COMBINED FLIGHT PROFILE TO INSERT TELECOMMUNICATION SATELLITE INTO GEOSTATIONARY ORBIT USING ROCKOT LIGHT-WEIGHT CLASS LAUNCH VEHICLE

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Khrunichev State Research and Production Space Center is carrying out Research & Development of unified space bus Yacht which should be inserted into the orbit using Rockot, Angara-1.1 or Angara-1.2 light-weight class launch vehicles. The Yacht space platform is equipped with electric propulsion module based on stationary plasma thrusters SPT-100. One of Yacht applications is geostationary telecommunication satellite. It is proposed to use combined flight profile to insert the geostationary satellite into GEO. Within this profile, the Breeze upper stage is used to form an elliptical transfer orbit. The insertion into GEO from the transfer orbit is provided by the spacecraft's electric propulsion module. The electric propulsion module is used during the GEO operations as well, including station-keeping and reaction wheels unloading. The carried out study shows that spacecraft having mass 450-500 kg with 100-150 kg of telecommunication equipment can be delivered into GEO using Rockot launch vehicle from Plesetsk or Baykonur launch sites.

ABBREVIATIONS

BOL	- Beginning of life
EOL	- End of life
EPU	- Electric propulsion unit
GEO	- Geostationary orbit
LEO	- Low earth orbit
LV	- Launch vehicle
PSS	- Power supply system
SC	- Spacecraft
SPT	- Stationary plasma thruster
TO	- Transfer orbit
US	- Upper stage

INTRODUCTION

Khrunichev Space Center is developing the new geostationary communication satellite based on unified space platform «Yacht»[1,2,3] (Fig. 1). The communication spacecraft has electric propulsion unit (EPU) having 4 stationary plasma thrusters SPT-100 of Fakel design bureau. The thrusters are placed on the rotation gears. Only 2 thrusters can run simultaneously. EPU provides interorbital transfer to the GEO, reaction wheels unloading, station keeping, and orbital maneuvers.

Electric propulsion allows reduce the cost of SC insertion into GEO. High specific impulse of electric propulsion thrusters leads to the decreasing of mass consumption which is required for insertion in comparison with conventional chemical propulsion. Therefore the lesser initial mass is required in the parking LEO when electric propulsion is used. So the smaller launch vehicle can be used. This is the main factor which decreases insertion cost. To reach acceptable economic indicators the common EPU and common power supply system (PSS) should be used for providing of interorbital transfer and on-GEO operations. The EPU's low thrust causes long transfer duration. The onboard systems lifetime expends during this transfer and solar arrays degradates due to multiply Van Allen Belts crossing. The transfer duration should be shortened to reduce these penalties. The following ways can be used to decrease transfer duration:

- Increasing EPU thrust when specific impulse remains invariable. The corresponding increasing EPU and PSS electrical power is necessary. This leads to growth of EPU/PSS mass and cost.

- Increasing EPU thrust when input electrical power remains invariable. The decreasing of EPU's specific impulse is necessary to realize this option. This requires EPU revision and leads to increase of mass consumption.
- Combined flight profile for insertion. Upper stage (US) delivers spacecraft from the parking LEO into an elliptical transfer orbit (TO). The interorbital transfer from TO to GEO is provided by spacecraft's EPU. This profile is named «combined» because both high- and low-thrust propulsion are used.

All ways of transfer duration shortening are compromise ones. The main shortcoming of first way is essential cost increasing because increasing power of solar array which is one of most expensive component. The solar array power should be found to supply communication equipment and other on-board systems at the EOL taking into account solar arrays degradation. So power excess of solar array in comparison with power providing SC operations at EOL can be considered as source of cost and mass penalties. The second way of decreasing transfer duration has restricted usage because restricted ability and effectiveness of required specific impulse realization. Therefore there was taken decision to use combined flight profile for new communication spacecraft of Khrunichev Space Center.

