Journal of Applied Mechanical Engineering
ISSN: 2168-9873

Research Article Open Access

Smallsat Rational Design

Zosimovych N, Niu X, Simeone A and Peng X

Intelligent Manufacturing Key Laboratory of Ministry of Education, Shantou University, Shantou, Guangdong, P.R. China

Abstract

This paper is devoted to the study of CubeSat rational designing technique. The CubeSat design process is comprised of choice of its trajectory, determination of components and main parameters of its systems, development of external and internal layouts, determination of the number of satellite-born antennas and their main characteristics. The potential of CubeSat has prompted the scientific community to rethink the existing spacecraft technologies and think about how to make them suitable for CubeSats. In terms of specialization of engineering works during Smallsat development, was formulated concept of the design process and recognized physical relationships to find some optimal design solution about compatibility of basic parameters and characteristics. The purpose of Smallsat rational design is to create a project of a vehicle for which the value of the selected criterion is close to the maximum or minimum value. As a result, the design algorithm version was making. The described method of rational design, of course, does not deny the process of intuitive creative thinking. This process reveals itself in assumptions and development of reference versions, as well as in ballistic design.

Keywords: Smallsat; Rational design; Propulsion System (PS); Complexity; Spacecraft; Analysis; Probability; Trajectory; Reliability; Design solution

Introduction

The size and cost of modern spacecraft vary depending on the application. Some you can hold in your hand while others like Hubble are as big as a school bus. Small spacecraft (Smallsat) focus on spacecraft with total mass less than 180 kilograms and about the size of a large kitchen fridge. Even with small spacecraft, there is a large variety of size and mass that can be differentiated [1]:

- Minisatellite, 100-180 kilograms
- Microsatellite, 10-100 kilograms
- Nanosatellite, 1-10 kilograms
- Picosatellite, 0.01-1 kilograms
- Femto satellite, 0.001-0.01 kilograms.

The Smallsat design process is comprised of choice of its trajectory, determination of its components and main parameters of its systems, development of external and internal layouts, determination of the number of satellite-borne antennas and their main characteristics, making programs: general one and for separate sessions (Figures 1-3). Furthermore, since it is not possible to determine any basic parameters for the systems and the requirements for the control system, and to program the work without understanding the behavior of the individual systems and their interaction, these problems must be solving in the design process.

Simultaneously with the above works, on-board systems, separate units, mechanisms and blocks of Small Sat are to be developed. As a rule, the design process is following by verification of the decisions taken involving the use of laboratory and test prototypes and models of Small Sat, its specific devices and units.

In terms of specialization of engineering works in the process of Small Sat development, design and calculation works, development of logical and electric diagrams and development of computation programs, modelling and computer analyses shall be done. The calculation and the modelling process include among others [2-5]:

- Design and strength checking calculations;
- Mass, momentum of inertia calculations, the center of mass position and positions of the main inertia axes;
- Thermal calculations;
- Calculations of internal and external disturbing moments;
- Gas environment calculations for hermetic compartments;
- Estimation of probability of meteorite impact and erosion of external surfaces, determining whether special protection measures (additional screens, thicker shells, more resistant coatings, etc);
- Estimation of radiation exposure for devices, glass, coatings and structural non-metallic elements;
- Dynamic analysis purposed to determine requirements or to check stiffness of the structure to eliminate mutual undesirable influence of mechanical and mechatronic devices and systems, and operation of the orientation system;
- Ballistic design;
- Power supply system calculations, orientation system and other system calculations.

Figure 1: 12U CubeSat (Aoxiang Zhixing).

***Corresponding author:** Mykola Nickolay Zosimovych, Intelligent Manufacturing Key Laboratory of Ministry of Education, Shantou University, Shantou, Guangdong, P.R. China, Tel: 86-754-8650-3386. Fax: 86-754-8650-3442; E-mail: nzosimovych@gmail.com

Received July 11, 2018; **Accepted** August 08, 2018; **Published** August 10, 2018

Citation: Zosimovych N, Niu X, Simeone A, Peng X (2018) Smallsat Rational Design. J Appl Mech Eng 7: 311. doi:10.4172/2168-9873.1000311

Citation: Zosimovych N, Niu X, Simeone A, Peng X (2018) Smallsat Rational Design. J Appl Mech Eng 7: 309. doi:10.4172/2168-9873.1000311

Figure 2: 6U (1 × 6U) CubeSat (ALTAIR 1).

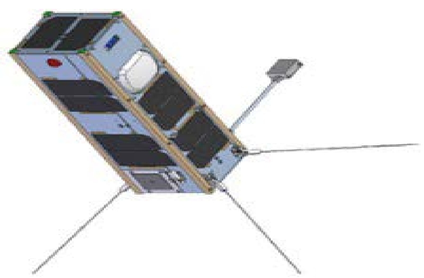

Figure 3: 3U CubeSat (CanX 2).

If we bind the design process with the development stages typical for any product [6,7], then this process should cover development and agreement of the technical specification for the Smallsat concerned, development of draft proposal, conceptual and technical design (Figure 4).

It is obvious that in the process of Smallsat design the basic parameters of separate systems, trajectory characteristics, operation program and the spacecraft design should be taking into line.

