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Features of space solar power station control system

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Abstract
The article is devoted to the creation of a control system for the orbital segment of a solar space power station and the principles of its creation. The simulation of the functioning of a demonstration solar space power station operating in an elliptical orbit with energy transfer in low orbit sections was conducted and high-precision control of the spacecraft at low altitudes was considered. The result of the performed studies is the computational confirmation of the capabilities of the measuring, calculation and orientation to provide the required accuracy of the orientation of the spacecraft for demonstrating the energy transfer from orbit to the receiver system. The obtained results give grounds for the construction of a demonstration solar space power station in the low orbit.

1. Introduction

Solving the problems of energy and environmental crises, and related problems, show themselves in recent decades as the most important sociopolitical needs of society, which can determine in the long term the pace of space technology development and promote modernization and innovative development. The rise in energy prices creates threats to economic crises. The damage from natural disasters repeatedly exceeds the cost of the largest space programs. The likely cause of increased floods and droughts is the unacceptable environmental load by means of terrestrial energetics that provides the ever-increasing needs of mankind. Active search is being made for non-traditional alternative energy sources, including solar ones [1-3].

An effective way to solve these problems is the creation of space solar power stations (SSPS) with a capacity of 1-10 GW with wireless electricity transmission to ground consumers and electricity supply of environmentally harmful and energy-intensive industries from outer space in the future. Such countries as the United States, Russia, Japan, China, the EU are considering the possibility of replacing all terrestrial energetics by space energy in the near future. Recent achievements in many areas of development indicate great prospects for the creation of space solar energy systems and space energy in general.

The analysis of the existing projects of SSPS showed the presence of the three most significant problems which need to be solved when implementing such power stations [4-6]. The first and most important problem is the creation of a large-size framework of photoconverters. The second problem is the creation of a highly efficient energy transfer channel with accuracy of guidance not less than 1 angular sec. . The stability of pointing should be maintained for a long time. The third problem is to ensure the optimum thermal regime of the SSPS, the heat must be evenly distributed and diverted to avoid deformation of the SSPS construction and the extension of life of the SSPS electronic component.

The ideology of constructing of the proposed SSPS of SSPS is based on the following four principles:

- creation of effective photo-emitting panels of a large-sized construction on the satellite;
- building of solar energy photovoltaic system;
- use of a short-wave SHF band or laser radiation to transfer energy from the satellites to the rectenna;
- location of the rectenna

The SSPS consists of three segments:

- space segment based on the latest satellite technologies with large-sized constructions of photoconverters;
- the second segment is a laser channel for energy transmitting to the Earth;
- and the third segment is a ground station for receiving and converting laser or microwave energy into electricity.

To implement the ambitious idea of placing the energy infrastructure in orbit, the stage of the demonstration space experiment with the maximum efficiency and power should be preceded, which can be created on the basis of modern achievements in electronics, circuit technology, and composite materials.

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The implementation of such project will help convince the public of the need to develop this energetics field.
When developing a demonstration space power station project, it is proposed to use a modern heavy class launch vehicle to obtain the maximum area of the photo-emitting modules.
The demonstration solar space power station (DSSPS) [7-12] includes:

- service platform based on the platform "Navigator" (Lavochkin Association), with an electrojet propulsion system;
- solar panels and an accumulator battery (AB) with ionistors;
- hexapod (six-link adaptor with the possibility of coordinated precision linear links elongation);
- a device for energy transferring to the ground.

When constructing the demonstration solar space power station (DSSPS), a modular principle is used in those elements whose composition and mass depend on the type of the launch vehicle. These include, mainly, the power supply system and the electrojet propulsion system. The source of electric power for the DSSPS power supply system is solar batteries with an initial electric power of 100 kW. Part of the power of the supply system is used for the life support of the spacecraft and electrojet propulsion system (about 10%). A set of on-board systems is installed on the satellite platform of the SSPS, ensuring the operation of the spacecraft at all stages of operation, including the possibility of insertion into orbit, control, orientation and stabilization at all stages of the injection, ensuring the operation in the final orbit, including implementation of corrections of the final orbit in order to guide satellites and conduct remote energy transmission experiment.
Currently, research studies and the development of a concept for the creation of a solar space power station have been completed [7-12]. Results of the carried-out work are given below

2. Features of solar space power station control system

An important component of DSSPS is the construction of a control system to minimize the risks associated with its operation. Let's consider the basic principles of its creation.
To build an opto-geometric scheme of a solar space power station for transferring energy from orbit to the Earth, it is necessary to observe the accuracy parameters of this scheme.
The values of the required accuracies, given for the guidance of the microwave transmission channel, are shown in figure 1.

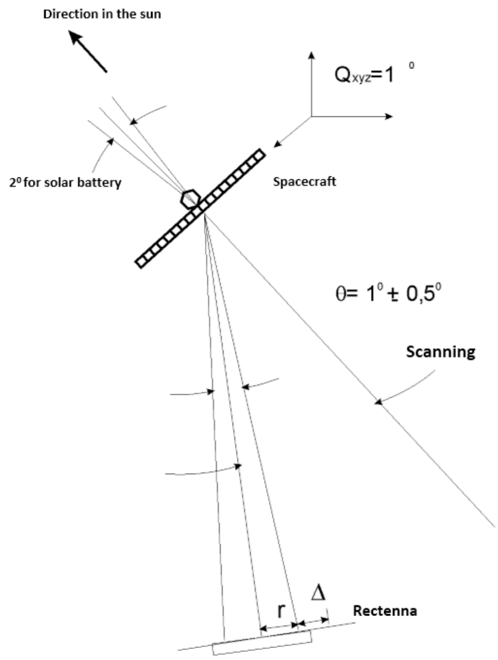

Figure 1: Optical-geometric scheme of the experiment

The task of pointing the channel of energy transfer to the rectenna can be divided into two tasks:
- the problem of controlling the angular motion of spacecraft;
- the task of pointing of the power transmission channel towards the received pilot beam.

