Current approach in dynamics and control of space systems

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Abstract

Satellites play an important role for humans. Weather satellites, navigational satellites, communications satellites and earth observation satellites serve data, which improve our life. Observation satellites also play an important part in a scientific field: planetary science, sun, extrasolar planet and X-ray astronomy. The orbits are a material factor in the mission design. Their optimized trajectories guarantee success of the mission, lower costs and reduce the amount of space debris. The orbit design differs depending on the goal of the mission.

Earth orbits

For the earth satellites focus is put on the satellite constellations and formation flying on the low, medium, geostationary and highly elliptical Earth orbits (LEO, MEO, GEO and HEO respectively). To reduce the probability of collisions, orbits must be designed with high accuracy and there should be an option to modify trajectories of the flight in space. This modification may lead either to stabilize the orbit or remove a satellite from the orbit when mission is complete. Shape of satellites may induce problems, especially a satellite with flexible appendages in the presence of perturbations has the attitude control problem. Satellite dynamics is nonlinear and depends on poorly known parameters and environmental disturbances; flexible appendages increase complexity. An adaptive robust Lyapunov redesign of feedback linearization is applied to the uncertain nonlinear system to achieve large-angle manoeuvres (Mattei et al., 2014). When a mission is complete or the functioning of a satellite is permanently damaged, there should be still possibility to activate de-orbiting process. Such solution brings a concept of utilization of solar sails for the de-orbiting of satellites. The proper attitude maneuver mechanization is proposed to harvest highest solar drag for Earth orbiting satellites. The maneuver is realized using a to-go quaternion calculated from body fixed frame measurements. The success of the attitude control during the continuous, as well as abrupt maneuvers is shown through simulations in Tekinalp and Atas, 2014.

Highly Elliptical Orbits (HEOs) and Libration Point Orbits (LPOs) offer predominance points for the observation of the Earth, the Sun and the Universe, therefore they are often selected for astrophysics and solar terrestrial missions. LPOs around L1 and L2 ensure a nearly constant geometry for observation and telecoms and are relatively inexpensive to be reached from the Earth, in addition to advantages for thermal system design. On the other hand, HEOs about the Earth guarantee long periods of uninterrupted scientific observation with nearly no background noise from radiations because of long dwelling times at an altitude outside the Earth's radiation belt. No guidelines currently exist for LPO and HEO missions' end-of-life; however, as current and future missions are planned to be placed on these orbits, it is a critical aspect to clear these regions at the end of operations. A detailed analysis of possible disposal strategies for LPO and HEO missions was performed by an ESA/GSP (General Studies Programme).

Proposed end-of-life disposal options exploit the multi-body dynamics in the Earth environment and in the Sun–Earth system perturbed by the effects of solar radiation, the Earth potential and atmospheric drag. The options analyzed are Earth re-entry, or injection into a graveyard orbit for HEOs, while spacecraft on LPOs can be disposed through an Earth re-entry, or can be injected onto trajectories towards a Moon impact, or towards the inner or the outer solar system, by means of the enhancement of solar radiation pressure with some deployable light reflective surfaces or delta-v manoeuvres. Five ESA missions are selected as scenarios: Herschel (past), GAIA (current), SOHO (current) as LPOs, and INTEGRAL (current) and XMM-Newton (current) as HEOs (Canalias et al., 2004). For each mission the disposal strategies are analyzed, in terms of manoeuvre sequences, optimal window for the disposal manoeuvre, in terms of time of flight and disposal characteristics, such as re-entry conditions or the hyperbolic excess velocity at arrival in case of a Moon impact. Next, a high accuracy approach is used for validating the optimized trajectories. At the end, a trade-off is made considering technical feasibility in terms of the available on-board resources and Δv requirements. The sustainability context and the collision probability in the protected regions is considered as well. Table 1 summarizes main mission constraints for the selected missions. The available fuel enables a trade-off analysis between the extension of the mission and the feasibility of reliable disposal strategies. All kinds of manoeuvres are influenced by the instrument lifetime (e.g. battery, transponder switches, reaction wheels) and components failures. Moreover, the disposal trajectory should be designed considering the pointing constraints, due to the thermal and the power subsystem or payload requirements. Other kinds of disposal strategies that exploit non-gravitational perturbations such as the effect of solar radiation pressure are constrained by the maximum area-to-mass achievable with the current spacecraft configuration or with minor changes to the operational configuration (Colombo et al., 2014).

Table 1. Mission constrains for the selected missions. Modified from Canalias et al., 2004.

