TRAJECTORY DESIGN FOR EXPERIMENTAL MISSION TO SUN-EARTH L_1 AND L_2 POINTS USING SEP

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Abstract

A mission to the Sun-Earth L_1 and L_2 points using solar electric propulsion (SEP) is considered. The basic mission steps are the following: a spin-stabilized spacecraft is assembled aboard space station and separated from it; then the spacecraft is accelerated in a spiral trajectory, transferred to L_1, and near L_1 inserted into a halo orbit, all by means of SEP. A possibility of its subsequent transfer to L_2 is also considered in frame of the mission. Description of the spacecraft and results of the trajectory design for the mission are given. It is shown that all the mission goals can be reached using the spin-stabilized spacecraft with the thrust direction orthogonal to the direction to the Sun.

Key words: Solar electric propulsion, Libration point, Halo orbit.

Introduction

At present a relatively light electrically propelled spacecraft for flight to the Sun-Earth L_1 libration point is being designed by Russian space organizations[1]. All operations in space such as launch from LEO, flight to L_1, and insertion into the halo orbit are to be fulfilled by means of the solar electric propulsion (SEP). This means that the spacecraft acceleration in the Earth vicinity necessarily will be performed in a spiral trajectory. The mission is experimental and has a few specific features, the main ones are the following:

1) The spacecraft components are to be delivered on board of the international space station[*] by the Progress cargo spacecraft. After its assembling on the station (partly inside and mostly outside it) it is to be detached and to fly using its SEP to the libration point L_1.

2) The spacecraft is spin-stabilized with immovable solar arrays and thrusters what simplifies the spacecraft construction and control and lowers the mission cost. On the other hand this puts strict constraints on the thrust direction because the solar arrays must be always directed to the Sun in order to supply the electric propulsion with maximum power. A unique spacecraft concept reducing the constraints has been proposed for the mission. However the thrust direction is still constrained what limits the maneuverability of the spacecraft.

The main goals of the mission are prediction of the magnetic storms and the solar wind exploration. However even more important mission purpose is testing the new technologies such as assembling the spacecraft on and its launch from the space station, the electric propulsion, operations and control etc.

A subsequent transfer of the spacecraft from L_1 to L_2 for the Earth magnetosphere tail exploration is also considered for the mission. According to one of the possible mission options the spacecraft can deliver and leave a small satellite in the L_1 halo orbit and then move to an L_2 one.

A brief description of the spacecraft and basic results of the trajectory design for the mission are given below.

The spacecraft concept

Design of the spacecraft equipped with SEP is subject to the following requirements:

- direction of the continuous low thrust must be close to the spacecraft velocity vector for a long time of the thrust run to provide maximum efficiency of the thrust;

- the solar panels must be directed to the Sun for all the time of the SEP run to provide the power-consuming thrusters with maximum electric power.

This requirements often contradict each other; especially this is true for the spiral orbit of the spacecraft where it performs hundreds of orbits and its thrust must follow the velocity vector in each of them. This would lead to the complicated both the spacecraft construction and control.

A simple and elegant solution of the problem has been proposed for the considered mission. The spin-stabilized spacecraft reminds a bike's wheel with the spacecraft body in the middle and the solar arrays along the rim (see Fig. 1).

[*] It also can be Mir station if its life in the orbit will be prolonged.