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In addition, the control system of the spacecraft should ensure the guidance of photoconverters on the Sun with an accuracy of +2 degree. Considering that the beam-steering system has the ability to change the direction of the beam in the +30 degree range, the task of fulfilling the requirement of the guidance of photoconverters on the Sun for the spacecraft control system becomes decisive.
When designing a spacecraft for this mission,, it is necessary to solve three major technical problems:

- firstly, to conduct choice of the actuators of the spacecraft angular motion control system,
- secondly, to ensure the stability of the angular motion of the spacecraft under conditions that are apparently of a low frequency of elastic vibrations of the panels of the photo-emitting modules;
- Thirdly, to select the orientation modes and the required instrumental composition of the measuring means.

In order to obtain sufficient electric power for the mission of the spacecraft, it is necessary to orient the normal to the plane of the solar panels to the center of the sun and stabilize it with respect to this direction with an accuracy not worse than 2 deg. This requirement refers to the need for the constant solar orientation mode during the execution of the satellite's mission,.
The large area of the photo-emitting panels (from 270 m^2 to 500 m^2) entails not only significant moments of inertia of the satellite, but also their greater difference between the body axes of the satellite. For the analysis of flight conditions, their values for the three coupled axes of the satellite were taken on a preliminary calculation equal from 20000 $kg \cdot m^2$ on one axis to 146000 and 160000 kg x m^2 on two other axes respectively Such a significant difference in moments of inertia also creates significant gravitational perturbation moments. The maximum of these moments acts along the axis of the intermediate moment of inertia and reaches 0.25 Nm. A slightly smaller torque level acts with respect to the axis of the maximum moment of inertia - 0.2 Nm, and their lowest values - along the axis of the minimum moment of inertia - 0.025 Nm. The magnitude of the maximum of the momentum along the body axes is 420, 70, and 540 Ns, respectively. Their parrying by the operation of jet engines of stabilization is inexpedient in view of the very high costs of the working fluid. So with a specific impulse of the engine 220 s, their installation at a distance of even 10 m from the center of mass, fuel consumption per turn will be up to 0.05 kg. In the worst case, the daily consumption is 0.75 kg, and for three years of existence - 370 kg. In addition, the operation of jet engines of stabilization can cause significant amplitudes of elastic vibrations of solar panels and worsen the operation of the microwave antenna transmission antenna guidance system.

3. Study of the dynamics of spacecraft with a high-precision guidance of laser beam

Within the framework of this research, an experiment was conducted where the greatest interest is the creation of a demonstration solar space power station operating in elliptical orbit with energy transfer in low orbit sections. Therefore, it is advisable to consider the high-precision control of the spacecraft at low altitudes.
Since the time of the experiment is 5 percent of the time of passage of the orbit onboard the spacecraft, it is planned to use a complex of lithium-ion batteries interacting with an ionistor battery, which will allow to slightly increase the power of the beam of the energy transmitter.
Ionistors have a very high specific power, which does not depend on the temperature range. Thus, when using ionistors paired with a powerful energy source, the peak power of the power system of the the transmitter increases.

3.1 The choice of the experiment orbit

From the conditions of the required but safe power density in the focusing beam (from 300 to 60 W/m^2) and the stable position of the orbit above the given region of Kazakhstan, provided for the demonstration experiment on remote energy transfer with the necessary accuracy of the experiment (up to 1.5 ").
The orbit from the conditions of the optimal power of the radio transmitter, the operating conditions of the energy transmitter is chosen to be 12 hours.
Figure 2 shows the conditional passage time of the track points, counting from perigee. The times are symmetrical with respect to the line of the aspid - line connecting perigee and apogee.

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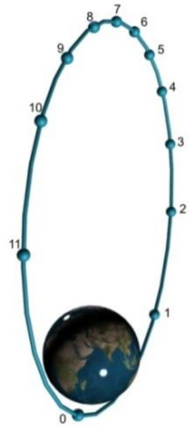

Figure 2: Orbit of a spacecraft with propagation times

The orbital parameters of the DSSPS spacecraft are shown in Table 1.

Table 1: Orbital parameters of DSSPS spacecraft

Orbital parameters	**Parameter values**
Orbital inclination	62.8 degree
Ascending node	224.4860 degree
Eccentricity	0.7417980
Perigee argument	49.1126 degree
Mean anomaly	255 degree
Apogee altitude	39980 km
Perigee altitude	above 500

Figure 3 shows traces of the trajectory of two successive orbits of the DSSPS spacecraft.

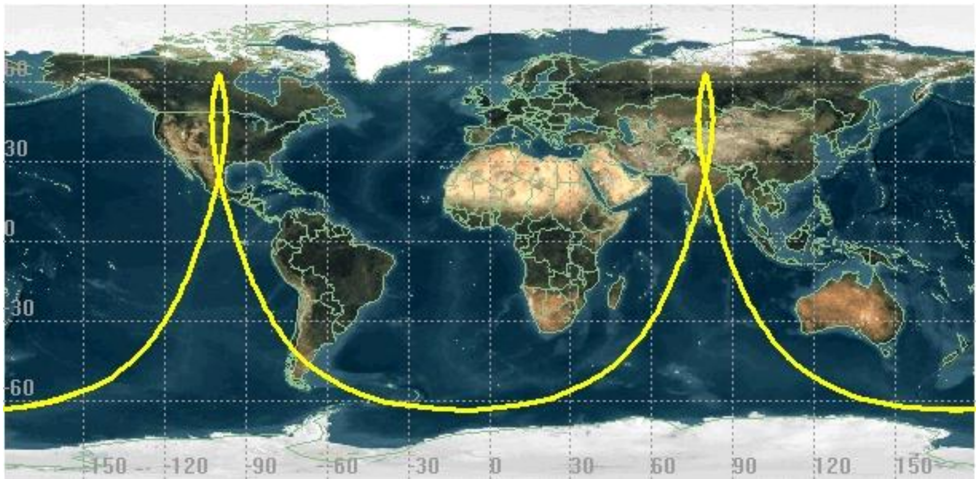

Figure 3: Trajectories of DSSPS spacecraft

The experiment was conducted in the range of angles of 225 - 255 degrees from perigee in the direction of motion of the spacecraft (Figure 4). The rotation velocity of the spacecraft will be up to 0.6 degrees per second. The change in the hexapod's position will be less than 15 degrees for forty minutes of the experiment. The time of the selection is

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the day of the summer (winter) solstice ± 5 days (eleven days). The duration of the experiment will be about 40 minutes for the lighted revolution (1 time per day).