Let us specify the main physical relationships [7]:

- The speed of "board-ground" information transmission depends on Smallsat remoteness from the ground facilities, the on-board antenna gains and the output capacity of the transmitter;
- The output capacity of the transmitter determines its weight and electricity demand;
- The electricity demand of the transmitter determines the capacity of the chemical battery, which is needed for the transmission of the information, and the required power of the electric generator, which shall be used to charge the chemical battery;
- The navigational accuracy of the electric axis of the antenna is determined by the orientation accuracy of the Smallsat in general (when stationary installed, or a program controls the antenna rotations). In addition, desynchronization between the orientation sensors and the antenna axis;
- The place of the antenna on Smallsat is essentially determined by its external configuration, i.e. the form and place of individual modules, external devices and units;
- The accuracy of antenna axis orientation for an antenna hard-mounted on the Smallsat, or an antenna rotating according to a program, determines the propellant flow-rate or electricity consumption required for the orientation;
- The duration of the data transfer session depends on the amount of information to be transmitted during the session [8,9] and the transfer rate;
- The intervals between the information transfer sessions determine the capacity and therefore the weight of the on-board storage device;
- If impulse control of jet engines takes place, the consumption of the propellant, necessary for Smallsat orientation depends on the accuracy of its orientation, disturbing moment, specific thrust and control engine arms, the minimum impulse value and Smallsat moments of inertia [10-13];
- Small Sat's moments of inertia shall be determining by its overall mass and the external configuration;
- The minimum impulse and specific thrust shall; be determined by the propulsion system (PS) used for orientation, its pneumohydraulic scheme and the thrust capacity;
- The individual systems' behavior and operation program, i.e., combination of the on-board controls and their weight determine reliability of Smallsat;
- The operation program and behavior depend on trajectory characteristics of Smallsat or trajectories of several Smallsat, solving the same problem simultaneously;
- The reserves of the propellant, necessary for the trajectory correction, and the required correction accuracies shall be determined by the nominal trajectory, error matrix at the end of the launch phase, correction strategy, i.e. their number and distribution within the trajectory, specific thrust of the propulsion system, trajectory measurements and their accuracy, and the number of ground facilities receiving information;
- The dead weight and the weight of the on-board cable network depends on the external and internal configurations of the Smallsat and on vacuum tightness of its modules, as well as on the type of devices installed, the level of control centralization, operation program and behavior;
- The type and characteristics of the on-board radio communication facilities, the orientation schemes of the Smallsat and its external configuration determine characteristics of the airborne antennas;
- The total surface and weight of the solar battery, if it is used as an electric power generator depend on the electricity demand of the equipment, the operation program, the trajectory, the orientation scheme of the Smallsat, its external configuration and the type of solar cell battery (omnidirectional, semi directional, etc.);
- The external configuration of the Smallsat and optical characteristics of external surfaces determine the characteristics of forces and moments of the light pressure. In some cases, moments of light pressure can be using as useful moments helping to adjust consumption of propellant or electric power for orientation of the Smallsat.

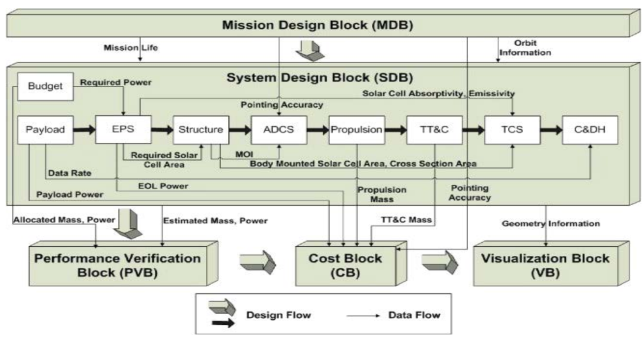

Figure 4: System engineering design tool [8].

Citation: Zosimovych N, Niu X, Simeone A, Peng X (2018) Smallsat Rational Design. J Appl Mech Eng 7: 309. doi:10.4172/2168-9873.1000311

Methodology

Physical relationships in the design process

The study of physical relationships in the design process is necessary, first, to find some optimal design solutions about compatibility of basic parameters and characteristics of Smallsat. The first task is to be certainly solving in the development of any project [14,15]. Taking into consideration relatively low cost of modern Smallsat, and the current methods of their test and control, it is difficult to imagine that the parameters of any systems could be incompatible in an orbiting Smallsat, or that its design could not provide for the operation of its devices. Such cases are extremely rare. Substantially the alignment of the basic system parameters with each other, and with the characteristics of the trajectory and the design, is the design itself in the usual sense of the word [6].

$$V_{x,mid} = \frac{v_4 + v_5}{2\cos\theta_1}, \qquad V_{y,mid} = \frac{v_5 - v_4}{2\sin\theta_1}$$

The second of the tasks set is the search for optimal combinations of parameters and characteristics. It is much more difficult than the first one and is not always solving. This is mainly due to the complexity of studies of this sort [14,15].

This complexity is aggravating by the fact that the external and internal configurations significantly influence the system parameters, mass and other characteristics of Smallsat (Figure 5).

The variety of Smallsat shapes due to minimum external shape limitations for most of them significantly complicates the formalization process enabling to find the best external configuration. Moreover, the technology CubeSat is the most effective form for Smallsat today [1].

To avoid a random choice, sometimes development of the components is assigning to different specialists with the following choice of the best option. But also, in this case the choice of the right option is often done on the basis of intuition of the project manager, and therefore, personal preferences, a wish to simplify the analysis and the following works, and other considerations are subconsciously involved in this choice, which does not always result in the best option or an option close to the best one.

At the same time, inadequate choice of the external configuration can lead to higher values of moments of inertia for Smallsat, increased weight of the on-board cable network, deterioration of characteristics of airborne antennas, complication of technology, etc.

To enable rational design, it is necessary to establish some criteria, which extreme values must be a goal in searching a combination of parameters and characteristics of Smallsat. These criteria are to be determining by the tasks set for a specific spacecraft, or a technical specification for the spacecraft, determining its purpose and operating conditions.

