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ELECTRIC ORBIT RAISING – ADVANTAGES, TRANSFER ASPECTS, SOLUTIONS

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Electric orbit raising (EOR) seems to become an important driver for future satellite and launcher designs. The majority of available commercial telecommunication platforms are already using electric propulsion (EP) technology for station-keeping, and many are now trying to exploit the higher specific impulse for the transfer from launch orbit to geosynchronous equatorial orbit (GEO). While nowadays those orbit transfers are pure chemical, the next step will be not only pure electric orbit transfers but also hybrid transfers.

Typical aspects of orbit transfers with electric propulsion and their impact on the trajectory will be discussed by this paper. First, the selection of a proper launch orbit and date is mandatory. Both might have strong influence on the transfer performance like the seasonal effects of eclipses. Next, there are several constraints and issues related to spacecraft sub-systems to be considered in trajectory computation and optimization, for example radiation, eclipses, heat dissipation and collision avoidance especially with assets in the GEO ring. Other aspects related to spacecraft technology are limitations in attitude, rotation and torques and are addressed as well as restrictions in thruster operations (e.g. cycling). Further, navigation and contact with ground station network or a single ground station must be taken into account for the whole transfer. Hybrid transfers, which combines chemical and electrical transfer are beneficial for very large GEO platforms in order to keep the transfer duration within 6-12 months. All transfer aspects together define the valid trajectory for the orbit raising.

Another aspect of electric orbit transfers is the operation of the spacecraft. Periodic updates of the attitude profile are proved necessary to cope with uncertainties and fulfil all transfer constraints. Only in that way the transfer follows the optimal trajectory as close as possible. The paper will discuss possible strategies for an operational concept where the spacecraft state will be updated by orbit determination to re-optimize the manoeuvre plan.

Finally the cross-impact of satellite concepts with the electric orbit raising capabilities and future launcher concepts will be discussed.

I. INTRODUCTION

In the last years the maturity of the electric propulsion (EP) has increased and was already used for orbit raising of telecommunication platforms. Since most of these satellites are located in a geostationary orbit and direct launch injection is very expensive, electric orbit raising is perfectly suited reducing launch costs because the satellite can be injected in a lower orbit, e.g. geostationary transfer orbit (GTO), where smaller launchers and therefore low-cost launch opportunities can be used. The advantage of EP technology in terms of mass saving for orbit raising with regard to pure chemical transfers outweighs the loss of the longer transfer duration due to the lower thrust magnitudes. On the other hand those low-thrust transfers have to deal with several specific issues for spacecraft operations, like handling of eclipses and collision avoidance with assets in the GEO (Geostationary Earth Orbit) ring. Other important aspects arise from the subsystems of the satellite: available power as well as thruster firings may be limited, and the spacecraft's AOCS system may constrain the optimal direction of the thrust vector.

However, within the next few decades the electric propulsion technology will very likely become the preferred solution for placing commercial telecommunication satellites in the GEO ring. And in the more near future some of the satellites may have to use chemical and electric propulsion as hybrid transfer solution.

II. LAUNCH

Launch Orbit Analysis

For end to end (E2E) trajectory optimization it is mandatory to include detailed analyses of the launch orbit [1]. In particular, some launchers (e.g. Indian PSLV) give the possibility to freely choose the conditions of the insertion orbit of the spacecraft. Unfortunately, the launcher manual gives only very limited information about the launch performance and which orbits can be achieved for a given payload.

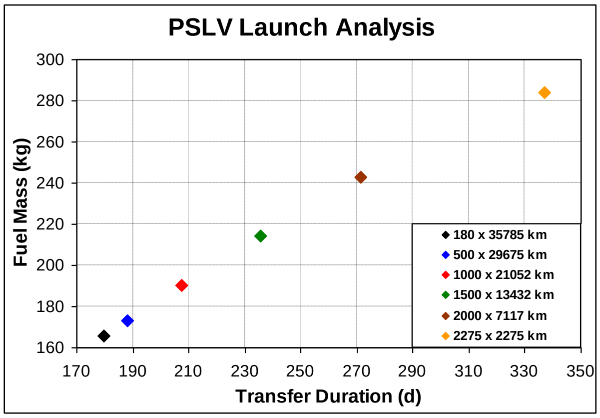

Fig. 1: Propellant mass and transfer duration of the PSLV launch analysis.