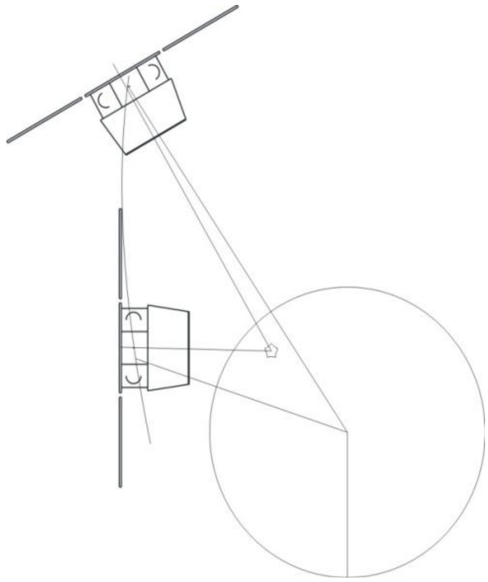

Figure 4: The precise pointing mode of the DSSPS spacecraft at the point of energy reception in the range of angles of 225-255 degrees from perigee

To determine the best lighting modes, the start time must be specified by ballistics scientists (summer solstice or winter) with a specific time for conducting energy transfer experiments in terms of minimizing control actions, which reduces the magnitude of the dynamic perturbations, which will positively affect the accuracy of the pointing [13-15].
The determination of the power density coming to the Earth and the diameter of the beam of the transmitter, the permissible error in the orientation of the spacecraft are presented in Tables 2-3.

Table 2: Determination of the power density coming to The Earth and the diameter of the transmitter beam

Power source,W	**Altitude, km**	**D beam on ground, m**	**Beam area,m^2**	**Density, W/m^2**
100000				
	16500	45,83333	1649,045	60,6411539
	16000	44,44444	1550,617	64,4904459
	15500	43,05556	1455,218	68,7182274
	15000	41,66667	1362,847	73,3757962
	14500	40,27778	1273,505	78,5234442
	14000	38,88889	1187,191	84,2324191
	13500	37,5	1103,906	90,5874027
	13000	36,11111	1023,65	97,6896695
	12500	34,72222	946,4217	105,661146
	12000	33,33333	872,2222	114,649682
	11500	31,94444	801,0513	124,835948
	11000	30,55556	732,909	136,442596
	10500	29,16667	667,7951	149,746523
	10000	27,77778	605,7099	165,095541

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9500	26,38889	546,6532	182,931348
9000	25	490,625	203,821656
8500	23,61111	437,6254	228,50594

Table 3: Determination of the permissible error of the spacecraft orientation

Error	**0,1 altitude from**	**rad**		**"**	**Accuracy estimation of pointing velocity**	**Angular velocity**	**Degree/sec**
	16500	$3,0303\times10^{-6}$	$3,0303\times10^{-6}$	1,250087	3,019899	0,000183024	0,010486
	16000	0,000003125	$3,125\times10^{-6}$	1,289152	3,087395	0,000192962	0,011056
	15500	$3,22581\times10^{-6}$	$3,22581\times10^{-6}$	1,330738	3,157977	0,00020374	0,011673
	15000	$3,33333\times10^{-6}$	$3,33333\times10^{-6}$	1,375095	3,231861	0,000215457	0,012345
	14500	$3,44828\times10^{-6}$	$3,44828\times10^{-6}$	1,422513	3,309286	0,000228227	0,013076
	14000	$3,57143\times10^{-6}$	$3,57143\times10^{-6}$	1,473317	3,390512	0,000242179	0,013876
	13500	$3,7037\times10^{-6}$	$3,7037\times10^{-6}$	1,527884	3,475825	0,000257468	0,014752
	13000	$3,84615\times10^{-6}$	$3,84615\times10^{-6}$	1,586649	3,565542	0,000274272	0,015715
	12500	0,000004	4×10^{-6}	1,650115	3,660013	0,000292801	0,016776
	12000	$4,16667\times10^{-6}$	$4,16667\times10^{-6}$	1,718869	3,759627	0,000313302	0,017951
	11500	$4,34783\times10^{-6}$	$4,34783\times10^{-6}$	1,793603	3,864815	0,000336071	0,019255
	11000	$4,54545\times10^{-6}$	$4,54545\times10^{-6}$	1,87513	3,976058	0,00036146	0,02071
	10500	$4,7619\times10^{-6}$	$4,7619\times10^{-6}$	1,964422	4,093895	0,000389895	0,022339
	10000	0,000005	5×10^{-6}	2,062643	4,21893	0,000421893	0,024173
	9500	$5,26316\times10^{-6}$	$5,26316\times10^{-6}$	2,171203	4,351844	0,000458089	0,026246
	9000	$5,55556\times10^{-6}$	$5,55556\times10^{-6}$	2,291826	4,493404	0,000499267	0,028606
	8350	$5,98802\times10^{-6}$	$5,98802\times10^{-6}$	2,470231	4,691808	0,000561893	0,032194

Typical Injection for launch vehicle (LV) of Russian development (with the help of the Fregat upper stage) consists of the following sections [16]:

- insertion into a datum orbit (carried out by the launch vehicle);
- flight on the datum orbit as part of the space head part (SHP);
- Injection to an intermediate orbit;
- flight in an intermediate orbit;
- injection to the transition orbit;
- flight along the transition orbit;
- injection to the final orbit;
- flight along the final orbit (separation and burial / inundation of the upper stage "Fregat").

