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Book of Abstracts

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Orbit Determination and Prediction Techniques #1 / 12

A fast an efficient algorithm for the computation of distant retrograde orbits

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An algorithm for the computation of distant retrograde orbits is presented. It is based on the computation of an approximate analytical solution of the restricted three body problem in the Hill problem approximation that provides accurate estimations of two basic design parameters. Notably, these parameters can be used for the computation of initial conditions of orbits that are periodic on average, and almost periodic in the original problem. Following application of iterative differential corrections results in the initial conditions and period of a true periodic orbit with the characteristics fixed by the design parameters.

The analytical solution consist roughly of a drifting ellipse, whose guiding center moves around the primary with long-period oscillations, and in which the linear growing of the phase of the satellite is modulated with long-period variations. The analytical solution splits in two parts of different nature. The first one provides the periodic corrections needed for converting osculating elements into the mean ones that describe the long-term evolution of the dynamics. The second part of the analytical solution gives the time history of the mean elements in the form of five Lindstedt series, which are needed for describing:
- the time scale in which the Lindstedt series evolve (1 series)
- the time evolution of the guiding center of the reference ellipse (2 series)
- the linear frequency with which the satellite evolves, on average (1 series)
- the long-period modulation of the phase of the satellite (1 series)

The use of the algorithm is illustrated with different examples, ranging from the typical case of 1:1-resonant distant retrograde orbits, in which the satellite remains always far away enough from the primary, to the challenging case of higher order resonances in which the amplitude of the libration of the guiding center of the orbit can take the satellite much closer to the primary.

Summary:

Interplanetary Flight and Non-Earth Orbits #1 / 19

A technique for designing Earth-Mars low-thrust transfers culminating in ballistic capture

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It is possible to design heliocentric transfers to Mars culminating in ballistic capture and, with an impulsive-thrust strategy, these have already been studied, but were found to be less fuel-efficient and longer-lasting than Hohmann transfers. The objective of the present work is to investigate the characteristics of Earth-Mars low-thrust transfers to ballistic capture.

Small spacecraft are very mass- and power-constrained, so orbit transfers are challenging for them, especially to interplanetary destinations. To try and shift this paradigm, the study was carried out

assuming the spacecraft to be a 16-unit CubeSat. In addition, to improve the validity of the results, ballistic capture was designed using a model that included many perturbing forces, namely third-body perturbations, solar radiation pressure and non-spherical gravity.

Some capture orbits were selected, each with a different arrival date at Mars, and targeted from Earth, on multiple departure dates. It was found that if the spacecraft is given enough time, the low-thrust strategy requires roughly the same fuel regardless of Earth departure or Mars arrival dates. In addition, terminating a low-thrust transfer to Mars in ballistic capture does not carry additional costs, when compared to simply rendezvousing with the planet. With the assumed spacecraft and departure conditions, only around 5 kg of propellant are required to reach Mars and get ballistically captured. Nevertheless, the spacecraft needs to fly for at least 3.5 years, which can be too long for a CubeSat.

Summary:

Optimization and Dynamics #2 / 31

AO-Car: Transfer of Space Technology to Autonomous Driving with the use of WORHP

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Online optimization and trajectory planning are key aspects of autonomous deep space missions. Taking into account individual target criteria, such as time or energy optimality, any spacecraft maneuver can be traced back to a general problem definition of the form “move the spacecraft from its initial state to a desired final state, while considering a dynamic model and avoiding collisions”. This corresponds to an optimal control problem (OCP) and can be solved using WORHP, the ESA solver for non-linear programming (NLP). More specifically, the OCP is transcribed into an NLP by time discretization and the corresponding optimal trajectory - including control variables - is calculated by sequential quadratic programming. In order to obtain a highly efficient solution algorithm, the naturally occurring sparsity of the Jacobian and the Hessian is exploited.

The effectiveness of this approach has already been demonstrated in several DLR projects, such as the deep space missions KaNaRiA and EnEx-CAUSE. In order to make such results immediately available for terrestrial applications, a transfer to current scientific questions is appropriate. Moreover, the transfer would provide a test platform and increase public acceptance. Conversely, the knowledge gained from terrestrial testing can help planning more detailed space missions.

In this work, the DLR project AO-Car for controlling an autonomous vehicle in road traffic is presented as such a transfer. The concept, originally developed in the context of KaNaRiA for trajectory planning and control, is successfully implemented on a research vehicle, a VW Passat. The vehicle is able to explore a parking area autonomously, to identify free parking spaces and to perform a parking maneuver. During the exploration, suddenly appearing objects are recognized. Depending on the scene, a collision avoidance trajectory is computed or an emergency stop is performed. The method presented is based on WORHP and offers a uniform framework for optimal driving maneuvers. It is highly flexible, as reaction speed and passenger comfort can be easily balanced by adaptive weighting of target criteria.

Summary:

Ascent #1 / 53

ASTOS 9.3 - Multibody Feature for Simulations of Flexible Launcher Dynamics

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This work presents the new set of equations of motion incorporated in ASTOS release 9.3.

The new implemented feature, based on DCAP multibody software, provides the building blocks to simulate a complete launcher scenario considering vehicle flexibility, sloshing effects, stages separation, engine pressure oscillations and complex aerodynamic loads distribution.
The interaction between those phenomenon and the ascent flight control logic could turn the entire vehicle dynamics unstable. The new functionalities allow ASTOS software to simulate and predict such catastrophic scenarios.

The multibody equations of motion feature let DCAP compute the entire system dynamics while ASTOS provides all the external forces such as aerodynamics, gravity accelerations and actuator output.
Five major features are organically embedded in the new MBS capabilities:

- a linear Euler-Bernoulli beam allows to approximate the flexibility and the frequency content of each launcher structure section;
- a spring-mass system model allows to simulate the propellant sloshing effect in the launcher tanks;
- the transition logic allows to model the separation process during the jettison of exhausted stages by changing the multibody topology;
- separation devices, such as hard-stops and clamp bands, can be employed to reproduce a more realistic scenario during stages disconnections;
- engine pressure oscillations effect can be accounted by providing a disturbance loads in time or frequency domain;

This work details on each of the above mentioned new functionalities, showing user input and results taken from typical example scenarios.