Only the nominal GTO is described as well as a circular orbit with maximum altitude. In the analysis the GTO was kept. In further steps certain periapsis altitudes were selected and the model for the initial launch orbit was optimized for maximum apoapsis altitude constraining the payload mass. The inclination was always kept at 17.8 deg. Finally, the last step was to achieve a circular orbit. It is obvious to identify the loss of orbital energy the more circular the achieved orbit becomes.

All orbits were used as starting condition for the low-thrust transfer to the GEO ring. The results of this launch analysis can be found in Fig. 1. As expected, the nominal GTO is the best initial orbit for the transfer to the GEO belt. It can be concluded that an initial orbit is better suited the higher its orbit energy, at least for pure GTO-GEO transfers. Limitations might arise on the lower limit of the perigee considering atmospheric drag.

Of course, the EP efficiency increases with more circular orbits since the low-thrust losses in high-eccentric orbits such as GTO are large. But the best initial orbit also depends on the launcher itself. For example, the Ariane 5 ECB is able to lift the satellite in a very high altitude circular orbit [2], whereas the PSLV is not able to do so.

Impact on Future Launchers

Since the design of satellites and the design of launch systems are highly connected, the situation is similar to the chicken-and-egg problem: during the design of a new satellite platform the list of available launch-systems is considered in order to define a mass-class for the spacecraft. Similarl during the design of a new launch-system the future trend in satellite masses is estimated in order to provide a valid alternative to the already existent launchers. It should be considered that the design of a new launch-system requires more time than the design of a new satellite, therefore typically the spacecraft design has to follow the rocket design.

An exception to the described procedure is identified in the radical changes introduced by EOR: new telecommunication satellites present requirements that are influencing the next generation of European launchers. Considering electric orbit raising, also small launchers like VEGA (i.e. its evolution) could be considered for telecommunication satellites. The final effect is that a new set of requirements is added to the launcher design: faring internal diameter, target orbit, etc.

While evaluating the correlation between satellites and launch-systems, another aspect must be considered: EOR is reducing the launch mass of the satellites. In a classical pure chemical orbit-raising GTO-GEO transfer around one third of the launched mass is propellant required to place the satellite in the final orbit. Thus, a new launcher-system designed for GTO should take into account the change in the trend mass for telecommunication satellites. The state-of-the-art capability to launch 7-8 metric tons could be not a driving requirement anymore in 10-20years.

Clearly, EOR will not be implemented in all new satellite platforms, but the fraction of spacecraft using it will rise in the future. And the launcher market should be aware of it.

III. TRANSFER

Of course, the objective of an electric orbit raising can be manifold, such as time or propellant. Special aspects related to the satellite sub-systems are detailed in the next sections. Any optimization of an EOR transfer may consider some or all of these issues [3]. Please bear in mind that the examples shown in this section are understood as reference transfer scenarios.

But before starting with optimization results of low-thrust orbit transfers the computation of the initial guess is briefly discussed. For sake of convenience analytic attitude control laws, as described for example by Pollard [4], are used. The control laws are for changing the semi-major axis, eccentricity, or the argument of periapsis, and have been augmented by an out-of-plane control strategy for efficient inclination change.

For a typical GTO to GEO transfer with an initial inclination of 27 degrees two different phases can be identified in the attitude control history: first half of the transfer the orbit energy is increased at a maximum rate with only a small portion for the inclination change. Once the desired orbit energy, here GEO orbit altitude, is achieved, the control of the remaining transfer circularizes the orbit shape and reduces the inclination to zero. This simplified orbit transfer is an excellent initial solution for the following optimization process.