The datum orbit is formed as a result of three stages of the carrier rocket - LV (direct excretion) [13, 16-19]. Transient and target orbits are formed as a result of a number of inclusions of the mid-flight engine of the accelerating block (deducing with delta v maneuver) (Fig. 5).

The SHP is separated from the LV. After separation, the "Fregat" upper stage launches the SHP into a working orbit of the "Molniya" type (12 hours) with an inclination of 62.8 degrees. After the end of the Injection phase, DSSPS is separated from the booster, and booster - is flooded.

After separation, DSSPS receives perturbations from the separation system. After some time, pyroclutrons are triggered on the DSSPS spacecraft, and solar batteries are being opened. A spacecraft with DSSPS with deployed solar batteries (SB) produces the solar orientation of the spacecraft, using a solar sensor and flywheel motors. After orientation of the solar system in the sun with the help of solar sensors described below, the power systems of the DSSPS spacecraft are charged, charging the AB and searching for ground communication stations.

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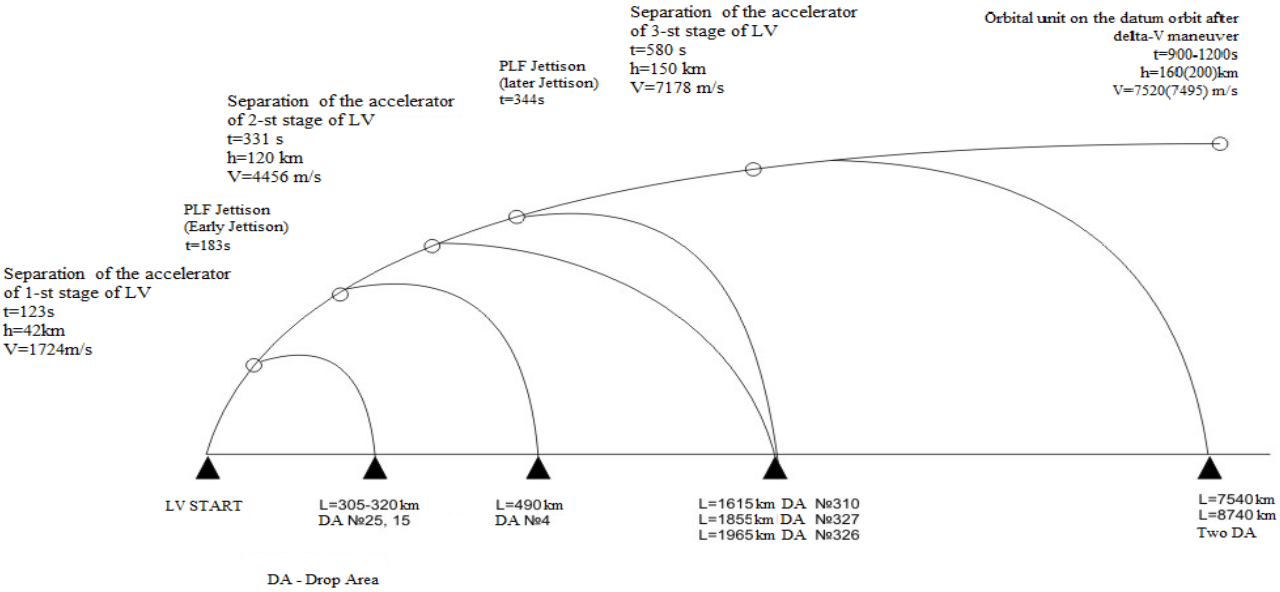
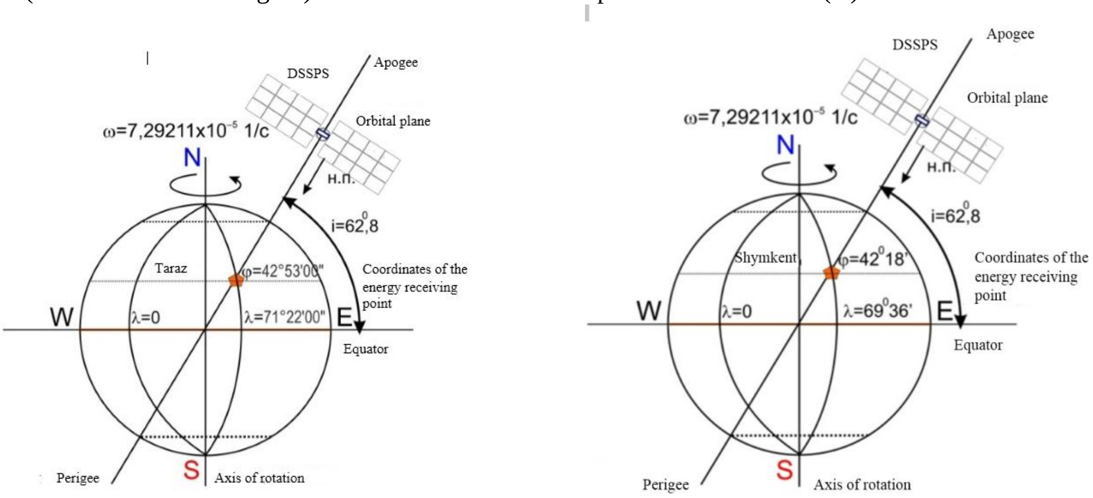


Figure 5: Injection scenario of the DSSPS spacecraft

As ground stations it is planned to use sparsely populated territories of Kazakhstan in the vicinity of cities (Figure 6):
- Taraz (Zhambyl region) - Coordinates 42 ° 53'00 "s. W. 71 ° 22'00 "in. (G)
- Shymkent (South-Kazakhstan region) - Coordinates 42 ° 18'00 "p. W. 69 ° 36'00 "in. (G)



a) Taraz *b) Shymkent*

]

Figure 6: The orbital plane crosses the energy receiving point

Such coordinates of the location of the energy reception point suggest the possibility in the more northerly zone of Kazakhstan along the route of the DSSPS spacecraft station to place active low-power laser targets to clarify the position of the spacecraft relative to the point of energy reception. The implementation of the mode of preliminary orientation will significantly increase the accuracy of the orientation of the DSSPS spacecraft and reduce perturbations during the experiment.