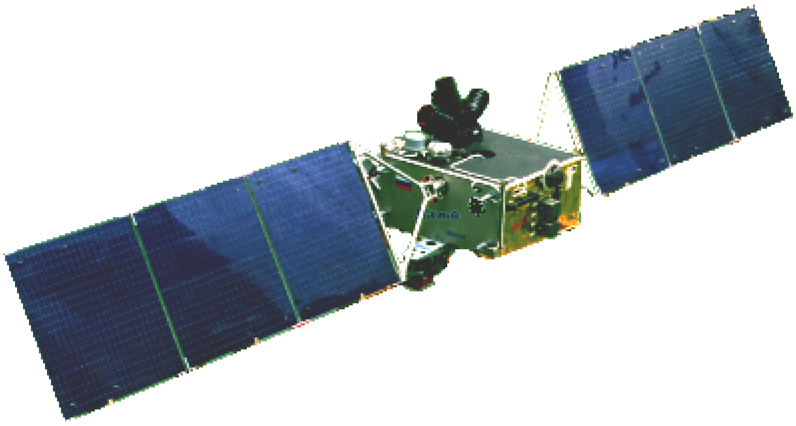


Fig. 1. Unified space platform "Yacht"

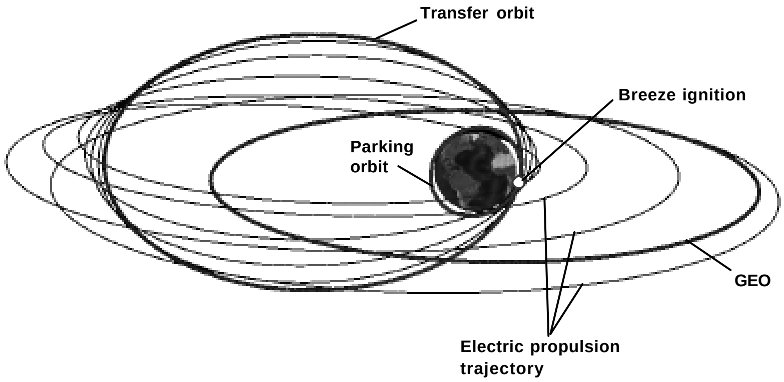


Fig. 2. Combined flight profile

FLIGHT PROFILE

The main flight profile version (Fig. 2) based on Rockot light-weight class conversion LV launching from the Plesetsk site. Rockot LV inserts SC with Breeze-KS upper stage into the circular parking LEO having altitude 250 km and inclination 75°. The first ignition of Breeze-KS is used for this insertion. The second ignition of Breeze-KS has place in the vicinity of orbital node. As result SC is delivered into an elliptical transfer orbit. The spacecraft's EPU runs after the upper stage separation. EPU provides transfer from the transfer orbit into the GEO.

Additional version of flight profile uses Baykonur launch site. Permissible parking LEO inclinations are 65° and 51.3°.

MAIN DESIGN AND TRAJECTORY PARAMETERS

The Rockot fairing and upper stage is modified to meet the requirements of communication spacecraft insertion. The general requirements are following: the transfer duration should not exceed 6 months, the communication equipment has mass 100-150 kg and power consumption 1000 W.

Upper stage design parameters variation allow to vary available mass in the parking LEO within some range. This defines available apogee altitude of transfer orbit when spacecraft mass is fixed. The transfer duration depends on TO apogee altitude, initial mass of spacecraft, EPU thrust and specific impulse, and EPU thrust control.

EPU is equipped by regular version of SPT-100 thrusters having proven 7500-hours lifetime and high reliability. So specific impulse is fixed and it equals to 1500 s. EPU thrust is limited by number of running SPT-100 at the same time. The decision to use 2 simultaneously running SPT-100 during interorbital transfer was taking in the early phase of R&D. So required solar arrays power is approximately equals to 3500 W. Such electrical power is enough to supply both simultaneously running of 2 SPT-100 and communication/service equipment operating at EOL including station-keeping taking into account 30-40% degradation of solar arrays.