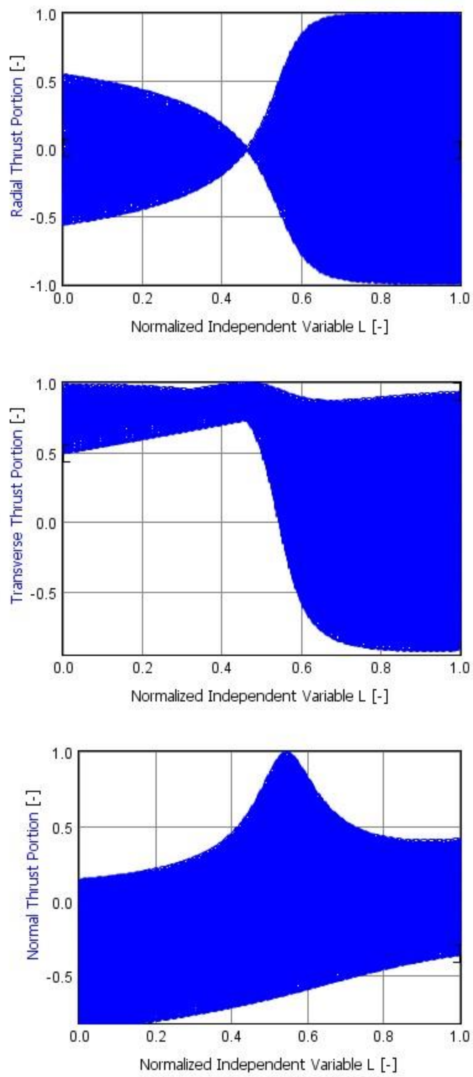

Fig. 2: Control history of a time optimal GTO to GEO transfer after converged optimization: radial thrust vector control component (top), transversal component (middle), and the normal component (bottom).

Time Optimality

The analytic solution is an excellent starting point for the optimization of the transfer while minimizing the transfer duration. After the optimization the transfer duration (and therefore the propellant consumption) is reduced by about 7%. The optimal control history of a time optimal GTO-GEO transfer is shown in Fig. 2 and the optimized orbital elements semi-major axis, eccentricity and inclination are shown in Fig. 3.

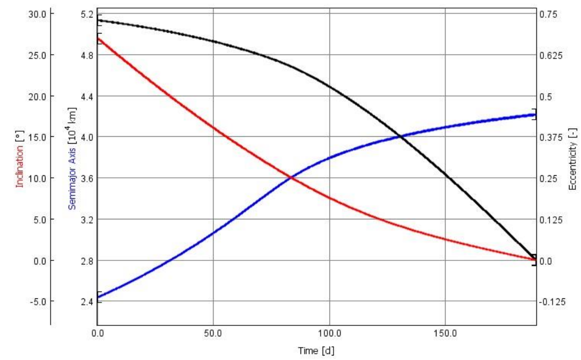

Fig. 3: Semi-major axis (blue), eccentricity (black) and inclination (red) of time optimal transfer.

Fuel Optimality

For the minimization of the propellant consumption the transfer duration needs to be extended. The longer the transfer with respect to the time-optimal solution the more propellant mass can be saved. But there is a minimum fuel consumption required to bend the trajectory to the desired target orbit. The optimal Pareto-front is given in Fig. 4. Here, the fuel consumption of 125 kg for a time optimal transfer can be reduced to about 100 kg when the transfer duration is extended by more than 50%. It seems that further extended mission duration will not significantly reduce the fuel consumption.

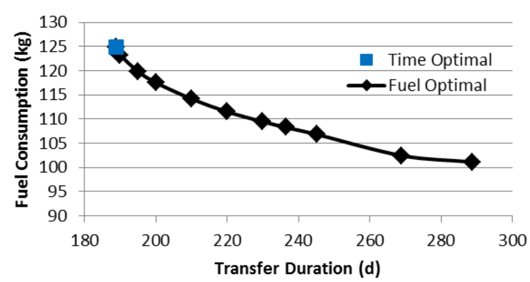

Fig. 4: Fuel consumption versus transfer duration of fuel optimal transfers.

GEO Box Targeting

Since a satellite is usually brought to a certain geographic longitude position, the GEO box, this phasing aspect must be considered properly. In the worst case the transfer duration is increasing by almost 1 day when targeting a specific GEO slot location. The phasing strongly depends on the initial launch epoch, orbit and target GEO box longitude. In the given example the spacecraft targets a longitude of 15° East while in the time-optimal transfer it was approaching at about 13° East. Thus, the transfer needs to be extended by almost one day for the phasing. The changes in radial and normal attitude control component are quite obvious in comparison to the time optimal solution (see Fig. 5 and Fig.2).