At the moment, the exhibition and dynamic models of the DSSPS are carried out. The exhibition model of the SC DSSPS - is a small copy of the DSSPS spacecraft, where the main constituent elements of the spacecraft are located. The layout was made in non-copy scale with the purpose of the best representation in the exposition from the viewpoint of the spectator's perception of the scale and functioning at Expo-2017, which is planned to be held in the city of Astana.

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For the dynamic layout, the mass inertial characteristics of the satellite platform (in the scheme - the platform and all incoming) are adopted similarly to the platform "Navigator". The weight of the platform is 740 kg.
The determining inertial characteristics - solar panels are accepted with an area of 141 m^2 of 24 elements 2.1 m by 2.8 m each. SB Panels constructively made of composite materials using modern technologies. The total weight will be ... 546 kg.
The energy transfer system - the payload is schematized to a mass imitator - a disk simulating the mass of a telescope with a transmitter, the mass located on a hexapod is 143 kg.
The difference from the exhibition layout - in the absence of a dynamic simulator, the orientation system of the solar battery. The orientation of the DSSPS spacecraft is made by rotating the spacecraft body with predetermined lens retargeting of the transmitter. The re-targeting of the optical axis of the transmitting device - telescope during the energy transfer experiment is carried out beforehand in the range up to 3 degrees from the direction to the Earth (the receiving site platform.
The dynamic model is used to numerically simulate the operations performed during the energy transfer experiment at the receiving point in order to assess the possibility of pointing the large-size DSSPS spacecraft structure by means of orientation and stabilization located on a navigator-type platform [20-23]. In other words, confirmation of the possibility of exact orientation of the DSSPS spacecraft axis to the required point of guidance by the currently available technical means (Figure 7).

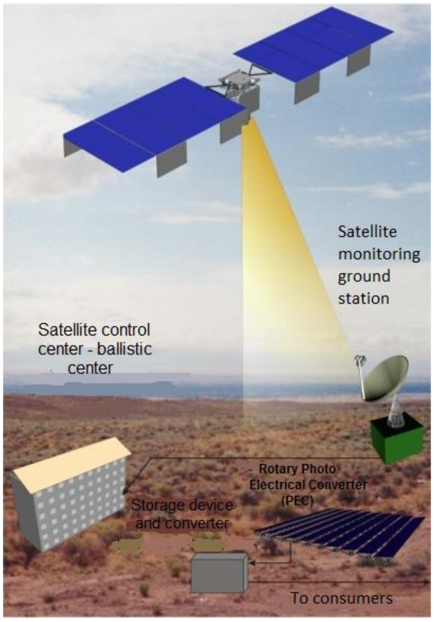

Figure 7: Scheme of the experiment

The receiving station is a field of 100x100 element with the ability to track a specific point of the celestial sphere (ie, the reception surface should be oriented perpendicularly to the incident energy beam) in order to improve the efficiency of the receivers. Elemental organization of receiving elements will significantly reduce wind loads on elements and reduce the cost of construction.
For the orientation of the DSSPS spacecraft, orientation sensors, actuators and an on-board computer will be required to set the sequence of actions and their priority in real time. The total orientation error will be composed of all the errors of the incoming elements in the chain.

3.2 Equations of the perturbed motion of spacecraft

The spacecraft is considered to consist of a body (the basic module "Navigator"), considered as a rigid body, and elastic oscillators (solar battery (SB) panels in the open position). A laser module (radiator) is attached to the body of the spacecraft by means of a six-link hexapod adaptor, designed to transmit a narrow beam of radiation to the Earth. The natural oscillation frequencies of the radiator installation are higher than 15 Hz, so in the low-frequency dynamic circuit, we consider it to be rigid.
We consider the following coordinate systems. The origin of the basic construction coordinate system (BCCS) lies in the docking interface of the base module (BM) "Navigator" with the adapter. The *X*-axis of the BCCS is the longitudinal axis of the symmetry of the SC, is directed toward the telescope and coincides with its optical axis. The

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Y and *Z* axes of the BCCS lie in the docking interface, along the *Y* axis, the wings of the SB are opened. The axes of the bound coordinate system are parallel to the BCCS axes, its origin is in the center of mass of the spacecraft.
SB panels are located symmetrically relative to the body, the plane of the panels is parallel to the plane *OYZ* of the bound coordinate system.
Table 4 shows the preliminary mass summary of spacecraft. The moments of inertia of the elements (components) of the spacecraft are calculated with respect to the center of mass of the corresponding component in the axes which are parallel to the BCCS.

Table 4: Mass summary of spacecraft

Name	***m*, kg**	I_{xx}, kg·m^2	I_{yy}, kg·m^2	I_{zz}, kg·m^2	I_{xy}, kg·m^2	I_{xz}, kg·m^2	I_{yz}, kg·m^2
BM “Navigator”	740	671	446	388	-2	-14	18
Telescope with the adaptor	143	56	35	35	–	–	–
SB	546	33529	1942	31587	–	–	–
Fuel	200	172	95	95	–	–	–

Table 5 shows the mass-inertial characteristics for a spacecraft with deployed panels of solar batteries.

Table 5: Mass-inertial characteristics of spacecraft in working position

***m*, kg**	***x*, m**	***y*, m**	***z*, m**	I_{xx}, kg·m^2	I_{yy}, kg·m^2	I_{zz}, kg·m^2	I_{xy}, kg·m^2	I_{xz}, kg·m^2	I_{yz}, kg·m^2
1629	0.346	0	0	34438	2913	32500	-2	-14	18
1429	0.367	0	0	34267	2821	32408	-2	-14	18

Designations: *m* - product mass; *x, y, z* are the coordinates of the center of mass in BCCS; I_{xx}, I_{yy}, I_{zz}, I_{xy}, I_{xz}, $I_{yz}$are the axial and centrifugal moments of inertia of the product in the bound coordinate system. The design tolerances for the variation of the axial and centrifugal moments of inertia range from plus 15% to minus 15% of the nominal values, and from zero to minus 15% for the mass.
The table shows:
(1) spacecraft in working position, completely fueled with a mass of 200 kg;
(2) spacecraft in working position without fuel.