The variety of Smallsat shapes due to minimum external shape limitations for most of them significantly complicates the formalization process enabling to find the best external configuration. Moreover, the technology CubeSat is the most effective form for Smallsat today [1].

To avoid a random choice, sometimes development of the components is assigning to different specialists with the following choice of the best option. But also, in this case the choice of the right option is often done on the basis of intuition of the project manager, and therefore, personal preferences, a wish to simplify the analysis and the following works, and other considerations are subconsciously involved in this choice, which does not always result in the best option or an option close to the best one.

At the same time, inadequate choice of the external configuration can lead to higher values of moments of inertia for Smallsat, increased weight of the on-board cable network, deterioration of characteristics of airborne antennas, complication of technology, etc.

To enable rational design, it is necessary to establish some criteria, which extreme values must be a goal in searching a combination of parameters and characteristics of Smallsat. These criteria are to be determining by the tasks set for a specific spacecraft, or a technical specification for the spacecraft, determining its purpose and operating conditions.

Due to the wide variety of modern Smallsat, it is impossible to enumerate all the criteria that their developers may encounter (Figure 6).

For some Smallsat the weight of the scientific equipment, which may be installing on the spacecraft, can be a criterion. In the simplest case, the trajectory and the orbit injection launch vehicle shall be set. They shall determine the overall weight M_0 of the spacecraft to be injecting on the specified trajectory. In this case the weight of the equipment M_{Sc} will be equation (1):

$$M_{Sc} = M_0 - M_{SS}, \tag{1}$$

where M_{SS} – is the total weight of the service systems, frame and on-board cable network necessary to ensure the operation of the spacecraft.

Thus, in the simplest case under consideration, when the trajectory, or rather narrow range of trajectories and the launcher are specifying, the task of rational design is reducing to minimization of the total weight of service systems, the frame and the on-board cable network when it comes to mathematics. In this case, the initial weight of the vehicle M_S can be considering a design value.

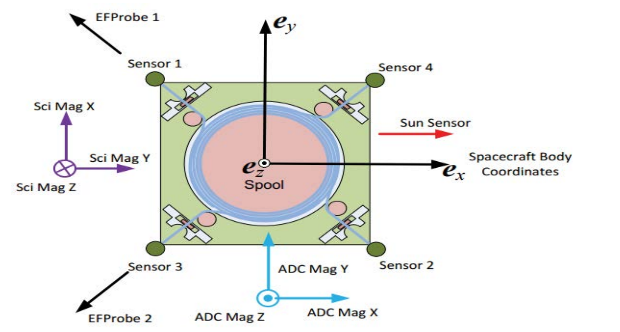

Figure 5: Dynamic Ionosphere Experiment (DICE) sensor map indicates the relative geometry, based on the spacecraft body coordinates [16].

Figure 6: Satellite comparison (Deep Space Industries).

Here we proceed from the assumption that the larger is the weight of scientific equipment, the higher is the scientific value of the spacecraft [17]. This assumption seems to be true providing a careful and informed selection from scientific tasks.

The described above approach to rational design does not depend on the weight of scientific equipment when the minimum of the total weight of the service systems is to be founding [14,15]. This approach has very limited application, as in most cases the weight of temperature control devices, electronics, the power supply system and the orientation system depend on the weight of scientific equipment, its purpose and operation program (Figure 7).

For the cases when the value M_s in the expression (1) cannot independent of the value $f_s M_s$ – sometimes it is possible to write the following [17]

$$M_{Sc} + f_{SS} M_{Sc} = M_0 - M_{Sc}^0, \tag{2}$$

instead of the specified expression, where M_{Sc}^0 – is the total weight of the service systems and the frame independent of the weight of the scientific equipment; $f_{SS} M_{Sc}$ – is an additional weight of the service systems and the frame, necessary for operation of the scientific equipment depending on its weight, composition and operation program.

Various methods of solving the problem of rational design are possible here. For example, we can search a minimum value in the expression (2), and divide the resulting value $M_{Sc} + f_{SS} M_{Sc}$ into the weight of the scientific equipment and an additional weight of service systems and frame $f_{SS} M_{Sc}$. If the function $f_{SS} M_{Sc}$ is quite simple, the expression (2) can be solved relative to the value i.e. find the expression

$$M_{Sc} = F\left(M_0, M_{SS}^0\right).$$

In this case, we can search for the maximum value directly M_{Sc}.

The above method of problem solving may be not strict enough in some cases. The matter is that the function $f_{SS} M_{Sc}$ characterizing the increase in the weight of the service systems and the frame necessary for the successful functioning of scientific equipment depends, as a rule, on the parameters of the temperature control system, the orientation system, and the power supply system (type of power generator and battery type), the frequency range of the radio telemetric system and the configuration of the spacecraft. If it is impossible to specify the above parameters and the configuration prior to the beginning of the computational analysis, it is impossible to use the formula (3) in the rational design, because to obtain the formula, it is necessary to know the exact type of function, $f_{SS} M_{Sc}$ and it is presumably determined by yet unknown versions of service systems [18].

If we seek a minimum of the value M_{SS}^0 in the expression (3), i.e. neglect the weight of the scientific equipment, it also can be approximate, as the versions of the service systems disregarded in the design process can provide a smaller value $f_{SS} M_{Sc}$ than the chosen versions [17]. In addition, it is possible that the type of function depends on the research program, which in its turn is determining by the value M_{Sc}.

Another method of solving the problem is more precise, though more painstaking. It supposes determination of the weight of the scientific equipment, its components and the operation program for all the versions of the service systems and configurations of the specified small Sat provided M_0. Here each version of the service systems and configuration is basically provided with complete or almost complete development of the project and final adjustment of the basic parameters of all on-board systems and characteristics of small Sat (Figure 8). As a rule, rational design in this case must be done by successive approximations. In this case, the expression of the type (3) can always be using to solve some specific problems. For example, for a power supply system consisting of some solar panel and a chemical battery of a type, the function $f_{SS} M_{Sc}$ can be easily specifying if it is possible to determine the dependence of the average electricity demand for the scientific equipment from its weight [5].