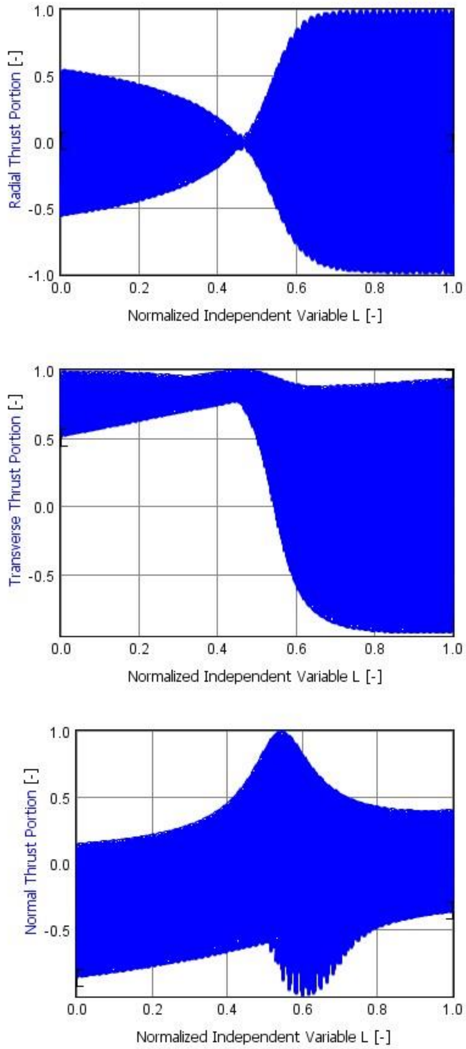

Fig. 5: Attitude profiles for time-optimal transfer with GEO box targeting: radial (top), transverse (middle) and normal (bottom) component.

Crossing the GEO Ring

Depending on the orbital parameters of the transfer trajectory there are three possible domains where the spacecraft might cross the GEO ring: at the beginning, mid of the transfer, and at the end. The GEO ring, or GEO belt, is understood as the area in space containing most of the operational satellites in geosynchronous orbit.

At the beginning of the transfer any crossing of the ring can be simply avoided by choosing a proper launch orbit. For example, the initial apoapsis can be located above GEO altitude. Next, a minimum inclination of few degrees is required to circumnavigate the GEO ring.

During the mid-transfer the situation is more complex. Here, crossings strongly depend on the argument of periapsis and eccentricity of the transfer orbit. It is required to have the instant radius of ascending and descending node not close to GEO radius. If so, the spacecraft would cross the ring two times per orbit, posing a serious threat for every asset in the GEO belt. Again, the orbit has to maintain a minimum inclination as well.

Whereas crossings of the belt can be avoided at the beginning and in the mid-part of the transfer by choosing proper orbit geometries, it is trickier when approaching the belt. Since zero inclination is required at the end of the low-thrust transfer to the GEO target the spacecraft may cross the GEO ring several times. To avoid the risk of a possible collision with assets in the ring, a special constraint is used forbidding the spacecraft to travel through the GEO ring. The situation is illustrated in Fig. 6 in a co-rotating frame. The x-axis is the direction from the centre of Earth towards the projected spacecraft position in the equator plane (r-bar), and the y-axis is the out-of-equator plane component pointing north (h-bar). The red rectangle is indicating the GEO ring. As it can be clearly seen, the spacecraft circumnavigates the GEO ring after consideration of the constraint. The fuel consumption and transfer duration is increasing by less than 0.02% (see Table 1).

It is known that the number of crossings strongly depends on the initial and final orbit conditions, in particular the argument of periapsis. The numbers given above are for a standard Ariane 5 GTO to GEO transfer where the final orbit is a parking orbit 500 km below GEO altitude.

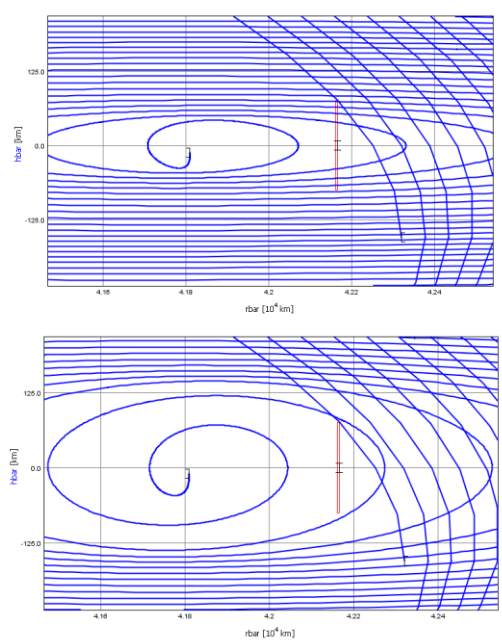

Fig. 6: Super-synchronous transfer without GEO ring avoidance (top figure) and with active GEO ring avoidance (bottom figure). The GEO ring is indicated by the red rectangle. Both plots show the transfer trajectory in a co-rotating frame.