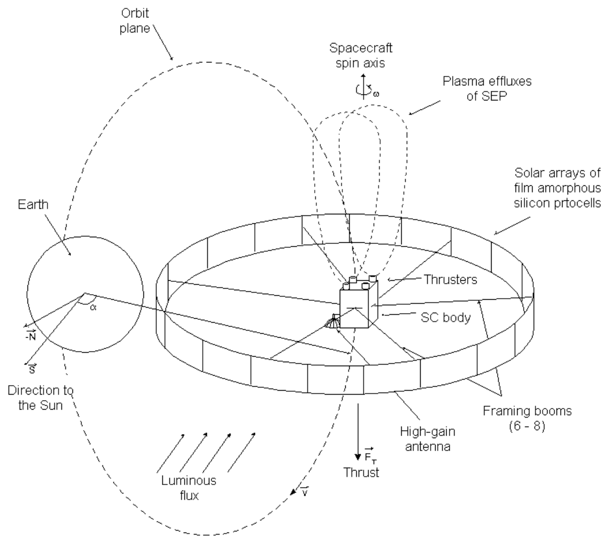

Figure 1: Overall view of the spacecraft

The arrays form a cylinder surface of 18 m diameter and 2 m height and look outside. Thus the total area of the arrays is about 110 m^2; however the effective area is within 36 m^2 (if the spacecraft spin axis is orthogonal to the Sun direction). The photocells cover 85% of the arrays so their effective area is 30 m^2. It is proposed to use a thin film amorphous silicon photocells for the solar arrays. This new technology will provide light, cheap, and sufficiently effective solar arrays which can supply the electric propulsion with power of about 3 kW. So large arrays with mass of about 60 kg including supporting structure are to be deployed and mounted by cosmonauts during their extravehicular operations.

The spacecraft will have 8 thrusters D-38 of the TAL type designed in the Energiya Rocket and Space Corporation. Characteristics of one D-38 thruster are given in the Table 1.

Table 1: D-38 thruster parameters

Parameter name	Parameter value
Power, W	750
Specific impulse, s	2200
Efficiency (including losses in PPU)	0.5
Thrust force, N	0.035
Mass flow rate, kg/s	$1.6 \cdot 10^{-6}$
Resource, hours	3000
Propellant	xenon

Four of the thrusters will be installed on one side of the spacecraft and another four on the opposite one; their thrust will be directed along the spacecraft spin axis in two opposite directions. The four co-directed thrusters run simultaneously while other four are relaxing. Initial spin rate of the spacecraft is 1 revolution per minute. Its spin axis is to be orthogonal to the Sun direction (see Fig. 1).

Thus any thrust direction can be chosen in the plane orthogonal to the Sun direction. Dry mass of the propulsion system including the thrusters, propellant tanks, and control system is estimated as of 44 kg.

Attitude determination and control system includes solar sensors, Earth infrared ones and star ones. Magnetic coils are used as actuators for the orbit altitude less than 25000 km. For higher altitude cold gas jets are to be used.

Scientific instruments of total mass of 15 kg include the following: magnetometers unit, ion and electron spectrometer, plasma sensor, high energy particles spectrometer.

The wet initial mass of the spacecraft is estimated as of about 290 kg. This includes 85 kg of xenon what provides about 7.5 km/s of the spacecraft characteristic velocity.

Spacecraft spiral orbit

A typical space station orbital parameters were taken for the initial spacecraft orbit: the circular orbit of the 400-km altitude and 51.6-degree inclination.

After the separation from the space station and starting its acceleration from LEO by means of SEP the spacecraft is moved in an expanding spiral orbit. While the spacecraft jet acceleration is much lower than the gravitational one its osculating orbit remains very close to the circular one of the growing radius. Optimal thrust direction in this case is always very close to the one of the spacecraft orbital velocity. However this is impossible for the spacecraft described above (except two short arcs in each orbit when the spacecraft velocity is orthogonal to the Sun direction). The following strategy of the SEP control taking into account the spacecraft concept has been selected for the mission: the SEP runs along two 120-degree arcs, ±60 degree from the projection of the Sun direction onto the orbit plane; the four thrusters providing the proper thrust direction run in each of the arcs (see Fig. 2).

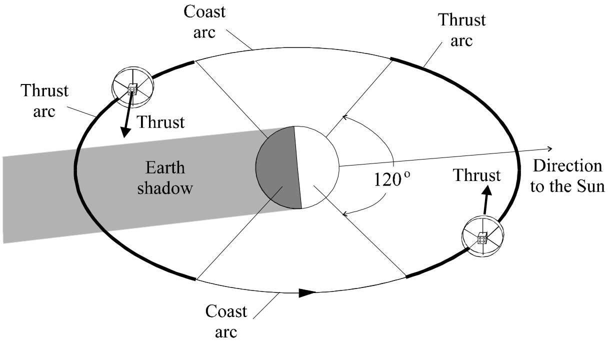

Figure 2: The SEP runs in the Earth vicinity

It is easy to obtain that the strategy means loss of 17 percent of the SEP effectiveness (and respectively higher propellant consumption) and longer with factor 1.7 time of flight comparing to the permanent tangential thrust. This is the payment for the simplified spacecraft and control.

The 120-degree arc has been chosen as a compromise: a shorter arc would provide higher effectiveness of the SEP but the flight time would increase; a longer thrust would lead to the less effectiveness of the propulsion.

