Master thesis : Development of an interplanetary orbital propagator

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Abstract

Humanity's interest is to explore the unknown, discover new worlds, find traces of life and understand the creation of our solar system. Interplanetary orbital propagators are developed in that context. They consist of tools allowing us to predict the spacecraft trajectories in our solar system. Propagators are used to support and design real-world missions. The main objective of the thesis is to develop a high-fidelity orbital propagator for interplanetary space missions. It is implemented in the MATLAB environment. The propagator should include accurate ephemerides of our solar system and high-fidelity dynamical models. More specifically, perturbations due to the non-sphericity of the attractors, the solar radiation pressure, eclipse models and point mass gravity attractions should be taken into account. Lambert's problem should also be included in the propagator. The general mission analysis tool GMAT R2017a developed by NASA is used to validate the results. The semi-analytic method VSOP87 based on integrated ephemeris DE200 is implemented in the propagator. The semi-analytic method VSOP87 takes into account the secular and periodic perturbations between the considered planet and the other celestial bodies (planets and asteroids). The errors represent respectively 0.003% and 0.01% of the semi-major axes of the inner planets and of the outer planets. The accuracy of the point mass gravity and solar radiation pressure models is directly linked to the accuracy of planetary ephemerides. The J_2 perturbation is independent of the ephemerides. It is validated with the S3L propagator developed by the University of Liège. The dominant point mass gravity perturbations within the spheres of influence come from the moons. The close proximity between the moons and the spacecraft is the key parameter explaining this fact. Only Earth's moon called Moon is modelled in the propagator. The Sun is the second contributor due its large gravitational parameter. Propagation lasting one day have been performed each day on January 2000 on an orbit escaping the Earth. The Moon changes the final spacecraft position by 75 km on average compared to the two-body trajectory, whereas the Sun changes it by 31 km on average compared to the two-body trajectory. These distances represent the errors with respect to GMAT the point mass gravity models included in the propagator should compensate. It is expected that these values approach 0. The averages of errors in final spacecraft positions for the Moon and Sun point mass gravity models reduce respectively to 1 km and 4 m. The approximate ephemerides of the Moon come from Astronomical Almanac which provides less accurate ephemerides than the VSOP87 planetary theory. As a result, the Moon point mass model gives rise to less accurate results. However, it reduces the error from 75 km to 1 km on average. In contrast, the Sun point mass gravity model reduces the error from 31 km to 4 m on average. The planetary point mass gravity perturbations are included to obtain high-fidelity propagation within the spheres of influence. In contrast, the planetary point mass perturbations change significantly the spacecraft trajectories outside the spheres of influence. A cannonball model and the eclipse model proposed in Curtis are chosen to model the solar radiation pressure perturbation. The eclipse model included in the propagator overestimates the shadow region compared to the conical eclipse model included in GMAT. However, for Mercury, which is the planet located the closest to the Sun, the error on the final position decreases from 11.23 m to 0.59 m by taking into the eclipse model. The distance, 11.23 m, is found considering that the spacecraft is in full illumination during the one day of propagation around Mercury. Eventually, the solar radiation pressure model is validated with long propagation. The case study is an interplanetary trajectory between the Earth and Mars in 309 days. After 300 days of propagation, the error compared to GMAT is 11 km which represents an error of 0.007 %. Lambert's problem coupled with the perturbation models is used to recreate the four first years of the Cassini-Huygens mission. The data of the reference curve come from JPL Horizons Ephemeris System. The mean relative error between the curve of the Cassini-Huygens mission and the propagator is 1.95%.