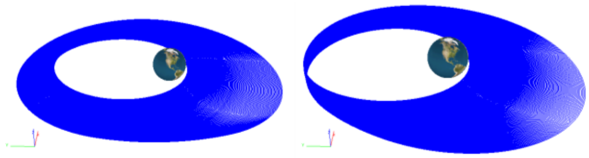

Fig. 7: 3d plot of super-synchronous (left) and sub-synchronous transfer (right).

	GEO Ring Crossings	Transfer Duration	Propellant Consumption
Super-Synchronous	7	100.0%	100.0%
Sub-Synchronous	0	110.9%	110.9%
GEO Ring Constraint	0	100.02%	100.02%

Table 1: Performance table for time optimal transfers regarding GEO ring crossings.

An alternative to reduce the number of crossings is the sub-synchronous transfer (see Fig. 7) with the spacecraft staying below GEO altitude during the whole electric orbit raising. Typically, the transfer duration and fuel consumption (in case of continuous thrusting) is increased by about 11% in comparison to a super-synchronous transfer with the GEO belt constraint.

IV. POWER

Telecommunication satellites located in a geosynchronous orbit are powered by solar energy. Huge solar panels producing Kilowatts of electric power have an important aspect to be considered: solar radiation pressure (SRP). Depending on the spacecraft mass it produces an acceleration of the vessel not to be neglected in the equations of motion.

Radiation

Due to the low-thrust of a typical EOR scenario the spacecraft may accumulate some days of dwell time in the van Allen radiation belt. The inner radiation belt extends up to 6,000 km altitude. Every orbit with a periapsis altitude below that value, the spacecraft travels through the region of higher radiation. High concentrations of electrons impact solar cells and reduce efficiency and produced power of the power subsystem. It must be already considered in the platform design for EOR capabilities, e.g. increasing the thickness of the solar-panel protection layer. If not, the expected lifetime of the satellite might be less with respect to a pure chemical transfer.

Eclipses

In case the propulsion system is fed with solar energy the effect of eclipses during the several months lasting low-thrust transfer cannot be neglected. Most of the platforms located in the GEO ring remain fully operational in eclipses in GEO. However, the ratio between eclipse duration and orbit duration in GEO is lower than 5%, allowing the batteries to provide the required power during that period. Instead during the transfer the duration ratio can be higher (up to 30%). Thus, it is required to switch off the engine(s) during eclipses. Of course, the transfer is prolonged by tens of hours.

The number and length of the eclipses strongly depends on the seasons and the initial orbit. An example for a typical GTO to GEO transfer is given in Table 2. The difference between the minimum and the maximum time spent in the shadow of Earth is more than 50%. This example is for initial orbits where the periapsis passage is located in Earth's shadow. In case the initial periapsis would point towards the Sun, the eclipses would be much longer.

Epoch	Duration of Eclipses
March 21st	68 hours
June 21st	58 hours
September 21st	81 hours
December 21st	92 hours

Table 2: Seasonal effect of solar eclipses on a typical GTO-GEO transfer with low-thrust.

Battery

As already mentioned in the paragraph above, one option to bypass the electric propulsion shutdown during eclipses is to equip the spacecraft with enough battery capacity to feed the thrusters. Certainly, the additional batteries required for the electric orbit raising have a negative impact on the mass budget.

Anyway, some battery power is available as it is required to stay operational during eclipses also in the GEO box. This capacity can be used to maintain the EP system in a low-power mode producing a smaller thrust magnitude.

V. PROPULSION

When having an electric orbit raising transfer the thrust magnitude may not necessarily be a profile. Thrusters are typically operated in on/off cycles, with either full throttle or the engine shut off. Thus, the trajectory is split into thrusting arcs and coasting arcs. Besides, there might be frequently periods when the engine is shut down. For example, the thrusters could be turned off for very precise orbit determination once per week.

Other limitations might be related to the firings of the engine itself: minimum or maximum thrusting durations. All the mentioned scheduling aspects of the propulsion subsystem have to be taken into account in the trajectory optimization.