However there is one problem: the thrust arc behind the Earth (with respect to the Sun) can be entirely or partly shadowed by the Earth for a long time (see Fig. 2). It is impossible to avoid the shadowing completely, the only way to diminish it is the appropriate selection of both the Sun position and the longitude of the ascending node of the spacecraft orbit at the launch time. Analysis shows that the spacecraft launch in June-July or December-January with the longitude of the ascending node of about 280 - 300 degrees minimizes the average shadowing down to 7.5 percent (i. e. in average about 7.5 percent of the whole thrust arc per one orbit are shadowed). However this optimal solution would lead to a high (higher than 50 degrees) final inclination of the spacecraft orbit to the ecliptic plane what is not good for the further insertion into the halo orbit. Therefore a compromise has been selected: launch in May or November with the longitude of the ascending node around 260 degrees. This gives the average shadowing of about 8.5 percent and the final inclination to the ecliptic plane of about 35 degrees (it is clear that the minimal possible inclination is about 28 degrees if the inclination to the equator is 51.6 degrees). The thrust arc shadowing and inclination to the ecliptic versus time are shown on Fig. 3; it is seen that one of the thrust arcs is completely shadowed at the beginning of flight. Radius of the spiral orbit versus time is shown on Fig. 4.

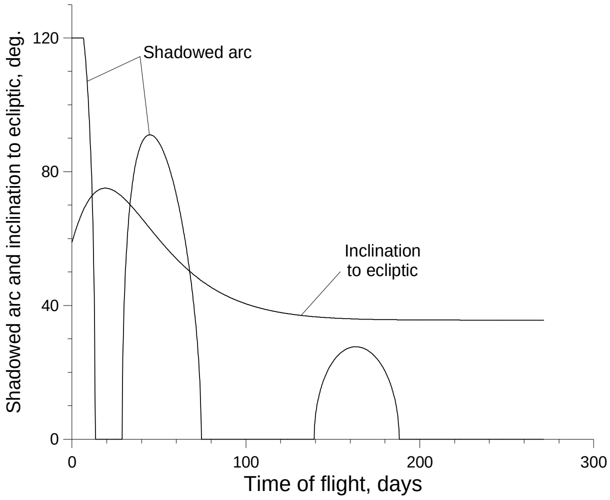

Figure 3: Shadowed arc and inclination to the ecliptic versus time

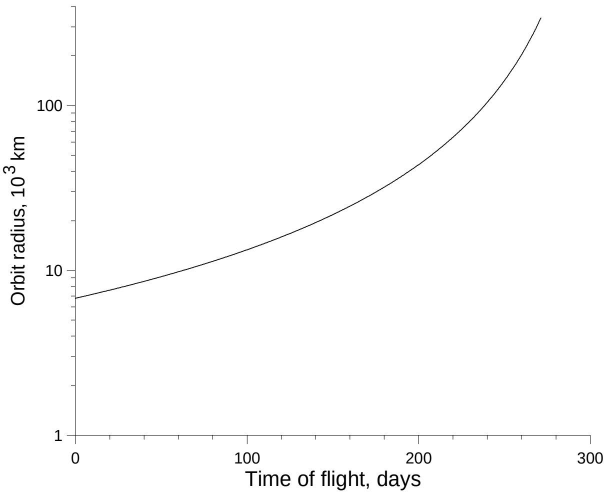

Figure 4: Orbit radius versus time

Note that the final inclination to the ecliptic is very sensitive to the initial longitude of the ascending node: 10-degree variation of the longitude changes the inclination in 3 degree. Since the ascending node precession is about 5 degrees per day for the space station that means that the launch window providing necessary inclination to the ecliptic is very narrow. Therefore the spacecraft must be separated in advance in order to start operations exactly at a given time.

The parameters of the spiral orbit in the Earth vicinity are given in the Table 2.

Table 2: Parameters of the spiral orbit

Parameter name	Parameter value
Time of flight, days	280
Number of orbits	1330
Consumed characteristic velocity, m/s	6850
Propellant consumption, kg	78.9
Spacecraft mass, kg	211.1

Flight to L_1 and insertion into halo orbit

Rather small halo orbit with the amplitude $A_y \approx 60$ thousand km has been selected for the mission. Fig. 5 gives two projections of the spacecraft trajectory: to the ecliptic plane (*xy*) and the orthogonal one (*xz*); the spiral shown on the figure starts at the radius of 50,000 km.

