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Mission Design Problems for Spectrum-Roentgen-Gamma Project

Ravil Nazirov,[1] Natan Eismont,[1*] Vadim Arefiev[1],Fedor Korotkov[1],Andrei Pogodin[2], Andrei Tregubov[3], Alexey Ditrikh[3]

[1]Space Research Institute of the Russian Academy of Sciences, Russia,[2]Lavochkin Association, Russia, [3]Russian Space Corporation Energia

neismont@iki.rssi.ru

Spectrum-Roentgen-Gamma (SRG) project is intended for whole sky review in X and Gamma ray bands. Besides, some particular area of the sky and chosen sources of radiation are to be explored separately with extended times of observation. For this two telescopes are planned to be used mounted onboard spacecraft of the same name (SRG). The last is to be launched into vicinity of Solar-Terrestrial collinear libration point L2 in 2018. As launch vehicle Proton with Block DM upper stage is planned. The technology of sky review consists from operations aimed to scan the sky by telescope axes by rotation spacecraft around axis which roughly following Sun direction. Telemetry data are transmitted to the Earth ground stations through mean gain antenna. Its axis coincides with rotation axis what generates some constraints on mission design. Among them are allowed amplitude limits of near libration point orbit along directions orthogonal to the Sun-Earth line. In the list of technical constraints there is the mass of onboard propellant needed for orbital maneuvers in order to decrease amplitude of the operational trajectory. So the alternative approach is developed based on the use of upper stage delta-V possibilities in order to execute such maneuver before spacecraft separation, taking into account constraints on permissible time interval between the last two upper stage burns . It was shown that such method does work decreasing the amplitude of operational trajectory to the demanded level. Its effectiveness increases with time interval since start from initial parking orbit to the next burn of upper stage engine. In the paper the cost of such approach is presented in terms of demanded delta-V.

Key Words : Libration point, correction maneuver

1. Introduction

Spectrum-Roentgen-Gamma spacecraft is planned to be launched in March 2017 by Proton-M with upper stage Block DM. It is to be put onto orbit in proximity of L2 solar-terrestrial collinear libration point. The spacecraft is carrying two telescopes to fulfill measurements in roentgen band in order to investigate the whole sky sphere scanning it during 4 years by rotation around axis lying near ecliptic plane and roughly following Sun direction. It means that the sky sphere will be covered 8 times. After that the observations are be switch onto more careful studies of some chosen sources of roentgen radiation or preferable regions of the sky.

Scientific information from telescopes is recorded onboard spacecraft and transmitted to the ground stations during intervals of visibility of the spacecraft. The mean gain antenna mounted onboard spacecraft is no steerable. It poses restrictions on the allowed angle between antenna axis and direction to the station. Besides, the maximum daily visibility intervals from ground stations are to be reached. The other constraint is the upper limit on the angle between spacecraft axis coinciding with the antenna axis and direction towards the Sun.

To satisfy these requirements one needs to minimize the amplitude of the spacecraft orbit considered in solar-ecliptic coordinate system and to control the spacecraft rotation axis coinciding with antenna axis, keeping it somewhere in between directions towards Sun and towards Earth.

To decrease amplitude, delta-V is to be applied when spacecraft has reached L2 vicinity. But for our case it is planned to consume propellant stored onboard spacecraft only for the tasks of attitude control and orbital correction maneuvers. To overcome this obstacle the possibilities to use upper stage for lowering the value amplitude were explored.

Also the other alternative supposing gravity assist maneuver near Moon was studied.

For decreasing the project design risks it is required to investigate the impact of launch date shift (delay) onto key mission parameters, which include duration of visibility intervals.