Hybrid Transfer

Some satellite platforms for telecommunication spacecraft have plenty of power because they can be equipped with many payloads. This yields quite large satellite masses in the order of 5-10 metric tons. Even with electric propulsion the total mass of the vehicle might not be reduced below 5 tons. For acceptable transfer times of 6-12 months maximum the provided thrust magnitude of the EP system must be several hundreds of Millinewtons. Unfortunately, the thrusters currently available on the market do not support it.

For such concepts it might be advantageous not to have a full EOR for the transfer, but to use on-board chemical propulsion to boost the initial launch orbit energy. In other words, the transfer trajectory is split into a chemical orbit raising (COR) part followed by an EOR part: a hybrid transfer.

There are some advantages of a hybrid transfer concept. First, the transfer duration is reduced. Thus such a transfer scenario might become feasible whereas pure EOR might not. Next, when using COR the boost in orbit energy happens quite at the beginning of the transfer when the spacecraft still encounters radiation in the van Allen radiation belt. A chemical burn increasing periapsis altitude above the radiation belt would mitigate the radiation threat. Certainly, the larger the chemical portion of the delta-v the lesser the propellant savings of the EP part.

Most challenging is the proper distribution of chemical and electric orbit raising. Typical constraints in the optimization process can be a minimum of radiation or of course a maximum total transfer time, or both.

VI. AOCS

The attitude and orbit control subsystem (AOCS) maintains the orientation of the spacecraft during its whole mission. It typically consists of thrusters, momentum wheels and sensors.

One aspect for example, the star trackers shall not be blinded by Sun, Earth or Moon. When analysing the electric orbit raising transfer, such events can be identified and, if required, taken into account in the optimization process as constraint. Alternatively the objective function can be augmented minimizing such occurrences.

Other restrictions in the attitude might be related to the electric propulsion itself as it should not point in the direction of the Sun. Thus, it is mandatory to constrain the optimizable thrust vector direction.

Special attention must be given to the momentum wheels. Since the spacecraft has large solar arrays, the inertia matrix has large entries. This might yield limitations in the rotation rates and torques of the spacecraft. In the following subsection two examples are introduced for the optimization of slew rates.

Slew Rates

In a typical GTO-GEO transfer the spacecraft experiences two slew rate maxima. One is during the first periapsis passages because of the high eccentricity and the second one is when the circularization of the orbit starts. This is typically the case after about 50% of the transfer. In the first part of the transfer the orbit energy is increased while thrusting almost in velocity direction, whereas in the second part the orbit is circularized keeping the inertial thrust vector direction almost constant. When the circularization starts the second slew rate maximum is reached in the pitch angle (RTN frame). The rates of the yaw angle are usually much smaller and its maximum is reached when the apoapsis radius is at its maximum (mid of the transfer) for very efficient inclination change.

Anyway, in the unconstrained solution the maximum pitch rates are 200 degree per hour (in the first periapsis passages) and 1400 degree per hour (at start of orbit circularization). In the given example the rates shall be minimized to 100 degree per hour in a first step. In a second optimization the slew rates shall be constrained to 50 degree per hour. The influence on the fuel consumption can be found in Table 3. When limiting the rates to 100 degree per hour the impact on the fuel consumption can be neglected. On the other hand, when limiting the rates to only 50 degree per hour the fuel consumption is increasing by 2.1%.

Scenario	Maximum Slew Rate	Propellant Consumption
Unconstrained	1400°/h	100.0%
Max. 100°/h	100°/h	100.06%
Max. 50°/h	50°/h	102.08%

Table 3: Example of constrained slew rate optimization.

In a second example the history of the slew rates is shown for the unconstrained time optimal transfer in Fig. 8. Here, the spacecraft with an initial mass of 1,000 kg is equipped with a 750 mN thruster. The time optimal transfer from 27 degree inclined GTO to the GEO belt lasts 37.8 days. The maximum rates are about 70°/h for inertial yaw angle and slightly more than 200°/h for inertial pitch. In the next optimization run the rates were constrained to an upper bound of 50°/h (see Fig. 9). The transfer duration increases by 0.6 days and the fuel consumption from 125.0 kg to 126.9 kg (plus 1.5%). An optimal solution is achieved within few minutes.