3.3 Equations of the spacecraft motion relative to the reference trajectory

The dynamic scheme of spacecraft is a system of differential equations describing the spatial motion of product taking into account the elastic oscillations of low-frequency oscillators.
The perturbed motion of the spacecraft is considered in a bound coordinate system ($O_{ob}X_{ob}Y_{ob}Z_{ob}$) with the origin at the centre of mass (c.m.) of the spacecraft, the direction of its axes is shown in Fig. 8.

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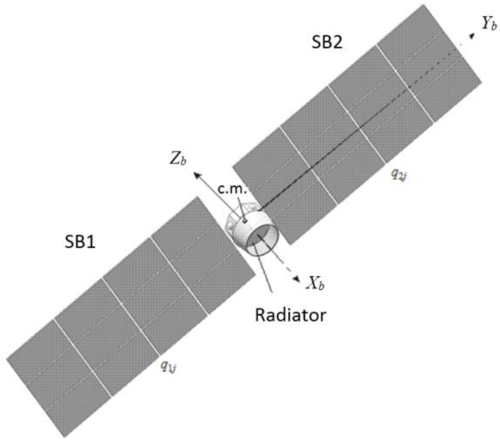

Figure 8: The coordinate systems and oscillators of the spacecraft under consideration

In the low-frequency dynamic scheme, the product is considered to consist of a shell, considered as an absolutely rigid body, and elastic oscillators: two Wing SBs (+*Y* and -*Y*). The plane of the deployed panels of the SB is parallel to the plane *OYZ* of the BCCS of the product. Let's designate: SB1 - the wing directed towards -*Y*, SB2 - aside + *Y*. Linear differential equations of motion of spacecraft in a bound coordinate system, taking into account the influence of oscillations of elastic elements and external disturbing influences, have the form [18, 24]:

Spacecraft body

$$m\dot{V}_x + \sum_{i=1}^{2}\sum_{j=1}^{n_i} a_{xij}\ddot{q}_{ij} = R_x + F_{xn}; \tag{1}$$

$$m\dot{V}_y + \sum_{i=1}^{2}\sum_{j=1}^{n_i} a_{yij}\ddot{q}_{ij} = R_y + F_{yn}; \tag{2}$$

$$m\dot{V}_z + \sum_{i=1}^{2}\sum_{j=1}^{n_i} a_{zij}\ddot{q}_{ij} = R_z + F_{zn}; \tag{3}$$

$$\begin{aligned} &I_{xx}\dot{\omega}_x - I_{xy}\dot{\omega}_y - I_{xz}\dot{\omega}_z + (I_{zz} - I_{yy})\omega_z\omega_y + I_{yz}(\omega_z^2 - \omega_y^2) + I_{xy}\omega_x\omega_z - I_{zx}\omega_x\omega_y + \\ &+ \dot{H}_x + H_z\omega_y - H_y\omega_z + \sum_{i=1}^{2}\sum_{j=1}^{n_i} b_{xij}\ddot{q}_{ij} = M_x; \end{aligned} \tag{4}$$

$$\begin{aligned} &I_{yy}\dot{\omega}_y - I_{xy}\dot{\omega}_x - I_{yz}\dot{\omega}_z + (I_{xx} - I_{zz})\omega_x\omega_z + I_{xz}(\omega_x^2 - \omega_z^2) - I_{xy}\omega_y\omega_z + I_{zy}\omega_x\omega_y + \\ &+ \dot{H}_y + H_x\omega_z - H_z\omega_x + \sum_{i=1}^{2}\sum_{j=1}^{n_i} b_{yij}\ddot{q}_{ij} = M_y + M_{yn}; \end{aligned} \tag{5}$$

$$\begin{aligned} &I_{zz}\dot{\omega}_z - I_{xz}\dot{\omega}_x - I_{zy}\dot{\omega}_y + (I_{yy} - I_{xx})\omega_x\omega_y + I_{xy}(\omega_y^2 - \omega_x^2) + I_{xz}\omega_y\omega_z - I_{yz}\omega_x\omega_z + \\ &+ \dot{H}_z + H_y\omega_x - H_x\omega_y + \sum_{i=1}^{2}\sum_{j=1}^{n_i} b_{zij}\ddot{q}_{ij} = M_z + M_{zn} \end{aligned} \tag{6}$$

Panel SB1

$$A(\ddot{q}_{1j} + \varepsilon_{1j}\dot{q}_{ij} + \omega_{1j}^2 q_{4j}) + b_{x1j}\dot{\omega}_x + b_{y1j}\dot{\omega}_y + b_{z1j}\dot{\omega}_z + a_{x1j}\dot{V}_x + a_{y1j}\dot{V}_y + a_{z1j}\dot{V}_z = 0, \tag{7}$$

$$j = 1,\dots, n_1; \tag{8}$$

Panel SB2

$$A(\ddot{q}_{2j} + \varepsilon_{2j}\dot{q}_{ij} + \omega_{2j}^2 q_{4j}) + b_{x2j}\dot{\omega}_x + b_{y2j}\dot{\omega}_y + b_{z2j}\dot{\omega}_z + a_{x2j}\dot{V}_x + a_{y2j}\dot{V}_y + a_{z2j}\dot{V}_z = 0, \tag{9}$$

$$j = 1,\dots,n_2. \tag{10}$$

In the equations of motion the following notations are adopted:
m – the mass of the spacecraft;
I_{xx}, I_{yy}, I_{zz} – the central axial moments of the spacecraft's inertia;