So, taking into account existing constraints, following main design and trajectory parameters define transfer duration and spacecraft mass in GEO at BOL:

- Initial mass in the parking LEO (it can be varied in a little range by means of upper stage and fairing modification);
- Apogee altitude of transfer orbit (it depends on initial mass in the parking LEO, upper stage design parameters, spacecraft initial mass);
- EPU thrust control during transfer from TO to GEO (it defines required mass consumption for given spacecraft mass in GEO at BOL).

The problem is to find compromise solution to provide insertion in GEO of maximum mass for an minimum time.

The mission design and analysis was carried out on base of solution minimum time optimal control problem for given spacecraft mass in GEO at BOL and TO apogee altitude.

LOW-THRUST TRAJECTORY OPTIMIZATION[4]

The problem of minimization transfer time was solved to obtain thrust steering. Pontryagin's maximum principle was used to solve the optimal control problem. Optimal thrust steering was found for motion in the Newton's gravity field without any constraints on attitude angles and their rates. Obtained solution was refined for more realistic model taking into account real earth gravity field and solar-lunar perturbations. As it follows from the optimality conditions, EPU should run during all transfer from TO to GEO (excluding eclipse trajectory arc when it is impossible to provide EPU running). The new software was carried out to solve problem of multi-orbits minimum time transfer between non-coplanar elliptical orbits. Analysis of optimization results shown in the Figs. 3-8. At the initial transfer phase spacecraft is accelerated using low yaw angle in perigee and one close to 90° in apogee (the yaw angle increases from 75°-80° to 90°, see Fig.4). A low thrust component, which directed against velocity, has place in the apogee vicinity within this phase (Fig. 3).