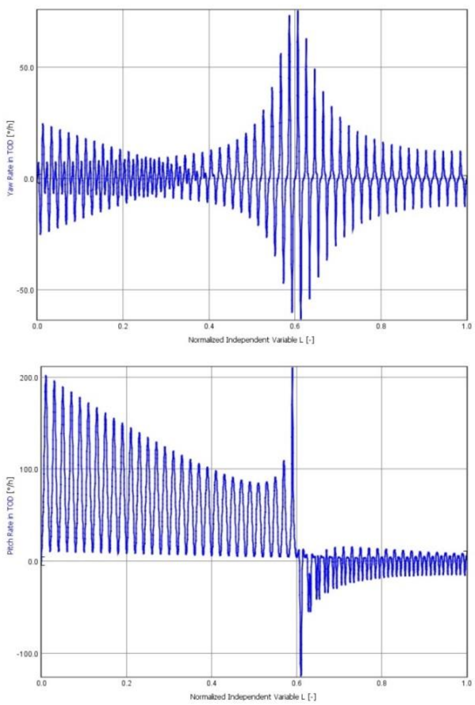

Fig. 8: Unconstrained slew rates (top figure: inertial yaw, bottom figure: inertial pitch) of GTO-GEO transfer with electric orbit raising.

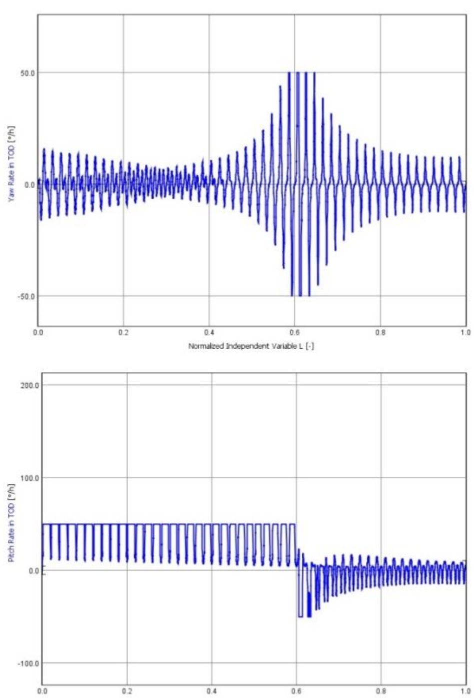

Fig. 9: Inertial yaw rate (top) and pitch rate (bottom) of GTO-GEO transfer with electric orbit raising. Both rates were constrained to 50° per hour.

VII. OPERATIONS

Once the spacecraft is launched successfully, the operational phase starts. Goal of it is to safely bring the satellite from its launch orbit to the desired target location in the GEO ring. Because of the low-thrust character of the transfer, it is proposed to have a periodic operational process, for example on weekly basis. The concept is based on earlier investigations on re-optimization of perturbed GTO-GEO transfers [5]. At the beginning of each cycle, the spacecraft control centre determines the orbit of the satellite. The first time this is done after the launch. Position and velocity as well as mass are taken as input parameters for the optimization software (EOR software). Further, the software considers the reference trajectory, which is computed before the actual mission, and updates the initial state. This is required because of small uncertainties in the initial orbit, for example because of injection errors.

Taking the reference trajectory and its updated initial state, the whole transfer is re-optimized. Then the manoeuvre plan for the next cycle (one week) is extracted and uploaded to the satellite. After one week the next cycle starts from the beginning with the transfer part already done being cut off from the reference trajectory.

In principle, the re-optimization process either optimizes the remaining transfer to the target or just the next cycle. Obviously, in the latter case some margins for the propulsion system are required. In [5] it was shown that an unperturbed GTO-GEO transfer, without any perturbations like third bodies or solar radiation pressure, can be used as reference trajectory for a simulated operational chain process. Taking into account a margin for the propulsion system, the spacecraft could follow the reference trajectory and compensate all disturbances caused by J2, SRP and third bodies, which have been considered in the dynamics of the re-optimization.

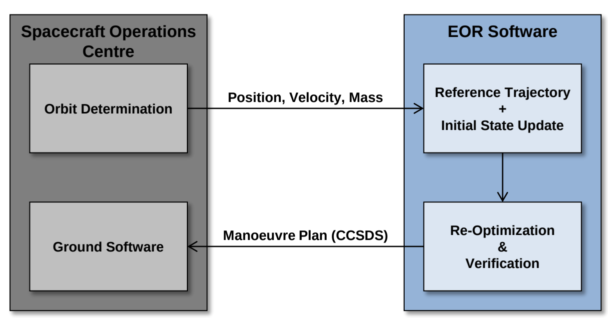

Fig. 10: Concept of an operational chain involving the spacecraft operations centre (left) and the optimization software for EOR transfers (right).