The last of the considered methods of solving the problem of rational design, proposes that the program of scientific research or works [17] conducted with the application of the spacecraft and not the weight of scientific equipment should be using as a criterion. The best version shall be the version of the spacecraft for which the program is most complete from any point of view, i.e. in this case we require a special criterion for evaluation of various research programs. The amount of scientific information transferred from the spacecraft during the flight [10] may be choosing as such a criterion for evaluation of different programs.

To illustrate the above, we shall note that it is theoretically possible, when between two versions of Smallsat the best one will be with lighter scientific equipment and with greater electricity demand for it, which enables more complete program of scientific research [18].

The reliability of the newly designed spacecraft should be indicating as a possible criterion. In some cases, a trajectory or orbit, a launch vehicle, the combination and characteristics of the scientific equipment, as well as the operation program for the equipment may be specifying. The trajectory and the launch vehicle, in its turn, determine the initial weight of the spacecraft (Figure 9).

The reliability can be expressing through probability of implementation of the basic task under which here it is necessary to understand operation of scientific equipment according to the specified

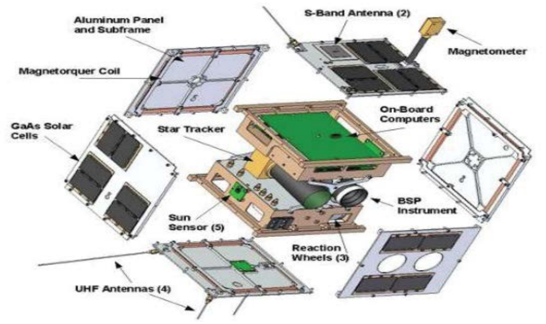

Figure 7: Basic components of small satellite.

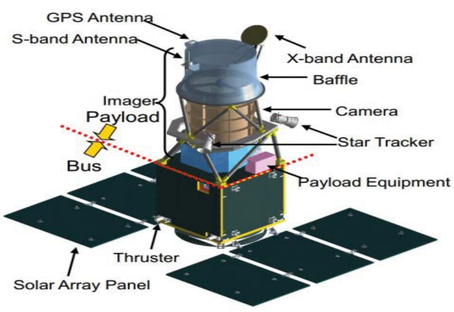

Figure 8: Satellite ASNARO-1.

Citation: Zosimovych N, Niu X, Simeone A, Peng X (2018) Smallsat Rational Design. J Appl Mech Eng 7: 309. doi:10.4172/2168-9873.1000311

program within the given time. This time is sometimes calling the vehicle operation time, or active existence time [17].

For numerical estimation, the reliability shall be regarding as a probability of flawless operation within a specified time, the failure being such condition of the on-board systems and devices, which makes impossible further functioning of the scientific equipment. To calculate the probability, it is possible to use the theory of reliability apparatus [3].

If we indicate the probability of flawless operation of the spacecraft within a specified time t_0 as B, we can write

$$B = B\left[\left(C_{m,n}\right),\left(T_i\right),\left(P_j\right),t_0\right], \tag{4}$$

Where $\left(C_{m,n}\right)-$ is the finite set of basic parameters of systems; m- is the system number; n- is a parameter number; (T_i) is a parameter set determining the trajectory of the spacecraft; (p_j) parameter set determining the operation program.

Among the many parameters of the system, there can be those, which are uniquely determining by the composition and characteristics of scientific equipment, and its operation program. The remaining parameters are free. Moreover, their choice is the result of rational design. A similar remark can be mentioning concerning parameters (T_i) and (p_j) for example, if we design an artificial Earth satellite with the specified height of a circular orbit and the specified deviation, the orbit injection time shall be free parameter. This parameter determines the orbit position relative to the Sun and stars and can be selecting to ensure maximum reliability of the orientation control system at the beginning of orbital motion, particularly when searching and capturing support landmarks. It should be noting that the probability value of flawless operation of the spacecraft is not significant. This value shall be using only as a criterion for analyzing different design solutions. If the technical task sets the value of reliability, B_o the process of rational design shall consider the condition $B \geq B_o$

The probability of Small Sat's flawless operation shall be determining by the reliability of its individual systems. The reliability of the system, in turn, is determining by the tasks solved by this system, its operation program and the general weight of the system, as the larger the weight, the larger is the number of redundant elements and devices, which can be using in this system. In this regard, the distribution of weights within separate service systems of the spacecraft becomes of special importance in the task under consideration. It is possible that this distribution of weights shall be sought outing as an extreme value search result [3].

We can specify the probability of solving this problem and minimize the number of Smallsat intended for solving the problem. In this case, if the scientific equipment must solve several tasks irrespective whether these tasks are to be solved during one launch or during several launches, it is good when comparing different project solutions not only to determine probability to perform all the tasks during one start, but as well the probabilities of performing individual tasks. This is because according to the assumptions made above, the number of spacecrafts solving all the tasks with a given probability will depend on the above probabilities. The necessary formulas can be obtaining from the known laws and probability theory formulas [18].

We shall note that the minimum number of Smallsat corresponds to the minimum cost of solving the problem or the minimum time to complete the entire program.

A criterion may be time of spacecraft functioning with reliability, which is not less than some set value [14]. Such a criterion is possible in cases when small Sat performs tasks continuously during the whole mission (communication satellites, meteorological satellites, Sun observation satellites, etc.).