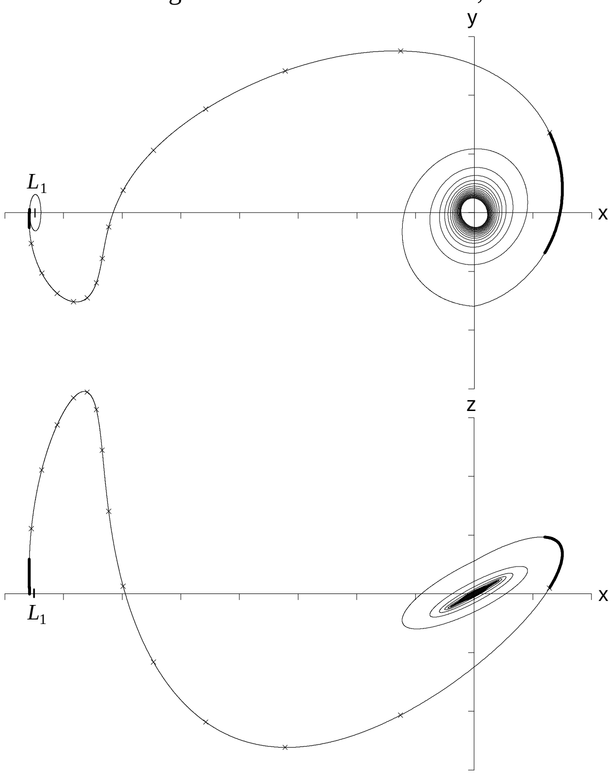

Figure 5: The spacecraft trajectory to L_1

The trajectory corresponds to the spacecraft launch in November; for the May launch the *xy* projection will not change and the *xz* one will be mirrored with respect to the *x* axis. The trajectory includes the spiral part, the flight to L_1, and the halo orbit. The bold arc at the end of the spiral orbit is the last thrust arc injecting the spacecraft into the transfer trajectory to L_1. This arc is shorter than the typical 120-degree arcs and asymmetric; this is to provide the necessary halo amplitude and the *z* component close to zero during the insertion into the halo. The thrust in the arc lasts 4.5 days and consumes 268 m/s of the spacecraft characteristic velocity (2.6 kg of xenon have been included in the spiral orbit propellant consumption). The bold part near L_1 shows the break maneuver inserting the spacecraft into the halo orbit. The crosses on the trajectory mark 10-day time intervals after the injection into the transfer trajectory, the axes ticks correspond to the distance 200 thousand km.

Parameters of the transfer to and insertion into the halo orbit are given in the Table 3.

Table 3: Parameters of the flight to halo orbit

Parameter name	Parameter value
Time of flight (after the spiral), days	140
Characteristic velocity of the insertion into halo, m/s	290
Propellant consumption, kg	2.8
Spacecraft mass in halo, kg	208.3
Amplitude A_y of the halo orbit, 10^3 km	62

As it can be seen on the Fig. 5 the planar halo orbit has been taken for the trajectory design. It is not necessary for the mission purposes, however the cost of this insertion is just a little higher than of one into a 3D halo with a reasonable A_z amplitude. So the cost, 290 m/s (see Table 3), is an upper limit for the insertion into a halo with 60-thousand-km A_y amplitude.

A tiny variation of the delta-V transferring the spacecraft to the libration point can dramatically change the halo orbit amplitude. The approximate dependencies are the following: the delta-V increment in 5 cm/s increases the A_y amplitude in 100 thousand km and reduces the characteristic velocity and the propellant consumption of the insertion into halo in 20 m/s and 0.2 kg respectively.

On the Moon gravity assist

The Moon gravity assist for the transfer to the L_1 has been analyzed only for the spacecraft with 3-axis stabilization. The main advantage of this maneuver is that it can put the spacecraft trajectory very close to the ecliptic plane and hence lower the delta-V of the insertion into the halo almost in 200 m/s (2 kg of xenon). However the Moon gravity assist may require waiting in a parking orbit for providing the Moon encounter conditions what can increase the total flight time in a few weeks.

Transfer from L_1 to L_2

At present this part of the mission can be considered rather as a mission extension, its profile is still completely uncertain. In particular the parameters of the L_2 halo orbit are not defined yet. Therefore different options have been considered for the trajectory design.