Conjunction Analysis

While raising the orbit it is vital to have a conjunction analysis. Any collision risk of the orbit raising spacecraft with other objects (active/inactive satellites, space stations, upper stages, etc.) and space debris has to be identified for the whole transfer and to be taken into account for the trajectory optimization. Because all those thousands of objects of space debris and manmade objects pose a serious threat for a spacecraft with large solar arrays such as a solar electric propelled satellite.

Especially low Earth orbits but also orbits close to the GEO ring pose certain risks (see also *Crossing the GEO Ring* in section III). For the assessment of the collision risk during the orbit raising the Conjunction Analysis Module (CAM) is used. It identifies any potential collisions of the spacecraft with in-orbit objects. The module uses a threat volume to analyse the close approaches with all objects part of an object catalogue (e.g. from NORAD), which is used as basis for the analysis. Additionally, the catalogue is extended by uncertainty information to assess the collision risk. All objects imposing the highest collision risk are listed, displaying the collision probability and the minimum distance considering nominal Two-Line Elements (TLE) data.

Navigation

For autonomous spacecraft operations the satellite has to know its orbit data. One or multiple of the Global Navigation Satellite Systems (GNSS), for example the Global Positioning System (GPS), are providing this data. Evaluating the link between the EOR vehicle and all satellites of the GNSS constellation yields the coverage. At least four GNSS signals are required for normal operation. When receiving fewer signals the positioning quality becomes worse.

As can be seen in Fig. 11, there are regions with almost no GNSS coverage: the apoapsis region and the periapsis passages of the first orbits. In the given example only one GNSS receiver is mounted on the spacecraft pointing to Earth. Since the Earth blocks all GPS signals there is an outage in low-altitude periapsis passages.

In altitude regions beyond the GNSS constellation the EOR transfer is dominated by outages. Since the transmitters of the GNSS satellites are pointing to Earth with a very narrow beam angle, there is no region in space with good signal quality, at least not outside the constellation sphere.

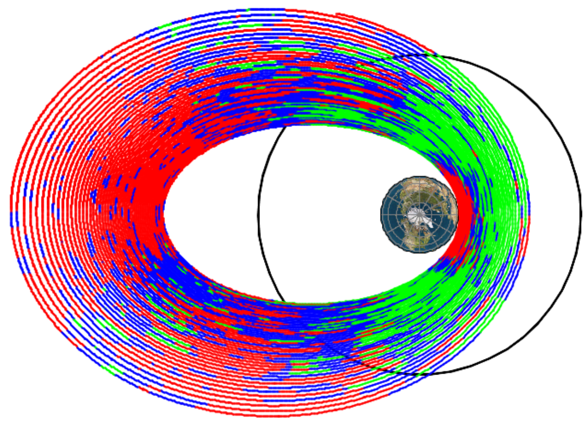

Fig. 11: Example of GNSS coverage for a typical EOR scenario with one GNSS receiver (here: first 50% of GTO-GEO transfer). The colour indicates the number of received signals: >3 (green), 2-3 (blue), <2 (red). The black circle indicates the GPS orbit altitude [6].

Ground Station

When operating the spacecraft during its transfer, frequent ground station contact is required for attitude determination and upload of the updated manoeuvre plan. In principle, one single ground station can be sufficient, especially for 24-h orbits. If not, multiple ground station might be required. A detailed analysis of the reference trajectory results in ground station visibilities and link capacities. Merging ground station analysis and launch orbit analysis (see section II) results in a very sophisticated E2E optimization.

VIII. CONCLUSIONS

The current state-of-the art of electric propulsion orbit raising has been presented based on experience from European EOR activities. Aspects of the injection orbit and the impact on launcher requirements have been discussed. The orbit transfer optimization has been presented addressing objective functions, targeting, eclipses, crossing of the GEO belt, and hybrid transfer. Further dependencies to sub-systems such as power and AOCS have been described. Finally, operational aspects and navigation issues conclude the paper.

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