Summary:

Optimization and Dynamics #1 / 13

Adaptive Pareto Front Sampling Based on Parametric Sensitivity Analysis in a Bi-Objective Setting

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In order to solve non-linear multiobjective optimization problems, one usually solves multiple scalarized subproblems. This provides a discrete approximation of the Pareto front which gives useful information for the decision maker who, in praxis, has to select one single solution. If the desired solution is not part of the precomputed discrete approximation one needs to apply interpolation techniques.
This contribution shows a method which uses information from parametric sensitivity analysis of the scalarized subproblems in order to choose the stepsize between samples adaptively to obtain a

better interpolation between precomputed solutions. The problems are solved with the NLP solver WORHP which provides sensitivity information in an efficient way by reusing the factorization of the KKT matrix of the last optimization iteration. We show the basic functionality of the presented method by applying it to several bi-objective optimization problems. The method can also be used for more than two objectives if one can identify the neighboring precomputed points which are then used for interpolation.

Summary:

On-orbit servicing and proximity #2 / 27

Analysis of a Rendezvous Mission in Non-Keplerian Orbit using Electric Propulsion

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This paper presents the analysis of a low-thrust rendezvous mission to a target non-Keplerian orbit of the circular restricted three body problem (CR3BP) in the Earth-Moon system. The dynamical characteristics of this system are revisited, and some non-Keplerian orbits (e.g., L1 halo orbits, NRO and L2 halo orbits) have been simulated to study their suitability for a rendezvous mission. Starting from analytical approximations, a shooting method has been used for the numerical description of these orbits. Afterwards, the different monodromy matrices related to the integrated non-Keplerian orbits has been studied to analyse their stability and to describe stable and unstable manifolds of the orbit, i.e., ballistic trajectories that can be covered by a spacecraft without any propellant usage.

The design of a low-thrust rendezvous mission in a non-keplerian orbit is approached as an optimal control problem, in which the solution is the thrust magnitude and direction along the path. The propellant consumption has been set as the objective function to be minimized, and the trajectory is subjected to a set of constraints ranging from thrust limitations and time requirements specified for each mission. The problem to be tackled is a rendezvous mission to a specific target in a L2 Halo orbit. Hermite-Simpson collocation method has been used for the numerical description of the dynamical constraints of the system. Then, the problem has been solved numerically with IPOPT (Interior Point OPTimizer). An unpowered trajectory integrated from the problem initial conditions is used for the solver initialization. The rendezvous mission implementation and the interface with that solver have been developed in Matlab.

Three different cases of rendezvous at the target L2 Halo orbit have been studied: 1) rendezvous from a nearby halo manifold, 2) rendezvous from a close halo orbit and 3) phasing manoeuvre starting from the same halo as the target. All the cases have shown to be feasible for being performed with the use of an Electric Propulsion thruster. The final optimal trajectories for those cases are presented. In addition, it has been shown that propellant consumption can be greatly reduced if the stability properties of the targeted orbits are exploited.

Summary:

Loitering, Orbiting #1 / 16

Are we really covering up the whole sky?

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As it's known that our Sun rises always in the east and sets in the west, similarly, all the stars in the sky also rise in the east and set in the west. But day time, due to sun's brightness, we would not able to see the stars in the sky. Apart from this, there are many other parameters that restrict the observation these celestial objects. A few of those are : Time period of rising and setting of an object, latitude, Magnitude of the object, altitude of the location of observation and limitation of the telescope covering the sky in all angular directions. Then the question arises, can we really covering up the complete sky in all angular positions for observation with the availability of telescopes and also with respect to its situated positions for observations? There is always the possibility that we could miss some part of the sky. This aspect is the motivation that prompted to take this topic of research. This is just a questionnaire to start up research. Research has to be yet startup, with the small experiments plot. To carry out this research the sample is taken first for Chennai and then applied for all other available Indian telescopes. This work further can be extended to all other telescopes situated around the world with different geographic locations and plotting observational coverages over the night sky and looking for the list of possibility of unobserved part of the night sky with the sufficient exposure and resolution.

Summary:

Interplanetary Flight and Non-Earth Orbits #2 / 37

Astrodynamics techniques for missions towards Earth trailing or leading heliocentric orbits

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Recent mission and system studies conducted for the European Space Agency have involved the design of transfers targeting Earth-trailing or Earth-leading heliocentric operational orbits, in a 1:1 resonance with Earth.
Airbus is currently leading two such studies on behalf of the European Space Agency: the Lagrange (Space Weather) mission targeting the Sun-Earth L5 Lagrange Point, and the LISA (Laser Interferometer Space Antenna) constellation of three satellites, selected as the third large-class mission of ESA's Cosmic Vision Programme, and whose operational configuration consists in a heliocentric triangular cartwheel formation.
As no spacecraft has ever flown to these destinations, the presentation will focus on the mission analysis techniques that have been used to address this very special class of interplanetary missions, characterised by some unique features and challenges.
In particular, the available injection and transfer strategies will be reviewed and thoroughly traded against the mission objectives and constraints: these include classical direct ascent strategies, but also low-energy escape options via the Sun-Earth L1 or L2 points, as well as advanced strategies involving Earth and/or Lunar Gravity Assists. The benefits and challenges of transfers augmented with Solar Electric Propulsion will be highlighted. A special attention will also be brought to the implications of the selected Launch Vehicle, in particular if characterised by a limited range of available declinations (DLA), the resulting seasonal variation of the transfer problem and the impact of the launch windows definition. Finally, orbital perturbations on station and disposal strategies will be addressed.
While the presentation will focus on the astrodynamics techniques for the relatively generic problem of flying a spacecraft into heliocentric space, the application to the Lagrange and LISA missions will be presented as study cases.

Summary:

Presentation of special techniques for mission design towards Earth trailing or leading heliocentric orbits at 1au.

Open Source Tools and Smart Computing #1 / 62

Astrodynamics.jl: A Julia-based Open Source Framework for Orbital Mechanics

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Intelligent Tools and Assistants #1 / 32

Astrodynamics.jl: A Julia-based Open Source Framework for Orbital Mechanics

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This paper presents Astrodynamics.jl an open source framework for high-performance, interactive orbital mechanics implemented in the Julia programming language. The implementation language was chosen based on previous work that demonstrated that it is possible to bridge the performance gap between compiled and dynamic programming languages by using Just-in-time (JIT) compilation and was presented at the last edition of ICATT in 2016 [1].