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I_{xy}, I_{xz}, I_{yz} – centrifugal axial moments of the spacecraft's inertia;
V_x, V_y, V_z – projection of the linear velocity of the center mass on the spacecraft's body axes;
$\omega_x, \omega_y, \omega_z$ – projections of the spacecraft absolute angular speed vector on the axis of the bound coordinate system
H_x, H_y, H_z – kinetic moments of engine-flywheels;
R_x, R_y, R_z – projections of external and internal disturbing and control forces on the body axes of the spacecraft;
M_x, M_y, M_z – projections of external and internal disturbing and control moments on the body axes of spacecraft;
F_{xn}, F_{yn}, F_{zn} –projections of forces from the hexapod drives on the axis of the bound coordinate system, H - at the current stage of development, we are considered equal to zero;
M_{yn}, M_{zn} – projections of the moments from the hexapod drives on the axis of the bound coordinate system;
q_{ij} – a generalized coordinate characterizing the elastic displacements of the i-th oscillator (i = 1, ..., 2) from its j-th tone of oscillations (j = 1, ..., n_i);
n_i is the number of considered tones of elastic oscillations for the i-th oscillator, n1 = n2 = 8;
a_{vij} - the coefficient of the attached mass, which determines the influence of the j-th elastic tone of the oscillations of the i-th oscillator on the translational motion of the center of mass of object in the direction of the axis * (* = *x, y, z*) and vice versa;
b_{vij} – the coefficient of the attached moment, which determines the influence of the j-th elastic tone of the oscillations of the i-th oscillator on the rotational motion of the object relative to the axis * (* = *x, y, z*) and vice versa;
ε_{ij} – coefficient of damping of the j-th elastic tone of oscillations of the i-th oscillator;
ω_{ij}^2 – coefficient of stiffness of the oscillator, equal to the value of the square of the circular frequency of oscillations of the j-th elastic tone of oscillations of the i-th oscillator.

3.4 Models of mechanical influences from the radiator drive

The radiator is attached to the body of the spacecraft by means of a six-link manipulator adaptor, the length of its legs can be changed and, thereby, the orientation of the radiator's mirror is changed. Each of the six independent legs of a hexapod consists of two rods and an active translational kinematic pair (drive). The radiator case and hexapod rods are considered as rigid.
The origin of the radiator co-ordinate system (RCS) *O'X'Y'Z'* lies in the plane formed by the three hinges of the attaching of the adaptor to the radiator case. The *OX'* axis coincides with the optical axis of the radiator, the axes *O'Y'* and *O'Z'* are parallel to the axes *OY* and *OZ* of the BCCS (Figure 9).
In our case, the manipulator mechanism must provide two degrees of freedom of the mirror - independent rotations of φ_{yr} and φ_{zr} relative to the axes *O'Y'* and *O'Z'* of the radiator's coordinate system. In this case, we can assume that the drives of the adjacent hexapod rods operate synchronously (that is, they create the same forces in the same direction).

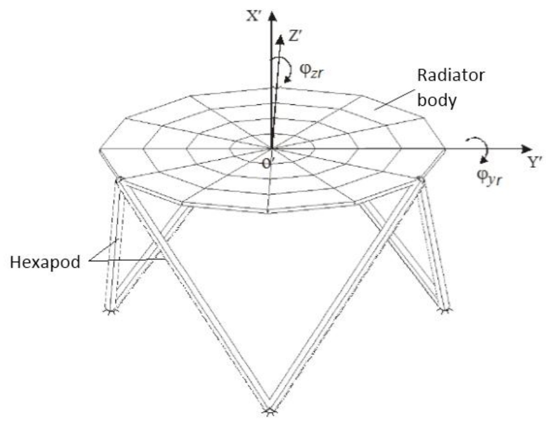

Figure 9: The coordinate system and the angles of rotation of the radiator

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In the motion equations of the spacecraft body, the projections of the moments M_{yn} and M_{zn} acting on the construction of the spacecraft from the operating drives of the manipulator are calculated by the formulas [25]:

$$M_{yn} = J_{yи}\dot{\omega}_{yи}; \tag{11}$$

$$M_{zn} = J_{zи}\dot{\omega}_{zи}, \tag{12}$$

where $J_{yr} = J_{zr}$ = 20 kg·m^2 are the moments of inertia of the radiator relative to the axes of the radiator co-ordinate system (RCS).
The functional dependences of the angular velocities of the radiator rotation relative to the SRS axes ω_{yr} and ω_{zr} on time will be determined after obtaining the cyclograms of the operation of the drives in the course of further constructive and experimental testing.

3.5 Calculation of the coefficients of the perturbed motion equations

With the use of the MSC.visual NASTRAN software complex, the coefficients of the perturbed motion equations were calculated for two flight phases: initial (spacecraft fully filled) and final (spacecraft without fuel).Calculations have shown that the coefficients vary slightly during the active existence of the spacecraft.
Table 6 shows the coefficients of the perturbed motion equations for all flight stages of the spacecraft.

Table 6: Coefficients of the equations of disturbed motion

Oscillator number (i)	**Tone of oscillation (j)**	ω_v^2, *1/s²*	ε_{ij}, *1/s*	a_{xij}, *kg*	a_{yij}, *kg*	a_{zij}, *kg*	b_{xij}, *kg·m*	b_{yij}, *kg·m*	b_{zij}, *kg·m*
1	2	3	4	5	6	7	8	9	10
1	1	0.28	0.008	12.96	0.00	0.00	0.00	0.00	122.39
1	2	0.42	0.010	0.00	0.00	0.00	0.00	27.15	-0.01
1	3	4.58	0.034	0.00	0.00	0.00	0.00	10.60	0.00
1	4	9.81	0.050	-7.50	0.00	0.00	0.00	0.00	-25.51
1	5	17.21	0.066	0.00	0.00	0.00	0.00	-7.14	0.00
1	6	22.35	0.075	0.00	0.00	-14.10	129.04	-5.18	0.00
1	7	42.83	0.104	0.00	0.00	0.00	0.00	4.49	0.00
1	8	52.14	0.115	-4.54	0.00	0.00	0.00	0.00	-10.33
2	1	0.28	0.008	-12.96	0.00	0.00	0.00	0.00	122.37
2	2	0.42	0.010	0.00	0.00	0.00	0.00	27.15	-0.01
2	3	4.58	0.034	0.00	0.00	0.00	0.00	-10.61	0.00
2	4	9.81	0.050	7.50	0.00	0.00	0.00	0.00	-25.52
2	5	17.21	0.066	0.00	0.00	0.00	0.00	7.15	0.00
2							-		
	6	22.35	0.075	0.00	0.00	-14.10	129.07	-5.17	0.00
2	7	42.83	0.104	0.00	0.00	0.00	0.00	-4.48	0.00
2	8	52.14	0.115	4.54	0.00	0.00	0.00	0.00	-10.32