Mission	Type	Dry mass [kg]	Available fuel [kg] (date)	Equivalent Δv [m/s]	Failures	Consumption per year [kg]	Pointing constraints
SOHO	LPO	1602	108-111 (31/12/2016)	140.8-144.59 (31/12/2016)	Loss of gyroscopes	2-1	4.5° < SEV angle < 32°
Herschel	LPO	2800	180 (1/1/2013)	130 (1/1/2013)	Helium finished	4	Constraints due to thermal management and star trackers operation, Sunshade pointing towards the Sun.
Gaia	LPO	1392	5 (EOL)	10 (EOL)	N/A	N/A	Spin axis precessing with an angle 45°around s/c-Sun line
INTEGRAL	HEO	3414	90 (1/6/2013)	59.99	-	8	Telescope never points closer than 15° from the Sun
XMM	HEO	3234	47 (1/6/2013)	33.26	Reaction wheel degradation	6	Telescope never points closer than 15° from the Sun

Non-Earth orbits

For non-Earth orbit missions optimizations for Moon, Mars and Near-Earth Objects (NEOs) directions are studied. Near-Earth Objects (NEOs) are of contemporary, since asteroids and comets hold the key clues to understanding the formation of the Solar System and the origin of the planets, and also have a speculated wealth of material resources. The exploitation of these resources has been discussed as a means to lower the cost of future space endeavors. Recently, a new family of so-called Easily Retrievable Objects has been presented. EROs can be transported from accessible heliocentric orbits into the Earth's neighborhood at affordable costs. The trajectory options for low-cost missions to EROs are explored in three different ways: (1) by analytic and numerical search methods, (2) by hybrid optimization algorithms including global search and (3) by local search to find the optimal rendezvous opportunities. A wide variety of multi-impulse transfer, gravity-assist transfer and gravity-assist transfer with mid-maneuvers to obtain low launch energy and rendezvous trajectories to EROs are considered. Detailed study of the best rendezvous opportunities for currently known EROs in the next 20 years are reported together with the relevant mission costs in Qiao and McInnes, 2014.

Asteroids

The next destination of space exploration are asteroids as they represent and unparalleled historical archive fundamental for our understanding of the origin of the Solar System. They are located mainly in the Asteroid Belt, between 2 to 3.5 AU and in the stable points of the Sun-Jupiter system. Some of asteroids orbit in the inner Solar System and can come closer to the Earth posing a threat to our home planet. In order to mitigate the risk of dangerous close encounters we need to find and study them for developing planetary defense strategies. Another interesting aspect is that some of these small bodies turn out to be even more accessible than the Moon in terms of deltaV requirements. For this reason Near Earth Asteroids have become the natural targets for the first manned missions venturing outside the Earth-Moon system in preparation for the exploration of Mars. Small bodies also represent deposits of raw materials already floating in space, especially rare metals and water ice. These can be used to foster future exploration and human presence in the Solar System. In order to fulfill any of the previous objectives it is required to send spacecraft capable to manoeuvre in the vicinity of asteroids (Turconi et al., 2014).

In the past two decades many milestones have been achieved, from the first flybys (Galileo, *NASA/ESA, 1991)* to missions featuring stable orbits and landing on the surface (NEAR-Shoemaker, *NASA, 2000)* or getting dust samples back to the Earth (Hayabusa, *JAXA, 2005)*. However, most of operations – with the exceptions of landing trajectories – have always been performed with a ground-in-the-loop approach. They require an extensive use of ground station time to carry through the mission with a substantial impact in overall costs. Moreover, with all the knowledge about the orbital dynamics of the spacecraft being on the ground, the operations rely on preplanned commands and mission profiles are constrained by communications delays. The technology for autonomous navigation in the proximity of asteroids is already available. What is missing is a way of modeling on-board the non-uniform gravitational potential of these small bodies and a guidance and control strategy that can rely on such inherently simple and approximate gravitational models. The irregular shape of asteroids creates an inhomogeneous gravitational field in which the spacecraft generally moves along non-closed and unstable trajectories. In order to guide a spacecraft in the vicinity of an asteroid, elementary characteristics of the small body need to be known: shape, mass and spin rate. Under optimistic conditions, it is possible to have an idea of those parameters from the ground, especially when dealing with bigger asteroids. For most of the bodies it is possible to perform light-curve analysis and albedo (Torppa et al., 2003; Carry et al., 2012). In case of close encounters with the Earth, good shape models are obtained via radar imaging (Hudson and Ostro 1989). However, most of the times, the accuracy of these data has to be significantly improved in order to carry out the mission successfully. Much of the detailed characterization work is therefore based on in-situ observations. Mass is counted thanks to flybys taking place before orbit insertion. Imaging the asteroid under various lighting condition enables the construction of a shape model and the determination of the spin rate (Miller et al., 2002; Bhaskaran et al., 2011). Knowing the mass and the shape, a density estimate can be computed. Using this model of a spinning shape with constant density, trajectory analysis for the following phases of the mission can be performed. Further perturbations that may be detected are used to improve the model. A better representation of the gravitational potential yields a more accurate estimation of the spacecraft trajectory and enables the ground controllers to safely guide the satellite at lower altitudes above the surface. All these steps of characterization actually rely on the measurement of relative position and velocity between the spacecraft and the asteroid. These measurements are the result of Range-Doppler campaigns accomplished using the large antennas of the Deep Space Network (DSN) (Yeomans et al., 2000). A polyhedral shape model (Werner and Scheeres, 1996), is created thanks to high-resolution images (Gaskell et al., 2008) and LIDAR measurements, combined with the trajectory of the spacecraft measured from the ground. The models of the asteroid are assembled on the ground as well, so the satellite simply executes, at prescribed times, manoeuvring commands previously prepared on the ground and has no knowledge of the dynamical environment in which it is immersed. Given the delay in communications caused by the distance between the spacecraft and the Earth, everything has to be carefully planned in advance and constantly monitored. In recent years the developments of the AutoNav software and in particular of its OBIRON component (On-board *Image Registration and Optical Navigation)* has demonstrated the feasibility of autonomous optical navigation in vicinity of asteroids.