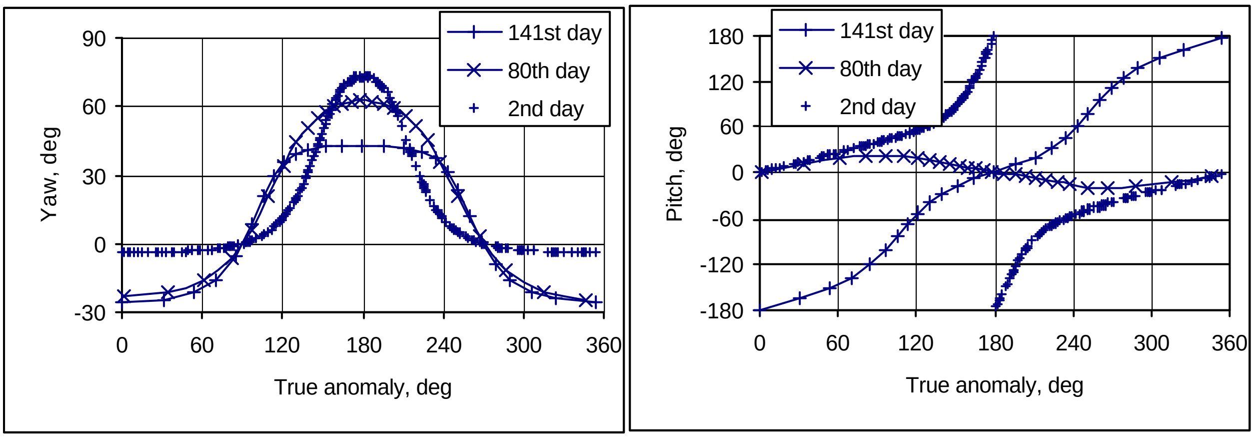


Fig. 3. Optimal yaw and pitch versus true anomaly

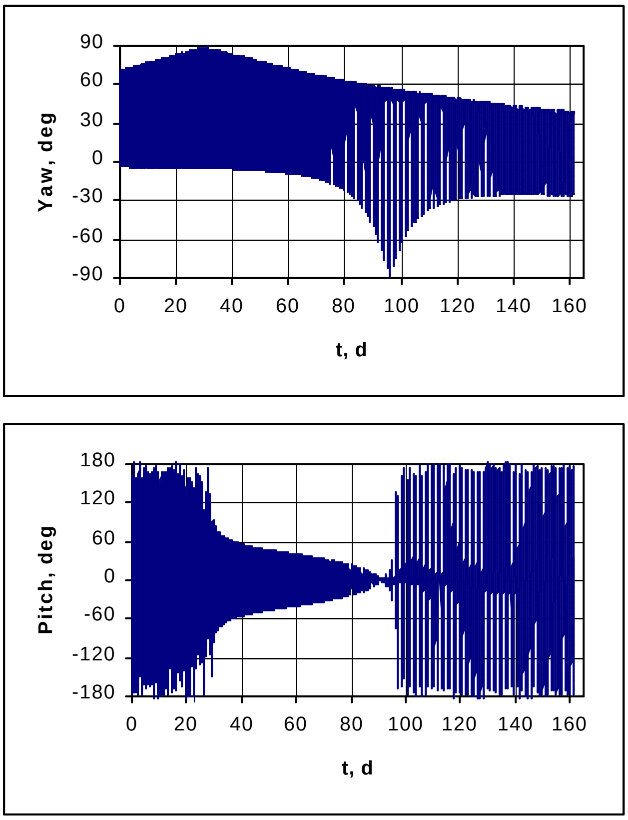


Fig. 4. Optimal thrust steering

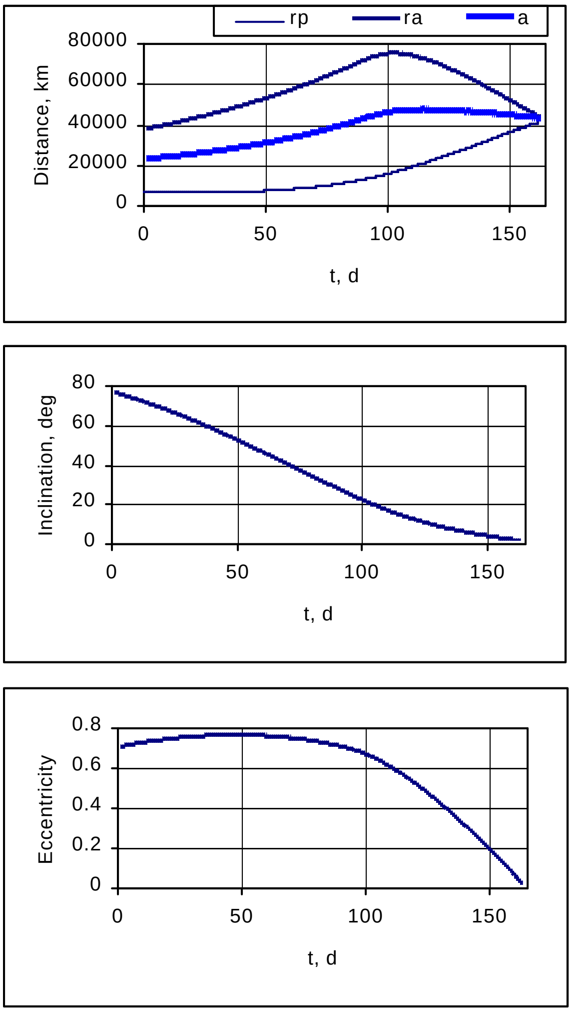


Fig. 5. Orbital evolution
(rp – perigee radius, ra – apogee radius, a – semi-major axis)