When developing small Sat, you have to deal with a number of restraints and requirements that need to be taking into account in the design process. Let us consider some of them:

- The purpose shall be the main requirement for a newly developed small Sat (Earth satellite in low orbit, in the middle orbit, etc.).
- When the launcher and the trajectory are specifying, the initial weight of the spacecraft should not exceed some value that can be considering as specified in project development.

Installation of the small Sat in the specified launcher requires that the overall dimensions of the satellite be within specified limits. Sometimes to meet this requirement some of the external devices and units must be placing on the unfolding elements of the spacecraft. In this case, the above devices and units take up their work position after separation of the spacecraft from the launcher. Such devices and units include those that have significant dimensions (e.g., solar panels and antennas) or those that should be removing from the main scope of the devices in the working position. The latter group includes some scientific instruments, such as magnetometer sensors.

- Earlier, when considering the criteria, we noted that a scope of scientific equipment might be specifying for some Smallsat, however, launchers and trajectories determining the initial weight of the satellites may be missing. It is obvious, that with some initial weight values of the spacecraft it is impossible to ensure work of scientific equipment with a specified weight.
- We can specify not only the weight of scientific equipment but also its characteristics and operation program.
- Often when developing a new small Sat, we can be guiding by available ground control and data reception terminals. This restricts in a way the satellite's operation program and its trajectory.

A few restraints and requirements that need to be considering in the small Sat design can be representing as fixed values of some basic system parameters, trajectory parameters and operation program or as maximum permissible values for some of these parameters. In these cases, restraints and requirements must be expressing as some constants in equations that specify physical relations. Unlike other values that may be included in these equations, constants do not change during the analysis of different versions of design solutions.

Restraints and requirements to the newly developed small Sat and first its purpose is expressed not only in the form of constants in equations of physical relations [15], but also in presence or absence of the equations and in the form of the equations themselves. This is natural, since the composition and technical meaning of the basic parameters of the systems, trajectory parameters and parameters of the operation program depend on the schemes of the on-board systems, the structure of the operation program and the flight scheme [6]. These parameters are essentially dependent on the purpose of small Sat and several restrictions and requirements. Hence, the equations of physical relations showing interdependence of all the above parameters [18] depend on the purpose of the satellite, limitations and requirements thereto (Figure 10).

Besides the availability of partial restraints and requirements

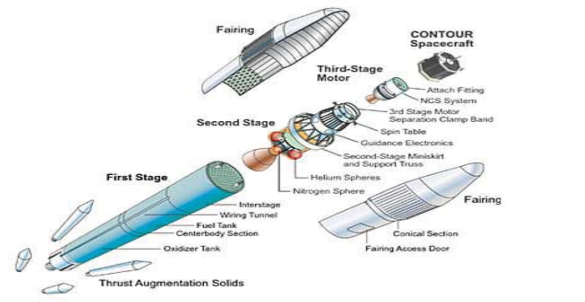

Figure 9: Smallsat position on a launch vehicle.

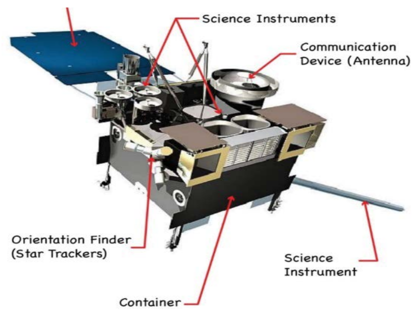

Figure 10: Smallsat design solution.

narrows the range of basic system parameters, trajectory parameters, operation program and even configuration diagrams considered in the design process. In this case, some equations of physical relationships will not have any solution if there are constants in these equations determining partial restraints and requirements.

It follows from the above that in the process of rational design we must consider equations and inequalities determining physical relationships characteristic for the spacecraft of this purpose or type, and limitations and requirements to it. These expressions will include some constants.

These equations and inequalities shall be writing as follows

$$\Phi_r\left[\left(C_{m,n}\right),\left(T_i\right),\left(P_j\right)\right]\begin{cases}=0,\\ \geq 0,\end{cases} \qquad (5)$$

where $n=1,2,...,N_m; N_m-$ is the number of basic parameters of m-th system; $m=1,2,...,M; M-$ is number on-board systems; $i=1,2,...,I; r=1,2,...,R;\ j=1,2,...,J$. Using the introduced symbols for all the basic parameters we can write the following expression

$$N_\Sigma=\sum_{m=1}^{M}N_m+I+J. \qquad (6)$$

To equations and inequalities (5) must be completed with an equation determining dependence of the chosen criterion K from the parameters $\left(C_{m,n}\right),\left(T_i\right),\left(P_j\right)$, i.e.

$$K=K\left\lfloor\left(C_{m,n}\right),\left(T_i\right),\left(P_j\right)\right\rfloor. \qquad (7)$$

In general, the expressions (5) and (7) may include time.

If all the expressions (5) are equations $R<N_\Sigma$, and then the task of seeking for optimal parameter values is confining to finding a constrained extrema of the many variables function. The relations of type (5) are simultaneously the constraint equations.

If some relations (5) are inequalities, the task of seeking the variable $\left(C_{m,n}\right),\left(T_i\right)$ and (P_j) minimizing the value *K*, is to be referred to as a task of linear or non-linear programming depending on the type of functions K and Φ [18].

The purpose of rational design is to create a project of a vehicle for which the value of the selected criterion is close to the maximum or minimum value. In this case, different configuration diagrams, different orientation schemes and different methods of creating control and corrective forces, etc. [18] should be considering. Depending on the versions of design solutions the functions K and Φ will change. Consequently, the rational design shall be confining to the investigation of the function K in the constraint equations (5) for different versions of the newly designed small Sat.