2. Trajectory design and technical constraints

It is assumed that spacecraft will be delivered onto transfer orbit to L2 by launch vehicle Proton-M with Block DM upper stage in such a way which does not demand following use of spacecraft engine in nominal case. In other words after separation from upper stage the spacecraft is to reach the goal orbit and to start motion along this trajectory without any additional delta-V. Such approach is known as one impulse transfer. Obviously it is some ideal case. In real situation in order to reach the orbit in L2 vicinity one needs to execute some trajectory correction maneuvers. Usually the number of these maneuvers does not exceed 3-4, depending on accuracy of trajectory parameters determination and the errors of delta-V during correction maneuvers.

The problem of delta-V vector choice is very similar to the trajectory design. The essence of this problem is to understand of what trajectory parameters are to be reached. Known as a boundary-value problem, standard approach to the transfer trajectory parameters calculation for the simplest case assumes determination of three components of velocity vector at initial time which allows reaching required three coordinates at some final moment.

In our case the goal parameters of trajectory are not rigid fixed. Instead it is required that spacecraft after reaching L2 proximity would not leave it during several years. As proximity we assume to be the space around L2 point containing the points not further from L2 than 1.5 million km. For our project these parameters are to be optimized with the goal to reach the minimum value of the maximum distance of the spacecraft from L2 in its motion around L2 taking into account the requirement to have not less than 2 hours visibility from Bear Lakes ground station each day of the mission. Really for the mission two ground stations are planned to be used, besides the mentioned one the Ussuriysk station is supposed to be operational but as some backup option. Both ones are quite big: Bear Lakes has 60 meters diameter dish and Ussuriysk 70 meters. Requirements of so large size of antennas are caused by the fact that spacecraft is not equipped by high gain antenna.

Trajectory design includes launch scenario construction it has in our case some peculiarities generated by the history of the project. Initially as launch vehicle Soyz-2,1b has been chosen. But after final development of scientific instruments when their mass was determined more precisely, much more powerful launcher Zenit-2 was proposed produced by Ukraine company. Finally under pressure of not technical reasons Proton-M with upper block DM upper stage appointed for SRG spacecraft launch. Its possibilities to put spacecraft onto goal orbit by more than two times exceeds the required mass of payload. But there are important constraints preventing to realize this enlarged characteristic: it is upper limit of the height of spacecraft separation from the upper stage and maximum allowed time interval from start to this event. In other words: spacecraft is to be separated from upper stage not too far from the ground stations and not too late from the start time.

3. Launch scenario and transfer trajectory optimization

As it was described earlier our mission constraints and requirements look different from the standard ones where it is assumed that criteria of optimization is the payload mass, i.e. the mass of spacecraft to be delivered to the planned orbit. In our case we need to choose free parameters of launch scenario and initial osculating elements of transfer orbit which give the minimum amplitude of spacecraft trajectory oscillation with respect to L2 libration point. Mentioned above constraints are to be satisfied as the requirement to be inside limit on maximum available propellant onboard upper stage.

Sequence of operations putting spacecraft onto the operational orbit begins from the launch of the head block (which includes spacecraft and upper stage) onto intermediate elliptical orbit with apogee height 196 km and virtual perigee below Earth surface (height equals minus 485 km). It should be mentioned that this phase of the launch scenario is the mandatory element for any Proton-M launch with upper stage. It is determined by two reasons: the first is requirement to avoid generation of debris in near Earth space, with described approach the last launcher stage enters the atmosphere on the first half of the orbit. The second reason is increasing mass of payload when the final part of delivering head block onto intermediate (parking) low obit is executed by upper stage.

Free transfer orbit parameters of optimization consist of the following: longitude of ascending node, argument of perigee latitude, argument of latitude, date and time of initial transfer orbit point. Also in this list perigee height and semimajor axis are included. Inclination of the orbit plane is fixed and practically coincides with inclination of parking low near Earth orbit equal 51.5°. For standard case when one needs to reach maximum payload (spacecraft) mass the optimal value of perigee height is to be minimal i.e. about 190 km. For this value one may put onto orbit near L2 spacecraft having mass up to6.5 metric tons, it should be mentioned that SRG spacecraft has 2700 kg mass.