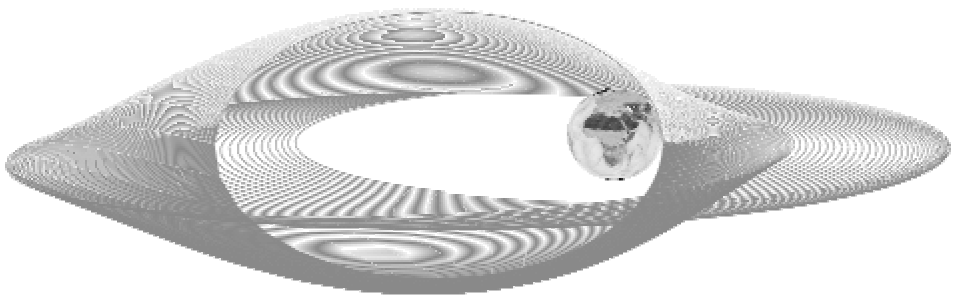


Fig. 6. Trajectory from transfer orbit to GEO

This thrust component partially compensates perigee growth due to acceleration in the greater part of orbit. At the second transfer phase spacecraft is accelerated during whole orbit. The maximal yaw angle near apogee decreases from 90° to 30°-60° and near perigee increases to 90° when orbit reaches maximal eccentricity at the end of this phase. When eccentricity close to maximum, the thrust projection on the orbit plane is directed practically along velocity. At third transfer phase maximal yaw angles near apogee and perigee decreases to 30°-40°. The spacecraft is accelerated in apogee to increase perigee and brakes in perigee to decrease apogee.

Dependencies of yaw and pitch angles versus time are presented in the Fig. 4. This example corresponds to transfer from TO having perigee altitude 250 km, apogee altitude 30000 km, inclination 75°; initial spacecraft mass in the TO equals to 630 kg, spacecraft mass in the GEO at BOL equals to 472 kg. Projection of optimal thrust direction on the orbital plane oscillates between tangential and circumferential direction during second transfer phase and this projection rotates in the bound (1st and 3rd) phases providing acceleration/braking.

The evolution of orbital parameters is shown in the Fig. 5. It should be noted that maximal apogee altitude can essentially exceeds GEO altitude. For example, the maximal apogee altitude is 68000 km for TO apogee altitude 30000 km and TO inclination 75°. The perigee altitude during 1st phase remains practically invariable.

An example of 3D-view of low-thrust trajectory is presented in the Fig. 6.

MISSION ANALYSIS

The following parameters were varied to analyze mission performance using combined flight profile:

- TO apogee altitude;
- Initial mass in the parking LEO;
- Spacecraft mass in the GEO at BOL.

Figs. 7,8 presents dependency of transfer duration with respect to TO apogee altitude and inclination when spacecraft mass in GEO at BOL is given (485 kg). Every points of these plots is computed using solution of minimum time transfer problem between given TO and GEO.

One can see from Fig. 7 that optimal TO apogee altitude exists for every TO inclination. This TO apogee altitude provides minimal transfer duration for given EPU parameters and spacecraft mass in GEO at BOL. Thus the optimal TO altitude is ~150000 km when TO inclination is 75° and it is ~70000 km when TO inclination is 0°.

But use Rockot LV with Breeze-KS US imposes constraints connected with ability to deliver SC into given TO. The dashed curve on the Fig. 9 represents dependency of required spacecraft mass in TO versus TO apogee altitude (TO inclination is 75 when TO inclination is 75°) to insert 485 kg in GEO. The solid lines represent spacecraft mass in TO which can be delivered using Breeze-KS US having given dry mass. Points of crossing dashed and solid lines defines TO apogee altitude which provides minimum duration of TO-to-GEO transfer using US having given dry mass.

Analogously it was computed dependency of transfer duration versus given initial mass in the parking LEO (within range 3350-3750 kg), spacecraft mass in GEO at BOL (within range 425-575 kg), and Breeze-KS dry mass (within range 900-1650 kg). These results are presented in Fig. 10. One can see, in particularity, the transfer duration is 140-190 days when spacecraft mass in GEO at BOL is 475 kg, Breeze-KS dry mass is 1000 kg, and initial mass in the parking LEO is varied within range 3350-3750 kg. So, the 100 kg increase of initial mass in the parking LEO leads to ~12 days shortening of transfer duration due to increasing TO apogee altitude. The transfer duration is 140-200 days when initial mass in the parking LEO is 3550 kg, US dry mass is 1000 kg, and spacecraft mass in GEO at BOL is varied within range 425-575 kg. So, 100 kg increasing of spacecraft mass in GEO at BOL leads to ~40 days increasing of transfer duration.