Finding the optimal parameter values for one record variant of functions K and Φ shall be describing as a specific task of rational design. This is essentially the task of optimizing some specific version of small Sat.

In some cases, the analysis of physical relations characteristic for some versions of small Sat allows to find an optimal combination of some parameters, which simplifies the solution of the rational design problem. Mathematically the above means that it is possible to extract from the system (5) a subsystem including only some parameters and find some specific criterion depending on these parameters and not contradicting the general criterion *K*. the tasks of this type can be called specific optimal tasks of small Sat design.

It should be noted that identification and solution of the optimal tasks of the type considered above can be of separate interest regardless of the general task of rational design at least because the solution of these problems can be significantly simpler, than the solution of the general problem [14].

The most probable criterion except for weight and reliability of the system or a group of the systems may be designating as information capacity of the "board-Earth" transmission line. This is also relevant for any spacecraft [17].

Physical relationships in the design process

Before we can provide a variant of the rational design algorithm let make some assumptions concerning equations and inequalities in expressions (5) [18]. These assumptions specify a class of spacecraft for which they are true and for which we are going to offer an algorithm.

The relations (6) include only equations, inequalities are absent. This is because inequalities arising from the requirements to the spacecraft and from limitations can be replacing by equations for many spacecraft's. If it is required that the initial weight of the spacecraft does not exceed the specified value determined by the trajectory and the launch vehicle, then when analyzing the different versions of small Sat and identifying its optimal parameters it is possible to accept that the initial mass of the spacecraft M_0 is equal to the maximum permissible value $M_{0_{\max}}$ minus some allowance i.e.

$$M_0=M_{0_{\max}}\left(1-\frac{\Delta M}{M_{0_{\max}}}\right).$$

The relative weight allowance $\Delta M / M_{0_{\max}}$ may be accepted within 0.1-0.2 [17] depending on the complexity and novelty of the developed small Sat and its systems.

Similar reasoning can be presenting also for the case when the minimum permissible reliability of small Sat is specifying, and in relations (5) we shall accept, that the reliability of the spacecraft is equal to this value with some allowance, which can disappear at the stage of detailed design.

A parallel development of separate compartments or modules and the spacecraft and matching of the parameters by successive approximations are possible as well. A detailed study of these issues is beyond the scope of this publication.

In the equation (5), time is not explicitly included, and all the parameters are constant values. Accordingly, these equations do not include any differential equations. This assumption is true in cases when there are "points concentrated" fragments with running engines in separate phases of flight, i.e. the fragments, which are characterizing only by speed modules necessary for trajectory corrections and maneuvers [18]. This assumption is appropriate when the propulsion system produces quite a high thrust.

If some of the parameters in the equations (5) are time-dependent, for example, the weight of a spacecraft or its moments of inertia due to fuel consumption on during correction phase, maneuvers or orientation process, the time may be included as constants obtained during the ballistic design phase. For example, the equations (5) may include members $\dot{m}t_1$ and $\dot{m}t_2$, where $\dot{m}-$ is an average fuel consumption per second need for orientation of the spacecraft. It is one of the varied parameters of the orientation control system depending on the moment of inertia, arms of the driven engines, disturbing moments, etc., t_1 and t_2- are constants determining the times of characteristic points within the flight trajectory.

The varied parameters do not include any parameters and characteristics of small Sat trajectory. It is assuming that the choice of the flight scheme, the basic parameters of the trajectory, as well as determination of the requirements to the spacecraft in terms of implementation of the necessary trajectory has been in advance, before determining the parameters of systems, configuration and the operation program. Such a stage of work, which is calling ballistic design, can often be starting immediately after receiving technical specifications for the small Sat. In cases when there is a dependence of the trajectory parameters from the parameters of some systems, and the latter cannot be determined in advance before the complex investigation of the spacecraft parameters, it is necessary to use the method of consecutive approximations. Ballistic design is an independent area of spacecraft design [18]. The ballistic design shall result in determined trajectory characteristics, the initial weight of the spacecraft, which can be injected in the specified trajectory by the chosen launcher, characteristic speeds, times for corrections and maneuvers, requirements to the control actions for the thrust vector positioning during corrections and maneuvers, and the necessary accuracies, and, in addition, all necessary data for the development of the orientation control system and operation program, such as, for example, the angles between possible optical guides and the times when the spacecraft is in the visual range of ground facilities. It should be noted that at the stage of ballistic design may be necessary to solve complex variation problems, multipoint boundary value problem, etc. Some of these tasks are studying in the publication [18]. The number of constraint equations (5) is less than the number of varied parameters. If this assumption is not fulfilled, the task of selecting the optimal parameters cannot be solved, as there are no free parameters to minimize the criterion K. Most likely it means that some free parameters have not been revealed, and it is necessary to review the parameters and the type of functions K and Φ.

During this assumption, the task of finding the optimal parameters is limited to finding a constrained extremum of multivariable function K in the constraint equations (5). In this case, the equations (5) and the criterion K shall include parameters characterizing the configuration, design and the operation program as varied parameters.

Results and Discussion

It was previously noted that the type of functions Φ and K depends on the versions of design solutions the configuration scheme, the orientation scheme, the construction and operation scheme for individual on-board systems, etc. It does not allow reducing the design process to solving one mathematically strictly set problem. This complexity can be overcoming in the following way.

Let assume that at some stage of the design process we found a satellite and a program version that met all the requirements and restrictions. This version of the satellite shall be calling a reference one. Suppose that it is characterizing by parameters $(\bar{C}_{m,n})$ and $(\bar{C}_{m,n})$ that we call the source parameters. These parameters will satisfy the equations (5), which consider the physical relations and constraints characteristic for the found small Sat variant.