There is a huge manifold of possible transfers from the L_1 halo orbit to L_2 one. Even if the both halos are given the transfers differ by the number and location of the active maneuvers, their values, number of complete orbits around the Earth, transfer duration, use of the Moon gravity assist etc. At this phase of the mission design we limited ourselves by the planar transfer from the L_1 halo of the amplitude A_y = 60 thousand km to an L_2 one.

Zero complete orbits around the Earth. The transfer trajectory for this case is shown on Fig. 6. Three active maneuvers by means of the SEP are needed for this transfer; they are labeled as $\Delta v_{1,2,3}$ on the figure. The thrust is directed toward $+y$ axis for $\Delta v_{1,3}$ and $-y$ for Δv_2.

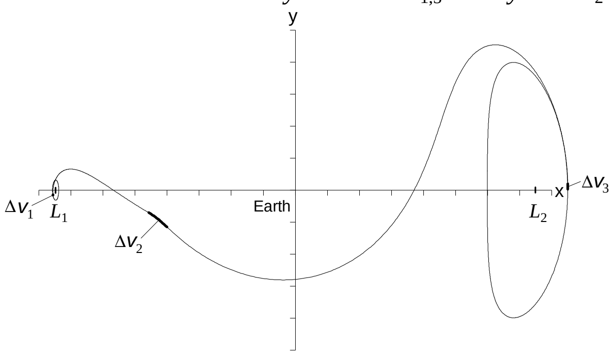

Figure 6: Transfer with zero complete orbits

The transfer characteristics are given in Table 4.

Table 4: Transfer with zero complete orbits

Parameter name	Parameter value
Consumed characteristic velocity, m/s	306
Δv_1	50
Δv_2	195.6
Δv_3	60.5
Time between Δv_1 and Δv_2, days	70
The transfer duration, days	181
Propellant consumption, kg	2.9
Final spacecraft mass, kg	205.4
A_y amplitude of the L_2 halo, thousand km	800

The duration of the transfer is relatively short in this case, just 6 months. However the available propellant allows only this large halo orbit around L_2.

One complete orbit around the Earth. Fig. 7 shows one of the possible transfers with A_y = 300 thousand km. For all three active maneuvers the thrust is directed toward $+y$ axis in this case.

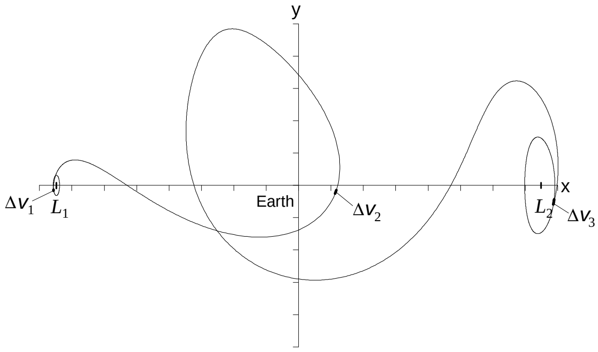

Figure 7: Transfer with one complete orbit

Parameters of the transfer are given in Table 5.

Table 5: Transfer with one complete orbit

Parameter name	Parameter value
Consumed characteristic velocity, m/s	224
Δv_1	65
Δv_2	18.1
Δv_3	141
Time between Δv_1 and Δv_2, days	82
The transfer duration, days	259
Propellant consumption, kg	2.2
Final spacecraft mass, kg	206.1
A_y amplitude of the L_2 halo, thousand km	300

This transfer has the longer duration (8.6 months) but can provide lower amplitude of the halo orbit for lower propellant consumption than the zero-orbit one. The available propellant could permit even lower amplitude than one indicated in the Table 5.

Two complete orbits around the Earth. An option for the transfer is shown on Fig. 8; here also the thrust is directed toward $+y$ axis for all maneuvers.

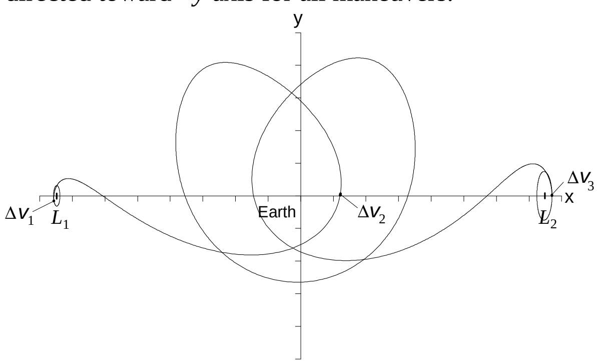

Figure 8: Transfer with two complete orbits

Table 5 gives the transfer parameters.