If transfer duration should not exceeds 6 months then the 575 kg (475 kg) spacecraft can be delivered in GEO only if US dry mass is not greater 960 kg (1140 kg). The transfer duration from TO having inclination 75° decreases to 4 months if initial mass in parking LEO is 3550 kg, spacecraft mass in GEO at BOL is 425 kg, and US dry mass is 900 kg.

Design estimation of Yacht space platform mass in GEO at BOL is 350 kg (without commercial communication equipment). So spacecraft mass in GEO at BOL should be 450-500 kg for communication equipment mass 100-150 kg. The dry mass of Breeze-KS US should not exceed 1120-1040 kg respectively.

CONCLUSION

The carried out analysis shows capability to use Rockot LV and combined flight profile to insert into GEO spacecraft having mass 450-500 kg with 100-150 kg of commercial communication equipment. The duration of transfer to GEO s 5-6 months for launching from Plesetsk site. The transfer duration can be decreased on one month using Baykonur launch site due to inclination of LEO parking orbit decreasing.

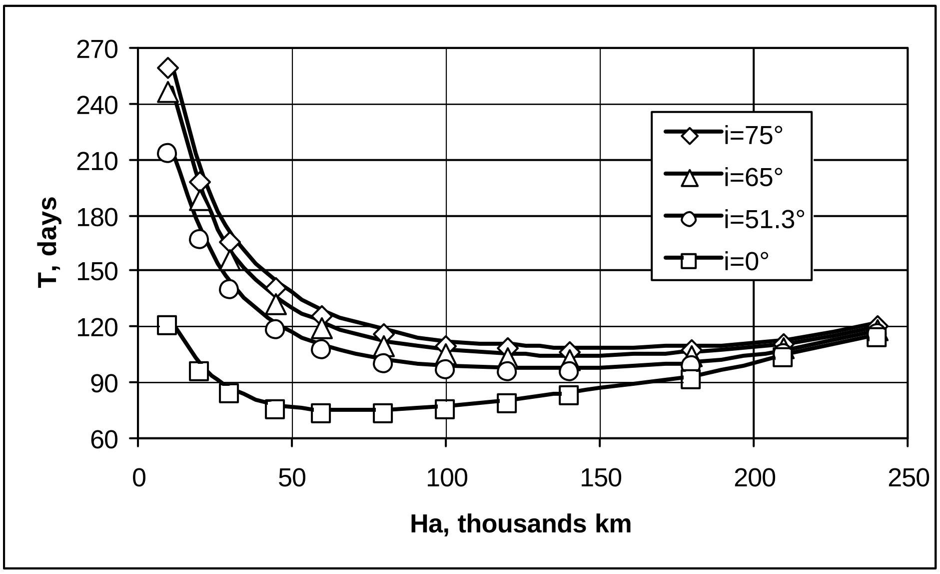


Fig. 7. Transfer duration versus apogee altitude
(SC mass in GEO at BOL 485 kg)