Obviously, the combination and technical signification of the small Sat parameters, and therefore the structure of the expressions (5) and the constants included therein will not change when the parameters within some intervals near the values $(\bar{C}_{m,n})$ and $(\bar{P}_j)$ change. We shall introduce the following symbols for the specified intervals of each parameter:

$$\left[C'_{m,n}, C''_{m,n}\right], \left[P'_j, P''_j\right]. \tag{8}$$

For convenience, further under the reference version we will understand the version characterized by parameter variation intervals (6).

Optimization of the reference version is limited to search of parameter values $(\bar{C}_{m,n})$ and $(\bar{P}_j)$ within intervals (6) and implementation of the maximum or minimum value of the criterion during execution of equations (3), written for the reference version.

It should be noting that the boundaries of intervals (6) shall be establishing from the applicability condition for expressions (3) for all values of parameters within these intervals, and their boundaries must be determined in advance for the reference version under consideration.

If there are several reference versions of the spacecraft, then each of them shall be providing with a system of relations (3), and the parameter variation boundaries (6) can be drawing up from the applicability conditions of these relations. After that, it is possible to optimize the versions and find extreme values of the criterion K for them using one or another mathematical tool. The comparison of values K will allow choosing the best version of the spacecraft with its optimal parameters in terms of the accepted criterion. This can be considering a complete process of rational design.

It may be that solution of the complete problem regarding the extrema of the criterion as a function of all basic parameters of the satellite may turn too laborious process even for one reference version. In these cases, it will be necessary to break the problem into a series of common problems and use the specific criteria mentioned above.

It is very important that the experience of creation and operation of similar spacecraft be using in the development of the versions. The qualification of the developers of the reference versions is of paramount importance. However, it should be borne in mind that the newly created small Sat may not have prototypes. In such cases, a sufficiently wide review of the possible reference versions is required.

Citation: Zosimovych N, Niu X, Simeone A, Peng X (2018) Smallsat Rational Design. J Appl Mech Eng 7: 309. doi:10.4172/2168-9873.1000311

The described method of rational design, of course, does not deny the process of intuitive creative thinking. This process reveals itself in assumptions and development of reference versions, as well as in ballistic design.

It should be noting that design of any spacecraft must consider considerations related to technological capabilities, a possibility to order devices and equipment from external organizations, costs, etc. The above considerations must be considering in the process of rational design when reference versions are being developing and the boundaries of the parameter change intervals [6] are being specifying.

In some cases, the launcher may not be included into the technical specification and will not be easily identifying during the analysis of this specification. In such cases, it may be necessary to dedicate the whole process of rational design or its part to suggest different options of the launcher. Here under the launcher we understand stages of a basic launch vehicle and additional stages necessary for injecting the newly developed small Sat into its flight trajectory.

As parameters characterizing external configuration of the reference variant, it is necessary to choose such parameters which influence the characteristics and parameters of the on-board systems, weight of the satellite and an ellipsoid of inertia (for example, sizes determining the surface of a flat solar panel, and the size determining its position relative to any base surfaces; dimensions of the instrument compartment, etc.). Obviously, the number of parameters should be minimal to avoid unnecessary complication the general extremum problem. If the surface of the solar panel can be finding from the analysis of the power supply system [18], the parameters characterizing the external configuration must include only the relation of the panel sides because this relation may depend on the satellite's moments of inertia, the panel weight and the moment of light pressure. Generally, it should be noting that the identification of parameters subject to variation, at the stage of solving an extremum problem is a very responsible operation. Minimization of the total number of parameters shall be taking only based on intuition of developers at the stage of reference version development (Figures 1-3).

In the context of the amount of calculations, the most complex is the stage of finding optimum parameters and extreme values for the criterion for all reference versions. For each reference variant, the problem is to study the function extrema

$$K\left(x_1, x_2, \ldots x_{N_\Sigma}\right) \tag{9}$$

in constraint equations….

$$\Phi_r(x_1, x_2, \ldots, x_{N_\Sigma}) = 0, \tag{10}$$

where $r = 1, 2, \ldots, R;$ $x'_k \le x_k \le x''_k, k = 1, 2, \ldots, N_\Sigma;$ x'_k, x''_k correspond to the parameter variation intervals boundaries (6).

There are different methods of solving this problem. First, we can try to exclude some parameters x_k using equations (8) and to investigate unconditional extrema of the function K of already N_z -R variables.

If it is difficult, i.e. to investigate the extrema of the function

$$\overline{\Phi} = K + \sum_{r=1}^{R} \lambda_r \Phi_r. \tag{11}$$

In this case, the required optimal parameters and multipliers λ_r are to be founding from necessary existence conditions of the internal maximum or minimum function $\overline{\Phi}$

$$\frac{\partial \overline{\Phi}}{\partial x_k} = 0, k = 1, 2, \ldots, N_\Sigma \tag{12}$$

and equations (8). However, we should keep in mind that the function $\overline{\Phi}$ can take the maximum or minimum value when the parameters are within the interval limits (6). In such cases, the conditions (9) are not applicable.

With a relatively small number of parameters, we can use the methodically simplest way to iterate through all possible combinations of parameters. The fact that all variable parameters have a clear physical meaning is intending to be for the benefit of this method. Therefore, in development of the reference versions it is possible to identify both the parameter variation intervals and technically practical minimal steps towards their change, connected with possible accuracy of realization of these parameters at the subsequent stages of development and manufacture of the spacecraft and its systems.