The Moon

In recent years, several countries have shown an increasing interest toward both automatic and manned lunar missions. Some challenging projects have the purpose of building a lunar base for future interplanetary missions. The development of a safe and reliable guidance algorithm for automatic lunar landing with a very modest residual velocity represents a very important issue for establishing a relevant connection between the Earth and the Moon surface. The bulk of the work on lunar landing dates back to the early years of the Apollo program. Recently, several authors contributed in the effort of developing a modern, reliable guidance algorithm for soft lunar landing. In particular, Chomel and Bishop proposed a targeting algorithm based on generating a two-dimensional trajectory that serves as a reference for three-dimensional guidance (Chomel and Bishop, 2009). Lee assumed a continuous time-varying thrust profile and referred to a nominal trajectory similar to that used in the Apollo 11 mission, capable of guaranteeing acceptable fuel consumption (Lee, 2011), together with safe touchdown conditions. In this year Cecchetti et al. proposed a new general purpose Neighbouring Optimal Guidance (NOG) algorithm, based on the general theory of NOG (Hull, 2003) overcoming the occurrence of singularities and the lack of an efficient law for the iterative real-time update of the time of flight.

Space debris

The possibility of collision of the artificial satellites with the Earth orbiting object is rapidly increasing. It is estimated that there is about 5500 tons of debris in Earth orbit due to upper stages of launch vehicles, abandoned satellites, etc. In addition, the number of objects is estimated to grow exponentially at a rate of 5% every year (FP7-SPACE, 2010). Today the number of that space debris fragments is higher than some hundred million, as depicted in Table 2. Space objects may start colliding into each other, generating fragments and thus increasing the already high number of debris, towards a sort of „collision reaction", known also as Kessler Syndrome. It is hard not to think about their possible influence on other future satellites, or on some place in the Earth when falling down. Concerns are about loss of image, loss of money, interruption of services, damages to goods or even injury to persons. United States Space Command is tracking thousands of debris with diameters larger than 10 cm (Klinkarad, 1993). In particular, the Low Earth Orbit altitudes are especially crowded, and the risk of collision is rapidly increasing. The new procedures set forth by NASA requires the spacecraft in low Earth orbit, shall remain in region no more than 25 years after the end of mission (Johnson and Stansbery, 2010). A similar regulation is about to put into force in the European Union as well (Anselmo et al., 2004). The removal of satellites after their operative life nowadays is not an option, but a strong requirement.

Normally, sufficient amount of fuel with working thrusters are necessary to de-orbit a satellite, just as the thrusters are needed for placing them to their orbit.

Table 2. Estimated Orbital Population of space debris (Tolle et al., 2014)

Size	Number	%Mass
> 10 cm	> 30,000	99.93
1-10 cm	> 750,000	0.035
< 1 cm	>166,000,000	0.035
Total	> 166,000,000	6,000 tones

However, this requires carrying a large amount of additional de-orbiting fuel with the satellite. The idea is to carry a light weight solar sail to orbit, and deploy it at the end of life for the purposes of de-orbiting the satellite. The Aerospace Engineering Department of METU is involved in an FP7 collaborative project to design, build and launch a CubeSat to demonstrate the functioning of solar sails for de-orbiting satellites (European Union FP7 project, Project no: 263248, 2010). The solar sail technology for interplanetary travel was also successfully shown in 2010 on Ikarus satellite. The sail types to be used, as well deployment mechanisms and sail folding have been thoroughly investigated. However, it is not sufficient to deploy the sail once in orbit, but to properly orient the sail with respect to the sun direction to get sufficient solar drag from the sun for de-orbiting. Thus, an attitude control has to be active to continuously orient the sail (Tekinalp and Atas, 2014).

The avoidance of collisions implies continuous control over all the „flying objects" to predict their orbit or trajectory, to compute them in order to find out whether some collisions would be possible and, in case, to perform manoeuvres with the spacecraft, to change orbits to decrease impact probability. This means an investment of conspicuous amounts of public money for the surveillance infrastructures, and, from a commercial space operation point of view, a consumption of propellant, which affect the operative life of the spacecraft and, therefore, shortage of services and loss of revenues. For the mitigation of those risks, Active Debris Removal is one of current hot spots in space research, necessary for space exploitation durability. Different techniques have been proposed for this challenging task, among them the use of tow-tethers and throw-nets seem promising. That opens new challenges for Guidance Navigation and Control (GNC) design, especially whenever flexible connections are involved (Benvenuta and Lavagna, 2014).

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