Studies have been fulfilled in order to estimate the possibilities of so broad list of free parameters in solving the problem of decreasing the amplitude of the operational orbit near L2 point satisfying the requirement to keep daily visibility duration from Bear Lakes ground station longer than given limits.

As a reference case, the one was taken with low 190 km perigee of transfer orbit. As representative list of launch dates the 20^{th} of each month during year 2018 was chosen. For each of these dates mentioned above parameters were used as optimization variables to maximize duration of minimum daily visibilities taking into account to keep low value of the orbit amplitude.

Table 1. presents the minimum duration values for launches at 20^{th} day each month of year 2018. As one can see the minimum duration happens for launch in May and it is estimated by 2.5 hours. For amplitude minimum values never reach values less than 760000 km and for some dates exceed 1100000 km. It is possible to choose such parameters which allow being

inside 800000 km for the all dates of launch in 2018 but

Table 1. Daily visibilities duration for each month of year 2018

Month #	Visibility hours Ussuriysk	Visibility hours Bear Lakes
1	7.7	5.1
2	9.5	8.0
3	7.9	5.4
4	7.1	3.7
5	6.7	2.3
6	7.6	4.4
7	8.4	6.0
8	8.6	6.5
9	9.4	8.0
10	7.0	3.4
11	7.0	3.3
12	7.3	4.2

To overcome this obstacle the approach supposing increasing perigee altitude was applied. This approach has been tested for several examples confirming its applicability.

It is obvious that for increased perigee of transfer trajectory the standard launch scenario is to be modified. It was done taking into account that the goal of launch is to reach maximum perigee height staying inside limit on available onboard upper stage propellant. As example the calculations have been fulfilled for date of start 15.03.2018.The parameters of the initial orbit (after head block separation from the third stage of Proton-M launcher) are presented by Table 2.

Table 2. Initial orbit after separation of head block from the 3[d] stage of Proton

Parameter	Value
1.Period T, s (hours, minutes, s)	4898,3 (1.21.38,2)
2. Perigee altitude, km	-484,6
3. Apogee altitude, km	196,2
4. Semimajor axis, a, km	6233,8
5.Inclination	51,51°
6.Longitude of ascending node	9,23°
7.Perigee latitude argument	274,0°
8.Argument of latitude of orbit in insertion point	85,3°

it leads to unacceptable decreasing of visibility duration.

The first burn of the upper stage (block DM) engine is to be 6 minutes after separation from launcher in order to put spacecraft onto low parking near Earth orbit with ~190 km altitude. The second burn is to be done approximately one hour after separation, and head block is transferred onto intermediate orbit. Final third burn puts spacecraft onto transfer to L2 trajectory. Time of this burn and other parameters of accompanying operations are chosen as it was described above what gives 15399 km perigee osculating altitude and 1129300 km apogee height. The other characteristics of these phases of launch are given in Tables 3 and 4. Launch phases are illustrated by Fig.1

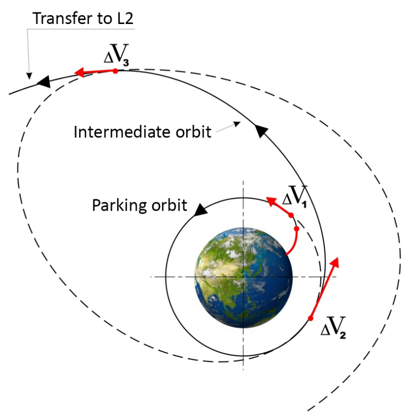

Fig. 1. Launch operations scenario

Table 3. Orbital parameters and delta-V for each of three Block DM burn

Parameter	Value		
	First burn	Second burn	Third burn
Time of engine switch on, hours, minutes,seconds	0.15.51	1.00.49	2.56.39
Argument of latitude of engine switch onpoint, degree	109,1	292,7	91,0
Time of engine switch off, hours, minutes, s	0.16.58	1.09.10	2.59.49
Duration of engine operation, s	67,0	501,0	190,2
Delta-V, m/s	243,6	2648,1	1997,7
Argument of latitude of engine switch off point, degree	113,6	330,8	92,1