We demonstrate the capabilities of the Astrodynamics.jl framework based on real-world applications, such as lunar transfer trajectory optimisation, and discuss how the unique properties of the Julia language enable us to develop human-friendly, easy-to-understand interfaces while at the same time achieving competitive performance in comparison to established libraries such as Orekit or AstroPy.

The Astrodynamics.jl open source project is part of a larger initiative to further the use of open source software in astrodynamics and space science in general. We want to highlight how the umbrella organisations OpenAstronomy and OpenAstrodynamics promote knowledge transfer, interoperability and use of standards across open source projects, communities, and programming language barriers.

[1] Eichhorn, H., Cano, J.L., McLean, F. et al. CEAS Space J (2018) 10: 115. https://doi.org/10.1007/s12567-017-0170-8

Summary:

This paper presents Astrodynamics.jl an open source framework for high-performance, interactive orbital mechanics implemented in the Julia programming language. We demonstrate its capabilities based on real-world applications, such as lunar transfer trajectory optimisation, and discuss how the unique properties of the Julia language enable us to develop human-friendly, easy-to-understand interfaces while at the same time achieving competitive performance in comparison to established libraries such as Orekit or AstroPy.

Low Thrust #3 / 34

Automated optimization of low-thrust trajectories

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Modern space missions often require a large velocity increment, which leads to the need to use main electric propulsion systems with a high value of the specific impulse to reduce the mass of the active propellant and increase the mass of spacecraft in target orbits.
To improve the efficiency of space transportation operations, optimization of the low-thrust trajectories is required. Optimization of such trajectories is a complex problem, due to a number of reasons, including the limited size of the domain of existence of solutions, the possibility of the existence of many optimal solutions, the nonsmooth dependence of boundary conditions on control parameters, the complexity of the mathematical model of optimal motion, the high sensitivity of optimal trajectories to variation in control parameters. The above factors significantly complicate the process of optimizing the trajectories of spacecraft with electric propulsion system and increase the relevance of the problem of automating the calculation of optimal trajectories.
Probably, there is no universal method for automating the solution of the problem, but a number of computational approaches and methods have been found that make it possible to achieve a high degree of automation of many practical problems within the framework of an indirect approach to solving the problem of optimal control based on the application of the maximum principle. These approaches and methods include the continuation method for automating the choice of the initial approximation, the high-precision calculation of the derivatives using the complex step method or the dual numbers in solving boundary value problems and in calculating the right-hand sides of differential equations for conjugate variables, smoothing discontinuous control.
A method for automated optimizing the trajectories of a spacecraft with electric propulsion system is developed, which includes the use of the above techniques and the sequence of solving problems of optimization of power-limited, minimum thrust acceleration, minimum thrust, and constant exhaust velocity trajectories, ensuring smooth continuation from one subproblem to another and with the diagnosis of existence solution of the problem.
The study was carried out with the financial support of the Russian Science Foundation (project No. 16-19-10429).

Summary:

Loitering, Orbiting #2 / 59

Basilisk: A Flexible, Scalable and Modular Astrodynamics Simulation Framework

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Verification and Validation Methods #1 / 17

Basilisk: A Flexible, Scalable and Modular Astrodynamics Simulation Framework

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The plethora of spacecraft simulation software tools is an indispensable part of modern spacecraft design processes. The continual increase in complexity of spacecraft missions and maneuver design, dynamical and kinematic design verification and post-launch telemetry analysis all heavily rely on software simulation tools. This simulation ability provides engineers with the tools to increase the quality of design and testing, by reducing cost and duration of development. For example proposed changes to a mission's configuration, parameter tuning or in-flight anomalies may be explored via

Monte Carlo simulation. Additionally, hardware in the loop testing (HWIL) allows for verification and validation of the spacecraft hardware and software systems in a controlled laboratory environment. Such HWIL testing can expose technical faults and system integration problems saving considerable project financial and personnel resources before launching the system to space.

Basilisk is an astrodynamics framework that simulates complex spacecraft systems in the space environment. The types of missions, which Basilisk can be used to simulate, lay on a spectrum with earth orbiting cubesats at one end and interplanetary probes and spacecraft constellations at the other. The hallmark of the Basilisk framework is its highly modular system architecture. Modular design has been the guiding principle throughout Basilisk's development. The result is that Basilisk implements only two core system components, the Basilisk message exchange and Basilisk simulation controller. These two components are the only components required to begin building a Basilisk simulation scenario.

While many simulation tools possess overlapping features with Basilisk, none others possess the unique characteristics of Basilisk. These characteristics are that Basilisk is a highly modular, Python user-friendly, open-source simulation framework that provides sufficiently accurate (fidelity is configurable) coupled vehicle position and attitude dynamics, along with optional structural flexing, imbalanced momentum exchange device and fuel slosh dynamics, with at least a 365 times speedup (one mission year in one compute time day). Furthermore, Basilisk is equally well employed during early mission design phases as it is later on during detailed design phases and further in post-launch telemetry analysis and spacecraft command sequence validation.

Beyond the two required Basilisk components, the user constructs a simulation scenario by including any number of Basilisk Modules relevant to their specific mission. A Basilisk Module is a stand-alone code. Modules typically implement a specific model (E.g. an actuator, sensor, and dynamics model) or self-contained logic (E.g. translating a control torque to a RW command voltage). The two core Basilisk components and most modules are written in C++ to allow for object-oriented development and fast execution speed. However, Basilisk Modules can also be developed using Python, for easy and rapid prototyping, C (to allow flight software modules to be easily ported directly to flight targets) and Fortran (to accommodate legacy space environment models).

Whereas Basilisk modules are developed in a number of computing languages, Basilisk users interact and develop simulation scenarios using the Python programming language. The Python interface to the C/C++/Fortran layer relies on the Simplified Wrapper and Interface Generator (SWIG) library. SWIG is a cross-platform, open-source library that generates Python wrappers (amongst many other languages) for compiled C/C++ code. With the Python user layer and Basilisk's cross-platform development, (currently developed for MacOS, Windows, and Linux systems) the modularity results in no compile time or run time dependencies between one module and another. This modularity provides the engineer with the ability rapidly develop and reconfigure their simulation scenario in the Python language.