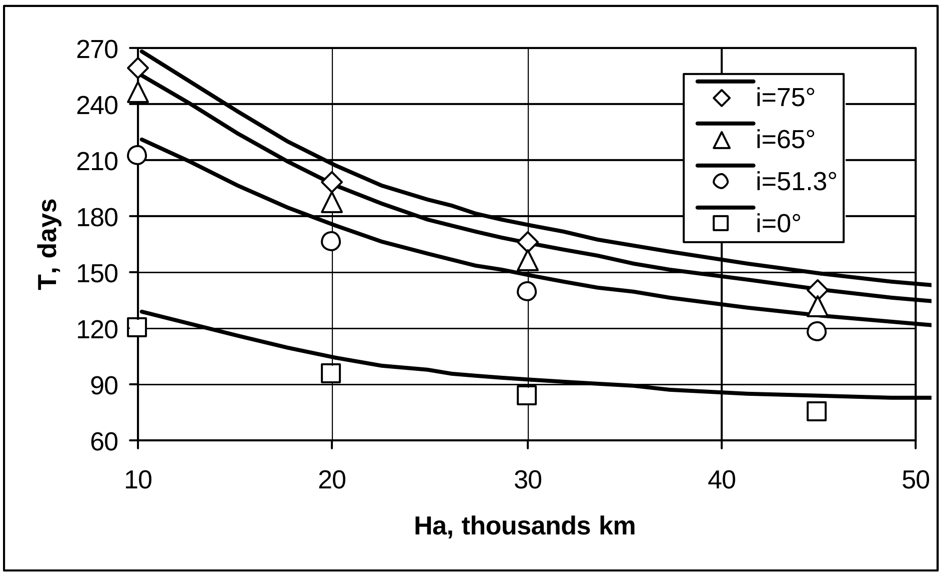


Fig. 8. Transfer duration versus apogee altitude
(SC mass in GEO at BOL 485 kg;
range of TO apogee altitudes 10-50 thousands km)

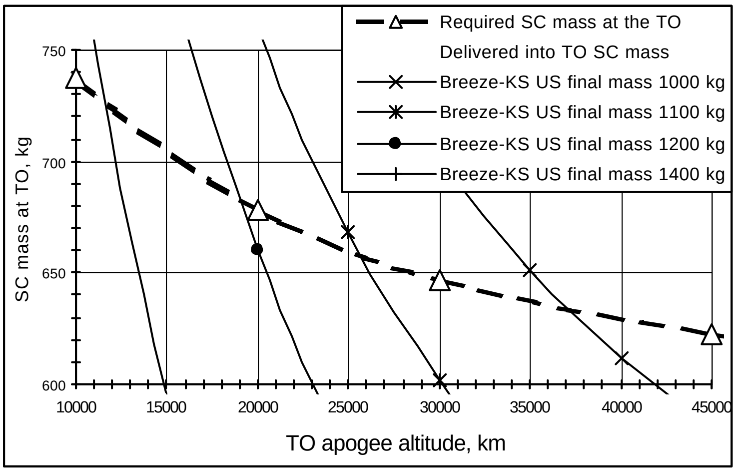
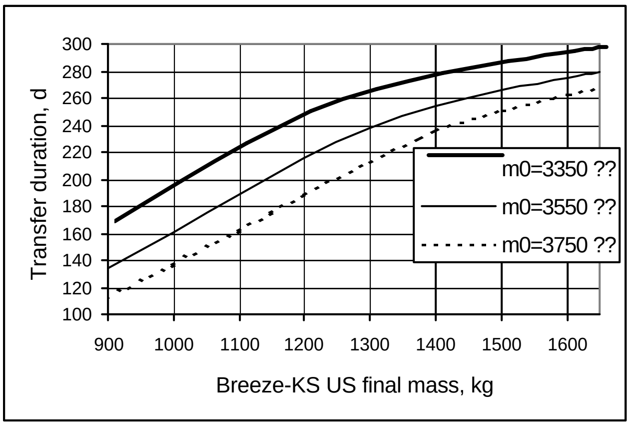


Fig. 9. Required and delivered SC mass at the transfer orbit vs. TO apogee



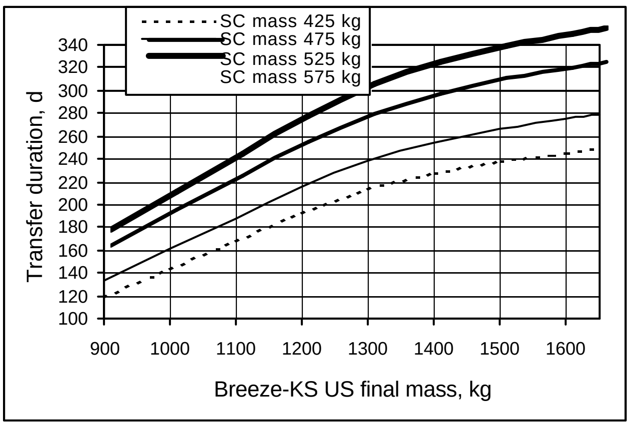


Fig. 10. Transfer duration vs. US final mass, mass delivering into the parking orbit, and SC mass at the GEO in BOL

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