3.6 Acceptable deviations of the coefficients of the dynamic scheme

Linear functions were used to approximate the boundaries of acceptable deviations. The upper limits of the acceptable deviations of the inertial coupling coefficients are uniquely related to the current values of acceptable deviations of the mass δ_m and the moments of inertia δ_j by the relations

$$\delta_a^{up} = (0.53\delta_m + 0.01)\times 100\,\%; \qquad \delta_b^{up} = (0.53\delta_J + 0.01)\times 100\,\%, \tag{13}$$

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where

$$\delta_m = \frac{m}{(m)_{nom}} - 1; \qquad \delta_J = \frac{J_v}{(J_v)_{nom}} - 1;$$

$v = x, y, z;$ the index "nom" is used for nominal values of mass and moments of inertia.
The lower limits of the acceptable deviations of the inertial coupling coefficients are determined by the relations:

$$\delta_a^{\text{low}} = \delta_a^{\text{up}} - 30\%; \qquad \delta_b^{\text{low}} = \delta_b^{\text{up}} - 30\%. \tag{14}$$

The shaded area in Fig. 10 is an illustration of the obtained dependences of the acceptable deviations of the inertial coupling coefficients from the acceptable deviations of the mass-inertial characteristics.

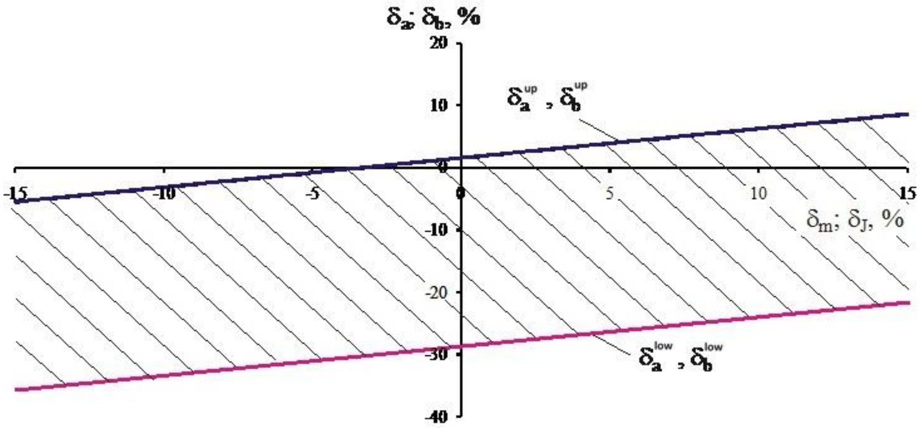

Figure 10: Limits illustration of the permissible deviation of the coefficients

The acceptable deviations of the squares of the circular frequencies are ± 50%. The acceptable deviation of the damping factors is ±50%.

3.7 The spacecraft attitude control system

The spacecraft attitude control system can be conditionally divided into two systems: a stabilization system and an orientation system that perform the following functions.
Orientation system:

- carries out the information processing of star instruments to determine the angular position of the spacecraft relative to the inertial space;
- implements the construction of the gyroinertial coordinate system according to the information of the Strapdown Inertial Unit (SIU), hereinafter referred to simply as the SIU coordinate system;
- determines the mismatch between the SIU coordinate system and the given coordinate system in the inertial space;
- carries the reduction of the SIU coordinate system to the given coordinate system in the inertial space;
- performs determination and compensation of the drift of the SIU coordinate system, caused by systematic errors of the zero signal of integrating gyroscopes.

Stabilization system:

- carries out the formation of control signals given to the actuators, thus it implements the chosen control law;
- determines the constant perturbing acceleration from the external and internal disturbing torques acting on the spacecraft, to compensate them by modifying the chosen control law.

The rationale for the separation of orientation control system into two subsystems: an orientation system and a stabilization system, can be that these systems operate with different time cycles.
The orientation system works with a tact (T_o) for obtaining information from the star sensors.
The stabilization system works with the tact of the on-board computer system (τ), which coincides with the tact of receiving the information of the SIU and the tact of issuing the control signal to the system of actuators.

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Conclusions

Based on the studies carried out on the proposed implementation of the instrument composition and the algorithms of the orientation control system, the following preliminary conclusions can be drawn:
When the spacecraft is flying along the Earth's orbit, the accuracy of pointing of the device for transferring energy to a given region of the Earth's surface and keeping it in this position is no more than 1.5 angular seconds.
If necessary, for a guaranteed increase in this accuracy, it is most expedient to switch to more precise measuring instruments, first of all - to the SIU.

Recommendations:

In the future, at the subsequent stages of the development of this project, it is necessary to work out the following issues:
The use of a power gyro instrument as the actuators of a control system on condition of a possibility of their application in the precision mode.
Simulation of the spacecraft ballistics not in the central field of the Earth, but with the application of a refined model of gravitational influence and taking into account ballistic errors in determining the position of spacecraft in orbit and their on-board implementation.
To consider the implementation of the spacecraft orientation program for the entire time of the spacecraft functioning.
To consider the alignment scheme of the position of the device for transferring energy relative to the measuring systems of sensors.

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