We will mention another possible way of solving an extremum problem, based on the varying parameters of a reference version of small Sat. Let us assume that this variant is characterizing by a set of parameters $(\overline{x}_k)$, which were referring to as initial ones above. You can try to linearize the functions K and Φ by developing them, for example, into the Taylor series near the point $(\overline{x}_1, \overline{x}_2, \ldots, \overline{x}_{N_\Sigma})$ and leaving only first-order terms about variations of the parameters $\Delta x_k = x_k - \overline{x}_k$.

The validity of such linearization can always be set, knowing that where and are the limits of the variation intervals for parameter If we take a maximum permissible error of the criterion we shall get

$$\Delta K = K - \overline{K} = \sum_{k=1}^{N_\Sigma} \frac{\partial K}{\partial x} \Delta x_k, \tag{13}$$

resulting from linearization, where $\overline{K} = K\left(\overline{x}_1, \overline{x}_2, \ldots, \overline{x}_{N_\Sigma}\right);$ $x'_k - \overline{x}_k \le \Delta x_k \le x''_k - \overline{x}_k$.

In this case, we get a system of linear equations:

$$\left.\begin{array}{l} \dfrac{\partial \Phi_1}{\partial x_1}\Delta x_1 + \ldots + \dfrac{\partial \Phi_1}{\partial x_{N_\Sigma}}\Delta x_{N_\Sigma} = 0, \\ \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots \\ \dfrac{\partial \Phi_R}{\partial x_1}\Delta x_1 + \ldots + \dfrac{\partial \Phi_R}{\partial x_{N_\Sigma}}\Delta x_{N_\Sigma} = 0. \end{array}\right\} \tag{14}$$

instead of the constraint equations (8).

In the expressions (10) and (11) all the partial derivatives shall be calculated in the point $(\overline{x}_1, \overline{x}_2, \ldots, \overline{x}_{N_\Sigma})$. by excluding a part of parameters using equations (11), we shall find a formula for ΔK as follows

$$\Delta K = \sum_{l=1}^{N_\Sigma - R} e_l \Delta x_l \tag{15}$$

instead of the constraint equations (8).

This is a hyperplane equation in a multidimensional space $(\Delta x_1, \Delta x_2, \ldots, \Delta x_{N_\Sigma - R}, \Delta K)$.

Therefore, the value ΔK can take the largest or the smallest value only at the boundaries of the value variation intervals Δx_l.

Thus, here finding of an optimal combination of parameters Δx_l shall be confining to calculation of the value for all possible variation intervals boundary combinations ΔK and selection of a combination Δx_l implementing the maximum or minimum values ΔK. Please note that the described method is a special case of the linear programming problem.

Conclusion

This publication summarizes CubeSat technology, provides examples of their scientific impact, and describes the design and the manufacturing of a CubeSat platform. The small Sat design process is comprised of choice of its trajectory, determination of components and main parameters, systems, development of external and internal layouts, determination of the number of satellite-born antennas and their main characteristics. Proposed paper will focus on estimating a concept and physical relationships in the design process, and on the rational design algorithm version. In terms of specialization of engineering works during Smallsat development, was formulated concept of the design process and established physical relationships to find some optimal design solution about compatibility of basic parameters and characteristics.

References

1. Mabrouk E (2015) What are Smallsat and CubeSats? NASA.
2. Bayley DJ (2007) Design optimization of space launch vehicles using a genetic algorithm. Auburn University Alabama, USA.
3. Furman TT (1981) Approximate methods in engineering design math. Int Sci and Eng p: 155
4. Zosimovych N (2017) Modeling of spacecraft centre mass motion stabilization system. Int Ref Journ Eng Sci 6: 34-41.
5. Zosimovych N (2016) Commercial launch vehicle design LAP Lambert Academic Publishing, Riga, Latvia. p: 184.
6. Nikolsky VV (2007) Basics of designing automatic space vehicles: SEDT (System Engineering Design Tool) development and its application to small satellite conceptional design. Acta Astronautica 61: 676-690.
7. Zosimovych NV (1997) Substantiation of monitoring space system parameters of the Earth. International Conference on Modern Technology. In: Aerospace Complex (Zhytomyr State Technological Institute) Sept. 7-9 pp: 170-173.
8. Аппазов РФ, Сытин ОГ (1987) Методы проектирования траекторий носителей и спутников Земли (М: Наука) p: 440.
9. Жданюк БФ (1978) Основы статистической обработки траекторных измерений (М.: Машиностроение) 384.
10. Zosimovych N (2017) Improving the spacecraft center of mass stabilization accuracy. IOSR J Eng 7: 7-14.
11. Zosimovych N (2017) Increasing the accuracy of the center of mass stabilization of space probe. International Conference on Новината за Напреднали Наука 10: 31-37.
12. Zosimovych N (2017) General technique for solving structural and parametric optimization of flying vehicle structures. Int Conf on Naukowa Przestrzeń Europy 11: 50-63.
13. Zosimovych N (2017) Structural and parametric optimization for flight vehicle structures. Int Sc Pract Conf Areas Sci 8: 59-68.
14. Fish C, Swenson C, Neilsen T, Bingham B, Gunther J, et al. (2017) DICE Mission design, development, and implementation: Success and challenges 26th Annual AIAA/USU Conference on Small Sat p: 1-20.
15. Сихарулидзе ЮГ (1982) Баллистика летательных аппаратов (М: Наука) p: 351.
16. Абалакин ВК, Аксенов Е, П Гребеников ЕАидр (1976) Справочное руководство по небесной механике и астродинамике (М: Наука) p: 864.
17. Тарасов ЕВ (1970) Алгоритм оптимального проектирования летательного аппарата (М: Машиностроение) p: 344.
18. Максимов ГЮ (1980) Теоретические основы разработки космических аппаратов (М: Наука) p: 320.