This modularity is achieved by the unique implementation of the Basilisk message exchange. A Basilisk module may read input messages and write output messages to the Basilisk messaging exchange. Decoupling of the data flow between modules is achieved via the message exchange. The message exchange acts as a message broker for a Basilisk simulation. Modules read and write data structures referred to as 'messages' to the message exchange. A message is defined as having a name and payload data structure (typically a C/C++ struct). The Basilisk messaging exchange manages the trafficking of messages and employs a publisher-subscriber message passing nomenclature. A single module may read and write any number of messages. A module that requires data output from another modules subscribes to the message by registering the 'subscription', to the specific message name, with the messaging exchange. Conversely, a module that desires to output data for other modules registers the 'publication' of that payload data associated with a specific message name. The messaging exchange then maintains the messages read and written by all modules and the network of publishing and subscribing modules.

This paper will be presented in three main sections. The first section will survey the current context of state-of-the-art commercial-off-the-shelf (COTS) and government-off-the-shelf (GOTS) spacecraft simulation software and provide a context for the Basilisk framework. Each of these COTS/GOTS systems posses' unique feature sets. These unique feature sets predispose each tool excel when applied to different classes of spacecraft system analysis, by different engineers teams.

The second section will give a detailed account of the aforementioned novel Basilisk system architecture, which allows for the rapid development of a simulation for of a wide variety of complex spacecraft systems. The key architectural components to be discussed are the Basilisk message exchange, the Basilisk simulation controller and the anatomy of a typical Basilisk module. A number of archetypal Basilisk simulation configurations will be presented from the perspective of the end user engineer to illustrate the functionality of this design.

Finally this paper will discuss and provide demonstration of the following core Basilisk capabilities:
- Controlling simulation fidelity and speed of any module in a simulation by dynamically adjusting integration rates for each module during simulation execution.
- Basilisk multi-processing Monte Carlo tools.
- Data logging, analysis and visualization tools for large (multi-gigabyte) data sets.
- Execution of a Basilisk simulation distributed across multiple computing devices.
- 3D Visualization of a simulation in real-time or after the simulation has complete via playback.

Summary:

Clean Space and Environment Modelling #1 / 11

Cataloguing performance assessment method of SST sensor networks

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The European Union is now developing a federated SST system composed of existing sensors and operations centres in Europe through the EU SST Support Framework. Potential future architectures are also being evaluated for the development of new future sensors, including both radar and telescope sensors and both tracking and surveillance sensors. This brings the need to analyse the performances of different sensor network architectures and topologies.

Typically, the performances of those architectures are normally measured in terms of number of *observable objects* and number of *catalogable objects*. By definition, an object is considered observable if the sensor network can observe the object at least once and generate the corresponding track. Similarly, an object is considered catalogable if it can be maintained in the catalogue through the update of its orbital information upon the generation of tracks corresponding to the object during survey observation activities. In order to do so, new tracks need to be correlated to the right object. Hence, the catalogability of an object is directly related to the ability of the system to correlate tracks properly. This depends on the revisit time (i.e., frequency of observations) for a given object population and sensor network, and furthermore, it is also driven by the on-ground infrastructure ability to maintain the catalogue to predict the orbits of the objects, depending on the accuracy of the radar measurements and Space weather indicators predictions (solar flux and magnetic field activity, among others).

Many previous studies are based on a rule-of-thumb stating that an object is catalogable if its revisit time is less than 24 hours. However, this assumption is not properly justified. Additionally, apart from the catalogability of an object, another aspect to consider is the accuracy of the orbital information being estimated from the correlated observations. Again, concepts such as catalogued and well-catalogued are normally used based on the revisit time of the objects. In the frame of these studies, two types of analyses can be performed: based on coverage analysis and based on full cataloguing processes. The first are less time consuming but of lower accuracy as they are based on analysing the observability windows of the objects of the population and based on rules-of-thumb for the revisit time (e.g. 24 hours) to determine the percentage of the population that can be catalogued. The second are much more time consuming and provide more insight but are driven by

specific implementations of correlation algorithms. Hence, it is normally preferred to use the former for preliminary design analyses.

This paper presents a new methodology suited for Low Earth Orbit and developed to determine through a coverage analysis the population of objects that can be catalogued by a given sensor network, as well as the expectable accuracy of the orbital information generated from observations of the sensor network. The number of catalogable objects is derived from the observable population, assuming that tracks can be correlated to the right object as long as the position uncertainty (i.e., covariance) of the objects do not overlap. The growth of the position covariance of each object is driven by two main uncertainties: the one of the initial estimation of the semi-major axis of the orbit and the one of the drag effect in the object. The former is related to the accuracy of the observations from the sensors (and also to the observation geometry) while the latter is related to the space environment knowledge (i.e. capability to model the atmosphere density). Depending on the object altitude, the drag effect may be dominant with respect to the initial uncertainty of the semi-major axis. Moreover, the longer the revisit time the more relevant the drag effect becomes. On the other hand, the more populated the orbits become, the more important the initial uncertainty in the semi-major axis becomes since the more objects there are, the sooner their position covariance overlap.

The model described is used to characterize the performances of a sensor network composed of a single survey radar and optimize some of its design parameters. The main free design parameters optimized in the analysis are the location of the radar (i.e., latitude) and the elevation of the radar field-of-view (FoV), while other design parameters are kept fixed: the azimuth of the radar FoV is kept pointing southwards at all times, and the size of the FoV. The power of the radar is also a parameter considered in the analysis in order to optimize the location and pointing elevation as a function of the radar power. It is important to note that the radar location and field of view constrain the orbit observability and revisit times. Depending on the location and field of view of the radar, the revisit times of the observable population vary and hence the number of catalogable objects.

Some of the **conclusions** justified in the paper and related to cataloguing performance that can be derived from the analysis are:

- The higher the number of objects, the more relevant the accuracy of the sensor is.
- For a given sensor location and accuracy, there is a **saturation limit** in the number of catalogued objects, even if more objects are observed.
- Higher elevations are better in terms of number of catalogable objects, while lower elevations are preferred in terms of revisit time and track duration.
- The rule-of-thumb of 1-day revisit time for catalogable objects is too crude as it does not take into account the number of objects of the observable population and the accuracy of the sensor.

Summary:

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Closing speech and farewell

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Closure of the Conference

Closure of the conference and farewell

On-orbit servicing and proximity #2 / 48

Commercial Collision Avoidance Service based on JSpOC SP Catalogue

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The support from JSpOC to current collision avoidance operations is priceless. As a result, most satellite owners and operators have signed with USSTRATCOM an SSA Data Sharing agreement and have Orbital Data Requests in place in order to have access to JSpOC conjunction assessment and collision avoidance support services.

