storage tank were also integrated into the module design (Fig. 3). Unification of communication interfaces and protocols for information exchange between the systems was carried out without any increase of the module's weights.

The mass of the modernized satellite and its structure amounted to 115.2 and 16.5 kg, respectively. The average density of the layout increased from 347.2-597.5 kg/m^3. All on-board equipment (37 devices) was modeled anew except for ones that are not appropriate to integrate (6 instruments-two blocks for determining the coordinates of stars, four flywheel engines).

The implementation of the developed methods allowed to reduce the weight of the satellite's structure by 34, ..., 38% due to the inclusion of the onboard equipment into the load bearing scheme. At the same time by increasing the stability of the installation sites of attitude control system's sensitive elements, the accuracy of determining the target coordinates of objects when processing images can be increased by 15-25%.

Design method taking into account itegration of onboard equipment into the load-bearing scheme of the satellite or another equipment system.

The design method taking into account itegration of onboard equipment into the load-bearing scheme of the satellite or another equipment system contains the following tasks:

Obtaining the general project layout data from the main designing unit. Analysis of the layout quality of the satellite's modules with the selected geometric parameters.

Transfer of requirements on geometrical parameters of satellite's modules to the co-executing enterprises. Calculation of mass-inertial (preliminary geometry of the module, module mass, position of the center of mass, the main moments of inertia) and thermal (thermal contact area, heat flux density, power consumption, requirements for the installation of EVTI, distribution of heat capacity), specification of economic and time characteristics of the manufacture of the satellite's module by the co-executing enterprises and their transfer of the estimated data to the parent enterprise.

Formation of the design appearance of the satellite's design, taking into account the assigned parameters of its elements (numerical simulation). Analysis of the design appearance of the satellite's design and materials of the co-executing enterprises. Correction of geometric parameters of onboard equipment modules and their transfer to co-executing companies.

Creation of design documentation for the satellite, its structural elements and onboard equipment modules. Preparation of proposals on correction of the standards in the terms of reference due to the use of new design, construction and manufacturing methods.

The design procedure taking into account the unification of blocks and onboard equipment elements in accordance with accepted standards and their inclusion in the satellite's assembly layout: The design procedure taking into account the unification of blocks and onboard equipment elements in accordance with accepted standards and their inclusion in the satellite's assembly layout contains the following steps:

Obtaining the general project layout data from the main designing unit. Formation of the tolerance range of combinations of geometric parameters of onboard elements starting from the analysis of the initial data. Express analysis of various combinations in terms of rationality of manufacturing and the choice of three to five optimal options. The transfer of Pareto-options to enterprises-co-executors of the satellite project. Calculation of the economic and temporal characteristics of the production of the equipment modules for each option by the enterprises-co-executors and transfer of the obtained data to the parent enterprise. Analysis of materials of enterprises-co-executors. Choice of the optimal variant by a complex criterion. Creation (correction) of the standard of Modular Design (MD) or its correction.

The MD standard contains a generalization of the results of studies to determine the optimal geometric characteristics of onboard equipment modules. The MD standard should be issued before the start of the design work for the satellite.

RESULTS AND DISCUSSION

Modeling results: The implementation of the developed methods allowed to reduce the weight of the satellite's structure by 34, ..., 38% due to the inclusion of the onboard equipment into the load bearing scheme. At the same time by increasing the stability of the installation sites of attitude control system's sensitive elements, the accuracy of determining the target coordinates of objects when processing images can be increased by 15-25%.

Modeling of attitude control and thermal control systems produced the following results: the average density of the layout of the MKA compartment increases from 300, ..., 400-550, ..., 620 kg/m^3 which contributes to the reduction of the ballistic coefficient, it also allows placement of more products in the payload area of the carrier rocket. The energy consumption of the thermal control system is reduced (by 45, ..., 65%) due to the optimization of the temperature conditions of the onboard modules operation. The energy consumption of the attitude control system is reduced (by 15, ..., 24%) due to the reduction of the main moments of inertia of the MCA.

CONCLUSION

During the normal operation of the satellite, the relative loss when implementing an external factor by a strategy with optimal responses will always be less than the relative loss when the external factor is implemented with a strategy that is not optimal but acceptable. The value of the efficiency criterion vector will be high.

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