Table 5: Transfer with two complete orbits

Parameter name	Parameter value
Consumed characteristic velocity, m/s	70
Δv_1	35
Δv_2	1.6
Δv_3	33.2
Time between Δv_1 and Δv_2, days	70
The transfer duration, days	319
Propellant consumption, kg	0.7
Final spacecraft mass, kg	207.6
A_y amplitude of the L_2 halo, thousand km	150

This is the longest transfer (10.5 months) but it allows any amplitude of the L_2 halo orbit for a very low cost. Note that in the case of two complete orbits a two-impulse transfer is also possible. A symmetric two-impulse transfer is shown on Fig. 9; here $\Delta v_1 = \Delta v_2 =$ 43.3 m/s (0.8 kg of the propellant for both) and the transfer duration is 307 days.

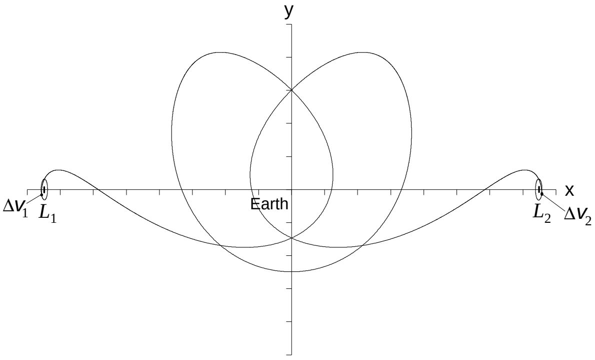

Figure 9: Symmetric two-impulse transfer

Common notes. In all three considered cases both the L_2 halo amplitude and the transfer duration can be varied by means of changing the active maneuvers values and positions. Lowering the amplitude in 30 thousand km costs about 10 m/s of the spasecraft characteristic velocity (~0.1 kg of xenon); lowering the transfer time in 10 days takes 5 - 7 m/s.

The Moon gravity assist can be easily performed in the considered planar (or near-planar) transfer. The phasing of the spacecraft trajectory necessary to provide the encounter with the Moon can be obtained by a very small variation of the launch maneuver from L_1 halo orbit. The gravity assist certainly could either lower the xenon consumption for the transfer or lower the L_2 halo amplitude or the transfer duration. However this maneuver has not been analyzed yet.

Conclusion

Table 6 summarizes characteristics of all the spacecraft movements; the duration of its stay in the L_1 halo orbit is excluded from the total flight duration because it is still undefined.

Table 6: Summary of the spacecraft transfers

Operation	Fflight time, months	Total Δv, km/s	Total xenon consum., kg	S/C mass, kg
Launch	0	0	0	290
Acceleration in the spiral orbit	9.3	6.85	78.9	211.1
Transfer to and insertion in L_1 halo	14.0	7.14	81.7	208.3
Transfer to and insertion in L_2 halo	20-24.5	7.21 - 7.45	82.4 - 84.6	205.4 - 207.6
Rest for the correction maneuvers	-	0.05 - 0.29	0.4 - 2.6	-

The run time of each thruster is within 2000 hours what is covered by the thruster resource time (see Table 1). Nevertheless an installation of a pair of the spare thrusters is also possible.

Thus the spacecraft concept accepted for the mission provides the fulfillment of all operations necessary for the transfer to the L_1 halo orbit and then to the L_2 one for reasonable time and propellant consumption. This is mainly due to the fact that the thrust orthogonal to the direction to the Sun is very effective for changing the orbital parameters in the libration points vicinity. Although this is also true for the planetary missions[2], so this concept can be applied for them as well.

References

[1] Experimental small spacecraft-2: Mission to the Sun-Earth libration point L_1. Internal report of IKI RAN No. 124/98, September 1998 (in Russian).

[2] Williams, S.N.; Coverstone-Caroll, V. Benefits of Solar Electric Propulsion for the Next Generation of Planetary Exploration Missions. *Journal of Astronaut. Sci.*, Vol. 45, No. 2, April-June 1997, pp. 143-159.