JSpOC issues CDM messages in case an upcoming conjunction is detected. These messages are the main source of information to satellite operators to proceed with their collision avoidance operations. Additionally, JSpOC supports these operations if the operator provides information on the planned station keeping and collision avoidance manoeuvres by running the screening process also against the operational ephemerides.

The main issues reported by operators in this process are the following:

- The potential discontinuation of the services provided by JSpOC who might concentrate on the monitoring of the space environment (cataloguing) and hence, the need for commercial services to answer to their particular needs.
- Short notice period for some particular upcoming conjunctions leading to urgent operations to react and perform the collision risk assessment and collision avoidance manoeuvre computation.
- Potentially iterative and long process with JSpOC during collision avoidance manoeuvre computation for post-manoeuvre analysis (to anticipate conjunctions after collision avoidance manoeuvres). This leads to additional delays and uncertainties during these critical operations.

In order to answer to these issues (and particularly the first one), GMV has reached a SSA Data Sharing Agreement with USSTRATCOM to access the Special Perturbations (SP) precise catalogue containing the US unclassified objects, to exploit it and generate derived products for the provision of commercial services. The SP catalogue is routinely downloaded from space-track site (currently, on a daily basis) and post-processed by GMV for the provision of its commercial collision avoidance services through its dedicated ***focusoc*** Operations Centre. Additionally, GMV has reached dedicated agreements with a large number of commercial SST data providers (radar and optical) to integrate their sensors with ***focusoc*** to collect data in case of high-interest events in order to perform orbit determination on target and chaser objects and refine the risk assessment in case of need.

GMV's ***focusoc*** Operations Centre integrates tools for conjunction detection, collision risk assessment and collision avoidance (***closeap***) and for SST data processing and orbit determination (***sstod***) which use state-of-the-art algorithms, support parallel processing and have been used in real operations by a large number of customers, integrated in operational ground control systems. This ensures the efficiency and reliability of the solutions provided by focusoc.

To mitigate the second issue listed above, GMV has also implemented an automated process to extend the span of the downloaded SP catalogue to make it cover a longer period of time both in GEO and LEO as well as MEO and HEO/GTO. This process consists on the fitting of the available ephemerides (by means of orbit determination and parameters estimation) and their propagation with detailed dynamical models. As a result, upcoming conjunctions can be anticipated before receiving CDMs from JSpOC and the urgency of collision avoidance operations is drastically reduced by counting on additional time for decision-making processes and on the possibility to perform collision avoidance by means of adapting upcoming station keeping manoeuvres, particularly in GEO.

To mitigate the third issue listed above, GMV has automated all the processes for conjunction detection and collision risk assessment. One-vs-all assessments are performed against the operational orbit as soon as provided by the operator (including planned manoeuvres) and as soon as a new catalogue is available from JSpOC. As a result, the operator can make any post-manoeuvre analysis for all manoeuvres performed, not only those aimed at collision avoidance ensuring the safety of all their operations. No manual intervention by any man-in-the-loop is required and therefore no additional latencies are imposed to the spacecraft operations.

The ***focusoc*** system description and operational tools used, the nature of the services provided, the process to download, fit and propagate the SP catalogue, the performances achieved and the collision risk assessment and avoidance operations performed, including real cases, will be presented. The applicability to several orbital regimes (from LEO to GEO) and different mission phases (not only routine but also LEOP, re-location, EOL, etc.) will also be presented along with the feedback received by the satellite operators currently making use of the service.

Summary:

Optimization and Dynamics #1 / 2

Comparison of different optimization methods to construct an acquisition plan finding the best compromise between calculation time and algorithm performance

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The observation of celestial bodies other than Earth can without any doubt benefit from Earth observation satellites. This study, conducted for the latter, can easily be applied to Phobos for example, as part of the MMX (Martian Moons Explorer) project.

Earth observation satellites realize several acquisitions, which are linked together through an acquisition plan. The calculation time needed to construct such a plan is limited; therefore the method usually employed is the greedy algorithm which gives a solution quickly but not optimal. The temporal and kinematical constraints are numerous, making this a high combinatorial problem. The biggest time cost is the management of kinematics, especially checking if the acquisitions can be linked to each other, that is, calculating the minimum duration of the rallying sequence "attitude maneuver + acquisition". In order to reduce the calculation time, the duration of each attitude maneuver and each acquisition can be calculated approximately, by adding an error margin to the result in order to ensure the feasibility of the acquisition plan. Therefore, a balance must be found between the calculation time an optimization method leaves to the kinematics and the margin associated to the needed approximation. Typically, an exact resolution method which leaves less calculation time for kinematics computation needs a rough approximation, and thus a big margin, which deoptimizes the calculated acquisition plan.

A simplified simulation model is used in order to quickly evaluate the quality of a plan constructed with a method (for example: greedy algorithm, branch and bound, stochastic greedy algorithm, genetic algorithm, simulated annealing, taboo search), as well as the needed calculation time and therefore the time left for kinematics computation. This model is based on a time windows representation.

The originality is to determine these elements according to different margins then determine which model for the calculation of kinematics, with the associated margin, gives the best results.

The results are different depending on the studied context; the goal is to find the method that offers the best compromise between quality of the plan and calculation time left to kinematics.

Summary:

Loitering, Orbiting #1 / 10

Correlation techniques to build-up and maintain space objects catalogues

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Human activity in the space has caused the growth of a very large population of resident space objects (RSO). More than 19,000 objects are currently catalogued by 18th SPCS (former JFSCC) with sizes starting around 10 centimetres in LEO and around 1 metre in GEO. Space debris has nowadays become a very important thread to space operations as high-risk collisions are predicted daily between operational spacecraft and space debris objects.

Most space agencies have their own programs to deal with this thread, both from a mitigation point of view (IADC guidelines implementation, spacecraft design, active space debris removal), and from an operations point of view (e.g., space surveillance and tracking, collision avoidance). On the other side, the space private sector has been developing and using its own solutions to tackle the problem.

One of the key aspects to implement such measures is the availability of a catalogue of RSOs, not only characterising the properties of the objects, but also providing precise ephemerides that allow the prediction of high-risk collision events accurate enough and time in advance.