Table 4. Trajectory parameters after each of three phases of DM block operations

Parameter	Value		
	Initial (parking) orbit	Intermediate orbit	Transfer to L2 orbit
1.Period T, s	5300,00	54022,5	4381313
2.Perigee altitude km	178,2	255,4	15398,8
3.Apogee altitude, km	206,2	48762,6	1129268
4.Inclination, degree	51,5	51,5	51,5
6.Longitude of ascending node in Greenwich coordinate system fixed at start moment	9,22	9,02	9,0
7.Perigee argument, degree	138,3	309,4	24,58
8.Altitude at the point of putting spacecraft onto transfer orbit, km	-	-	24865,8

4. Algorithm for transfer trajectory design

Bisection method has been used in order to determine the value of osculating semimajor axis a of transfer trajectory similar to the one described in Ref. 1). For this two values of semimajor axis were assumed: minimum and maximum ones. Also two limit values of absolute value of X solar-ecliptic spacecraft coordinates (*Xabs*) were chosen: low limit and upper one. Solar-ecliptic coordinate system is determined as rotating one with the center in the Earth center, X with direction from Earth to Sun, Z axis – orthogonal to ecliptic plane, and Y axis adding to three. Numerical integration was fulfilled on the interval not exceeding 450 days for the a equal mean of minimum and maximum values. If during these calculations *Xabs* reaches upper limit then the interval for a is decreased by two times and as minimum value of semimajor axis the used one is taken. If the minimum *Xabs* is reached, then value of parameter used as a is taken as maximum one for the next iteration of calculation. This quite standard procedure and it is considered as converged one when the time when the *Xabs* limits crossed reaches 450 days for our case. If more durable interval of spacecraft flight along trajectory keeping in proximity of L2 is to be calculated the same algorithm is used, only the number of iterations increases. Requirements for enlargement of this interval are to be satisfied for the cases when the mentioned above parameters of trajectory are to be kept, namely minimum demanded visibility time from ground station and maximum allowed amplitude of orbit with respect to L2. Besides spacecraft eclipse is to be checked during all active life of the spacecraft.

It should be mentioned that for these calculations double precision presentation of data is to be used.

The algorithm was applied for our studies assuming that the other osculating elements were being chosen from some sets dictated by given constraints (for example the inclination of orbit was normally fixed to be equal 51.6 degrees, perigee altitude 190 km or more) and additional heuristic grounds. Practically for any chosen date of launch there is very wide combinations area of osculating orbital parameters for transfer trajectory putting spacecraft to orbit near L2 without any additional delta-V impulse. Appropriate semimajor axis value does exist for any of them and can be calculated by described algorithm as it was shown for example in [2]. So we have sufficient level of flexibility for the project optimization.

5. Orbital maneuvers

Orbits in proximity of collinear libration points L1 and L2 are unstable what means that without correction maneuvers spacecraft would leave this proximity. The problems of optimal orbital control for keeping the spacecraft on operational trajectory are studied in many papers; some of the results of these explorations published in paper[1)] were implemented in real missions. Besides keeping spacecraft on the trajectory near libration point orbital maneuvers are necessary for decreasing amplitude of the orbit, especially its component along Y axis in solar-ecliptic coordinate system as it was done for example for Planck project. Z component change may be also the goal of delta-V applying, for example for eclipse avoiding. In the mentioned above paper formulae are presented for calculation required values of delta-V as functions of amplitudes demanded change. For change A_y amplitude ΔV impulse should be applied at the time 4.7883 days before XZ plane crossing (there are two such points on each orbit):

$$\Delta V = \Delta A_y \cdot 3.648001 \cdot 10^{-7} s^{-1}$$

To change A_z applying minimum ΔV the direction of impulse should be along Z axis and applied at the point where $z=0$. Its value is determined by formula:

$$\Delta V_z = \Delta A_z \cdot 3.952326 \cdot 10^{-7} s^{-1}$$

As to the optimal impulse direction for A_y change, the vector of impulse should lie in ecliptic plane and to be orthogonal to the line having +28.6° angle with Sun-Earth direction. Any delta-V impulse laying in plane orthogonal this direction will not to escaping spacecraft from L2 proximity. But it should be taken into account that given figure is calculated in linear approximation of the trajectory. For real planning bisection method should be applied with the use of the figures received by numerical integration. As it was done for trajectory design the goal of bisection procedure is to chose delta-V direction which keeps trajectory in L2 proximity during given duration.

For our project it was not planned to use maneuver intended for nominal changing trajectory parameters, i.e. the trajectory without errors is to satisfy the requirements. So it was assumed that during motion to the L2 vicinity the propellant is allowed to spend only for corrections of launcher error during putting the spacecraft onto transfer trajectory and then for keeping spacecraft on operational orbit. For solving these tasks on transfer trajectory about 50 m/s is planned from initial total amount equal 150 m/s. It means that propellant mass reserved for the tasks of spacecraft planned 5years maintenance are large enough to hope for flexibility of planning the motion control operations, even taking into account the fuel required for attitude control. Now as the area of risks the estimation of propellant mass, required for delta-V preventing spacecraft leaving vicinity of L2, is considered. Required delta-V for orbital correction maneuvers may be compared with the ones consumed during previous missions fulfilled to the L1, L2 vicinities. Basing on the figures received during maintenance during Planck, Hershel, Gaia, SOHO [3] and other missions, one may expect that without some emergency cases the expenditure of delta-V will hardly exceed 10 m/s per year. For example in case of Planck these maintenance maneuvers demanded less than 1.5 m/s per year. The maneuvers for that are supposed to be executed following optimization requirements[1)], i.e. with the delta-V vector directed along line as described above. Decisions related to possible additional maneuvers to modify trajectory characteristics may be done after some level of experience would be reached.

6. Results analysis

Following Fig. 2,3 present two trajectories: first one (blue) for launch from low parking orbit directly, the second one (orange) – for launch from intermediate high elliptical orbit. Parameters of orbits before final burn of engine are given in Table 4, besides argument of perigee latitude for departure orbit, what is taken as 10°. The value of semimajor axis was equal 565886.814 km for high perigee case.

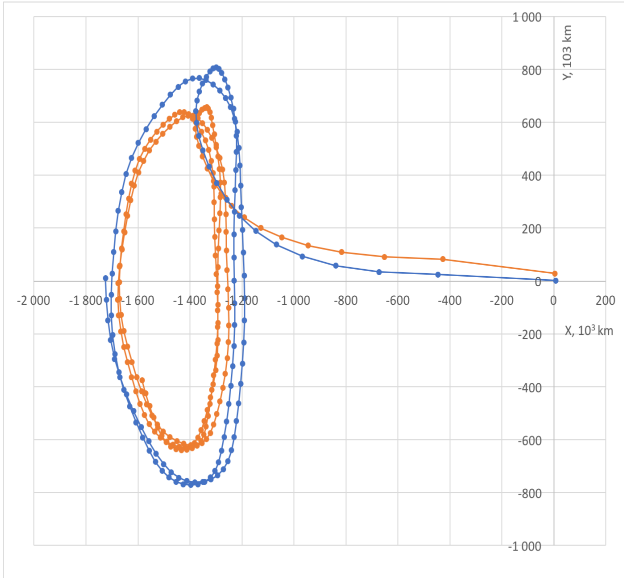

Fig. 2. Projection of transfer and operational trajectories on XY plane in solar-ecliptic coordinate system. (Orange – high perigee, blue – low perigee, tics – three days)