To build-up and maintain a RSO catalogue, in addition to the required sensors (radars, telescopes, SLR stations), it becomes necessary to have the required ground-segment infrastructure able to process efficiently all the data provided, in form of observation tracks, from those sensors.

Since 2007 GMV has developed and used methods to identify, track and catalogue RSOs. The SST Catalogue Maintainer Software (***catmai***) is GMV's software capable of maintaining a catalogue of man-made Earth orbiting objects and their orbital information through the processing of measurements from a pre-defined space surveillance network of sensors.

catmai is composed of an initial orbit determination tool, an orbit determination module, a track-to-orbit correlator, a track-to-track correlator, an orbit-to-orbit correlator and a catalogue post-processing component for the analysis of the cataloguing performances.

This paper will focus on the methods implemented in GMV's cataloguing solution and their performances in terms of success rate and false positive detection for the following processes:

- **Initial orbit determination**, to obtain the first estimation of the orbits from very few observations and with no a-priori information. A set of initial orbit determination methods are available for different number and type of measurements.
- **Orbit determination**, to improve the first estimation of the orbits by considering all available data, via both sequential and batch least-squares approaches. These methods are used during the correlation processes as well as for updating catalogued orbits.
- **Track-to-orbit correlation**, to correlate uncorrelated tracks (UCTs) with already catalogued objects for catalogue maintenance. Correlation is performed in the measurements domain, i.e. synthetic measurements are compared against real ones provided by the sensor network. The figure of merit considered is the distance between real and synthetic tracks (Euclidean, Mahalanobis or Bhattacharyya). The track-to-orbit correlation algorithm involves: *Synthetic tracking generation*, *pre-filtering*, *Synchronisation*, *Residuals computation* and *Correlation statistics computation.*

The performance of the track-to-orbit correlator has been evaluated on both optical and radar cataloguing maintenance scenarios. It is able to provide success rate around 99.5% (true positives), around 0.6% of false negatives while avoiding false positives.

- **Track-to-track correlation**, to associate and correlate UCTs among them in order to identify new objects not previously catalogued (i.e. catalogue build-up).
  The main concept behind this track-to-track association method is a multi-step filter that sequentially applies IOD and simple OD methods to all possible combinations of uncorrelated tracks from survey activities.
  The performance of the track-to-track correlator has been analysed on radar cataloguing build-up scenarios, leading to success rates around 99% (true positives) and the false positive rates lower than 0.1%, while keeping a high track usage rate (close to 99%).
- **Orbit-to-orbit correlation**, to correlate objects of the catalogue with those from external catalogues such as Space-Track's public catalogue. The correlation information of all objects of the involved catalogues is maintained from one analysis to another. This history of the correlation process is stored and used to ensure that two objects that used to correlate keep on correlating even if there is a manoeuvre not detected in one of the orbits through an outlier detection process.

The performance of the orbit-to-orbit correlator has been investigated by correlating a precise catalogue with the Two-Line Element Sets (TLE) catalogue for more than 15,000 objects. Results after one month of analysis reveal a success rate of more than 99.9% (true positives), around 0.08% of false negatives and only less than 0.01% of false positives.

The experience on these subjects gained by GMV with its own software solution for catalogue build-up and maintenance will also be described in the paper, using data from real sensors: more than 30 telescopes, radars and SLRs in five continents, covering all SST telescopes in Spain (OAM, TFRM, TJO, IAC, IAA), Airbus's GEOtracker telescopes, SpaceInsight telescopes, AIUB telescopes in Switzerland, ESA's OGS, Russian ISON telescopes network, and radars such as TIRA in Germany, Chilbolton in UK, ESA's MSSR, and the Spanish Navy SLR station, among others.

Summary:

Loitering, Orbiting #2 / 61

DESEO - Design Engineering Suite for Earth Observation

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Loitering, Orbiting #1 / 44

DESEO - Design Engineering Suite for Earth Observation

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The Design Engineering Suite for Earth Observation (DESEO) is a software toolkit to support mission analysis and preliminary system/subsystem design activities for all phases of Earth observation missions.

DESEO has been designed to be used by mission and system engineers throughout all phases of an Earth Observation mission (from Phase 0 to Phase E), whenever they need accurate and fast quantitative results to support design trade-offs and assessment analyses.
DESEO is a modular, flexible and self-standing application, designed so as to provide the user with a comprehensive set of mission-related and system-related computation modules and with post-processing utilities to yield meaningful numerical and graphical results.

DESEO has been designed in order to support system studies based on first-order estimation of spacecraft system/subsystems performance and mission analysis assessments, with powerful visualisation capabilities.
The tool is able to generate outputs for a specific set of inputs, and in certain cases it can also provide parametric results as a function of given variables (e.g. orbit altitude). The tool main objective is firstly to be an analysis tool (i.e. used to evaluate a given design). Nevertheless, some of its components have been developed for identifying an optimal design.

DESEO is provides the capability to perform more than forty different analyses with the capability of simulating multi-spacecraft scenarios. It can be run as stand-alone tools (command line) or operated via a Graphical User Interface (GUI). The provided analyses encompass orbit selection, orbit propagation, attitude computation, coverage analyses, timeliness analysis, ground station contact analyses, orbit maintenance, EOL analysis, OBDH analysis, delta-V budget assessment, power budget analysis and basic astrodynamics computations (geometric calculations, transformations, analytical formulas). Upgrades are currently on-going, with the objective to extend the mission design to constellations and to improve the coverage analyses with the inclusion of clouds statistics.

DESEO has been proven to be able to fully cover mission analyses tasks for EO Phase A mission studies and its components have been used as building blocks for complex simulators in a wider scope of the EO missions.

The GUI provides functionalities to manage the input insertion process (e.g. importing data from a database or from other input files), to check input boundary and consistency, to execute and monitor the analysis process (by means of log messages and progress bars) and to visualise outputs (3D interactive visualisations, Gantt charts, Cartesian plots, cartographic map representations and tables).

The Analysis Processes are the core of DESEO, in charge of performing the Mission and System Analyses. The DESEO Analysis Processes have been developed in C++, while the GUI is implemented in Java based on the Eclipse RCP.

DESEO has been developed to run on Windows, Mac OS X and Linux Operating Systems.

Summary:

Clean Space and Environment Modelling #2 / 66

DRAMA 3.0.0: A one stop shop for the verification of space debris mitigation requirements

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Clean Space and Environment Modelling #2 / 28

DRAMA 3.0.0: A one stop shop for the verification of space debris mitigation requirements

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Formalized and internationally supported space debris mitigation guidelines have been in place for several decades. Since 2010, the International Organization for Standardization (ISO) has published a comprehensive set of space system engineering standards aimed at mitigating space debris. These standards and guidelines reflect the common requirements and practices around the globe and are nowadays made applicable to most newly developed missions worldwide. In order to verify these requirements, ESA maintains and develops since 2001 the comprehensive tool DRAMA (Debris Risk Assessment and Mitigation Analysis) for the compliance analysis of a space mission with space debris mitigation standards. DRAMA 2 was released worldwide and free of charge in 2014, to support mission designers comply with ISO 24113:2011. Now DRAMA 3 is in the final stages of it development to support the same objective in view of the major revision of the standard, scheduled for publication early 2019. Major extensions include: An update of the underlying space debris environment model to include MASTER-8 in support of collision avoidance and impact assessments, revision of the standard process for casualty risk assessment with an extensions of the orbital parameters and materials covered, and the establishment of python bindings to facilitate greater flexibility for the power users. This paper will present the full functionality overview with a link to the requirements.

Summary:

Tutorial #1 / 70

Developing flight dynamics software at ESOC

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Low Thrust #2 / 24

ELECTRO: a SW tool for the ELECtric propulsion TRajectory Optimization

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ELECTRO: a SW tool for the ELECtric propulsion TRajectory Optimization

Low-thrust orbit transfers are becoming increasingly attractive thanks to the mass savings they offer and the maturity of electric propulsion technology. For this reason, there is an interest in developing fast, but still reliable trajectory optimisation methods that can be applied in the preliminary phase of the design of a mission. The tool presented is based on the averaging of the equations of motion written in equinoctial elements over true longitude. The calculus of variations is used to identify the optimal control law. In particular, the indirect optimisation method used here is based on a sequential gradient-restoration algorithm. Perturbations such as zonal gravity harmonics are included as well as shadowing effects, which need to be modelled because electric propulsion is normally switched off during eclipses. For the eclipse detection algorithm, an analytical formulation for the extreme points of the eclipse is mandatory, since the entry to and exit from the eclipse set up the limits of integration for the averaging. Further, low-thrust trajectories require a continuous variation of the thrust direction and this has to be compatible with the capabilities of the attitude control system of the satellite. This constraint can be formulated in terms of maximum angular rate for the satellite axes, maximum angular momentum and/or maximum torque. A discussion on how to cope with these constraints in the optimisation method will be presented. The capabilities of the developed tool are illustrated with examples of transfers to Geostationary Earth orbit (GEO).

Keywords: trajectory optimization, low-thrust, electric propulsion, indirect methods

Summary:

Interplanetary Flight and Non-Earth Orbits #1 / 14

Efficient design of low lunar orbits based on Kaula recursions

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Preliminary design of artificial satellite missions commonly relays on the use of simplified models that comprise the bulk of the dynamics. In the case of the gravitational potential, the amplitude of long-term oscillations of the orbital parameters is roughly one order of magnitude larger than the short-period oscillations. Because of that, dealing with just the few more relevant zonal harmonics of the potential is generally suitable for the initial steps of the procedure. In addition, the long-term evolution of the orbital parameters is customarily investigated through averaging procedures that remove the higher frequencies of the motion, in this way notably speeding the process of mission design.

However, there are cases in which the use of simplified models is not an option and full zonal potential models must be used instead. The paradigm is provided by the moon, where, due to the irregular character of the moon gravity field, mission designing of low altitude lunar orbits needs to deal with tens of, contrary to just a few, zonal harmonics. The analytical approach is still possible, but the requirement of handling huge expressions formally usually discourages mission planners, who then resort to numerical procedures. Still, useful compact recursions for dealing analytically with the problem exist in the literature since many years ago, yet limited to the equations of the averaged flow.

On the other hand, the correct computation of initial conditions requires the mathematical transformation from mean to osculating elements, and vice-versa, which may be crucial for the design of missions dealing with unstable orbital configurations, which is the common case of science orbits under third-body perturbations. Based on Kaula's popular recursions, we derive new formulas for the efficient mean to osculating transformation. While some efforts in providing these transformation equations have been made by different authors in the past, we show that the performance of the new formulas clearly surpass existing proposals in the literature. The new Kaula-type recursions, together with Kaula's original recursions for the averaged zonal potential, provide a compact and efficient way of handling analytical solutions of full potential models. The use of this kind of solution is illustrated with application to the design of low lunar orbits.

Summary:

Low Thrust #3 / 72

Efficient design of low lunar orbits based on Kaula recursions

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Preliminary design of artificial satellite missions commonly relays on the use of simplified models that comprise the bulk of the dynamics. In the case of the gravitational potential, the amplitude of long-term oscillations of the orbital parameters is roughly one order of magnitude larger than the short-period oscillations. Because of that, dealing with just the few more relevant zonal harmonics of the potential is generally suitable for the initial steps of the procedure. In addition, the long-term evolution of the orbital parameters is customarily investigated through averaging procedures that remove the higher frequencies of the motion, in this way notably speeding the process of mission design.

However, there are cases in which the use of simplified models is not an option and full zonal potential models must be used instead. The paradigm is provided by the moon, where, due to the irregular character of the moon gravity field, mission designing of low altitude lunar orbits needs to deal with tens of, contrary to just a few, zonal harmonics. The analytical approach is still possible, but the requirement of handling huge expressions formally usually discourages mission planners, who then resort to numerical procedures. Still, useful compact recursions for dealing analytically with the problem exist in the literature since many years ago, yet limited to the equations of the averaged flow.

Based on Kaula's popular work, we re-derive the long-term potential of the zonal problem in closed form and show that Kaula's approach in orbital elements provides much more efficient formulas for the construction of the mean elements potential that recent alternative proposals in the literature. The necessity of having available efficient expressions for the long-term zonal potential, from which the evolution equations of the orbit are directly derived, is illustrated with application to the design of low lunar orbits.