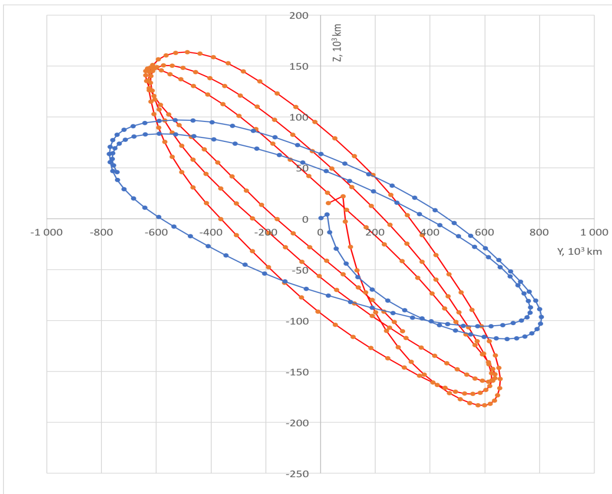

Fig. 3. Projection of trajectories on XZ plane

One can see that using scenario with high perigee (15399 km) intermediate orbit allows to decrease Y amplitude of L2 orbit by 170000 km for launch at March 15 what is considered as quite meaningful figure. Analysis of the application the same approach for the other dates of launch confirms this conclusion.

Figure 4 gives the values of spacecraft visibility duration assuming 5 degrees above horizon for two ground stations Bear Lakes and Ussuriysk depending on the days since launch for the given above trajectory with start at March 15, 2018 for the scenario included intermediate orbit and high perigee for transfer trajectory.

For the mission the critical figure is minimum duration of the visibility. In the presented case 5 hours figure which corresponds to Bear Lakes looks as quite satisfactory and it is very close to the one for low perigee launch scenario (5.4 hours). It means that proposed scenario, decreasing the amplitude does not lead to the visible decreasing of visibility duration. This statement may considered as correct for practically any date of launch during the year with the only exclusion in May when during several days we lose the visibility from Bear Lakes, but it is kept from Ussuriysk.

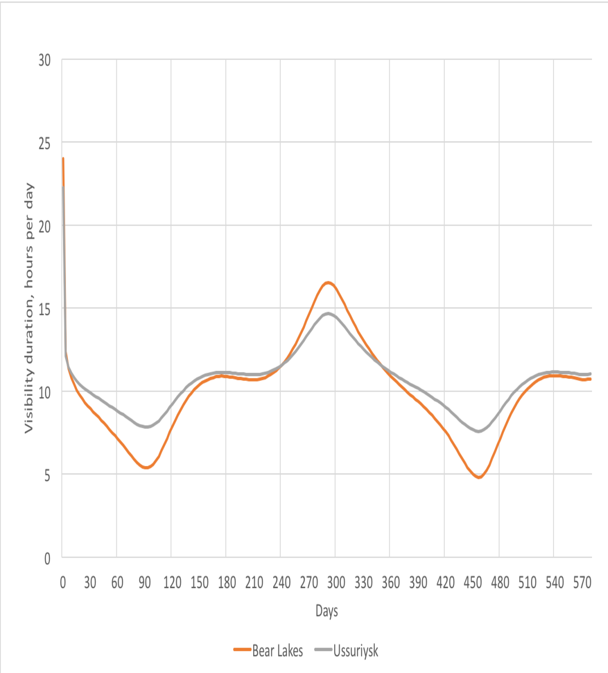

Fig. 4. Days from launch, start to transfer trajectory from high perigee intermediate orbit

As it was mentioned above the principal requirement applied for our project mission design is minimize possible risk in framework of satisfying the demands generated by planned physical experiments onboard spacecraft. It is assumed that some of spacecraft characteristics seeming now excessive (propellant mass for example) after first tests of the system may be used for improving the quality and enlarge the possibilities of project at large as it was done in the famous ISEE-3 (ICE) mission by R. Farquhar [4)].

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