Summary:

Orbit Determination and Prediction Techniques #1 / 15

First tests of the C/C++ version of the Draper Semi-analytical Satellite Theory (DSST)

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The history of the creation of the Draper Semi-analytical Satellite Theory (DSST) started at the Computer Sciences Corporation, with support from the NASA Goddard Space Flight Center (GSFC), in the early 1970s. Then, its development continued at the Draper Laboratory in the 1980s and 1990s. Since 2001, some enhancements to the DSST have been achieved by the technical staff at the Massachusetts Institute of Technology (MIT) Lincoln Laboratory. The original DSST Fortran-77 code exists both as part of the Goddard Trajectory Determination System (GTDS) suite and as the DSST Standalone program. In addition, it can also be executed remotely through the Astrodynamics-Web-Tools service at the University of La Rioja. More recently, DSST has been re-implemented in Java, in the Orekit flight dynamics library. The open-source concept applied to the space domain attempts to provide free access to the software tools needed to operate safely and efficiently in space to all spacecraft and space-system operators. In this work, we address one of the tasks from the list proposed in the paper entitled: ''Open Source Software Suite for Space Situational Awareness and Space Object Catalog Work" (Cefola et al., 2010): the migration of the original Fortran-77 DSST code to C/C++. In the initial stage, the original design of DSST has been maintained as much as possible, with the aim of taking full advantage of the validation and verification process developed over DSST during the last decades. Then, the parallelization of the semi-analytical theory for multicore and GPU technologies, by using the commonly available parallel programming environments, will be analyzed. We will also consider the development of a parallel DSST orbit propagator based on the Picard-Chebyshev concept. Finally, we will take into account the complexities associated with applying the new DSST C/C++ version to orbits with arbitrary central bodies, including lunar, planetary, natural-satellite, and asteroid orbiters. Part of the validation and verification process of the DSST C/C++ version will be presented here. The resolution of the incompatibilities discovered during this process is an important advance for the following stages of this project.

Summary:

Open Source Tools and Smart Computing #1 / 3

Free CNES Flight Dynamics Tools

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For numerous years, CNES Flight Dynamics teams have made freely available some astrodynamics tools and libraries as MSLIB library. Nevertheless, these tools, essentially coded in Fortran language needed different versions of compilation depending on used platforms (Solaris, Linux, Windows …) which didn't ease its installation and therefore limit their dissemination. Some years ago, CNES astrodynamics subdirectorate made the decision to switch to Java language in particular to insure portability whatever the target machine was. As a consequence old generation astrodynamics tools were translated to Java and improved on the process. Moreover, and as a consequence of the new language, these new tools (or new versions of tools) became more easily exportable keeping them available as freely available tools and libraries. The translation and improvement effort includes both low-level libraries as PATRIUS or GENIUS and more sophisticated tools with their own Graphic User Interface (GUI) as PSIMU.

This paper will describe these different tools and libraries always linked to Flight Dynamics applications, their interaction and dependency as well as their dissemination mode (open source, freeware).

Initially we will describe low-level libraries as PATRIUS uniquely devoted to Flight dynamics aspects and GENIUS for scientific GUI development. Secondly we will also present GENOPUS library which is based on both previous ones and allows providing "intelligent" widgets as the one used for defining orbit parameters.

Then, we will present some tools based on these building blocks as PSIMU (for any kind of trajectory extrapolation around Earth) or MIPELEC (optimization of low thrust propulsion). We will also give as example, tools used in operational contexts as ELECTRA.

To finish, means to get and use these tools will be described via the CNES Web site, their licenses, Wikis (including tutorials and Javadoc) or even training course.

Summary:

This paper will describe different Java Flight Dynamics tools proposed by CNES in a free delivery mode (open source, freeware). These tools may be low-level libraries (PATRIUS, GENIUS) as well as more sophisticated ones (PSIMU) which are fully portable and therefore easily usable both for studies and operational activities.

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Human Enabled Robotic Architecture Capability for Lunar Exploration and Science

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Key note speech by ESA about the exploration of the Moon

Interplanetary Flight and Non-Earth Orbits #1 / 1

Interplanetary trajectory design and mission analysis using gravity assists

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In this paper, a mathematical model was developed to design and optimize interplanetary trajectories that include gravity assist. The method of patched conics and a solver of the Lambert problem transfers are used to cast the space trajectory design process as an optimization problem, subsequently solved by using MATLAB. This model has been tested to provide an overview of the processes involved in the interplanetary trajectory design and analysis of the Juno mission to Jupiter. The resulting mission analysis compared to the actual data published.

Summary:

Open Source Tools and Smart Computing #1 / 47

JSatOrb: ISAE-Supaero's open-source software tool for teaching classical orbital calculations

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JSatOrb is an ISAE-Supaero's software tool dedicated to orbital calculation and designed for pedagogical purposes, with professional level features outputs.
It has been initiated to find a soft which would fill the gap between local teachers developed tools and professional tools, exploiting state of the arts
algorithms concerning space mechanics calculus. Even if current provided open source libraries are not fully compliant with our pedagogical requirements (simplicity, flexibility, multi-plateform and ergonomics), they provide complete and accurate calculus methods that are dedicated to a professional use. However, GUI part is not the main concern when used by space engineers which only require API access.

Concretely, JSatorb project is open-source (MIT license) and under development. It is inspired from current full-stack implementation methods. Ergonomic and intuitivity are at stack concerning the front-end, which is mainly based on Angular (https://angular.io/) and Cesium (cesiumjs.org). Efficiency and correctness on calculus are provided by the back-end part, which relies on Orekit (https://www.orekit.org/). Developed in Java, Orekit is a space dynamics open source library. It depends only on the Java Standard Edition version 8 and Hipparchus (https://hipparchus.org/) version 1.0 libraries at runtime. It is widely used by ESA and CNES.

Summary:

Open Source Tools and Smart Computing #2 / 65

JSatOrb: ISAE-Supaero's open-source software tool for teaching classical orbital calculations

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Low Thrust #2 / 5

LOTNAV: A low-thrust Interplanetary Navigation Tool

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LOTNAV has been for 10 years the ESA reference tool in the design of finite-thrust and ballistic interplanetary spacecraft trajectories and the preliminary assessment of navigation and guidance issues on the computed trajectories. Within DEIMOS Space LOTNAV has recently undergone a considerable update effort to enhance its modularity and flexibility to extend its applicability to new navigation problems. The aim of this paper is then to present LOTNAV capabilities and its most important updated features (i.e. the inclusion of ephemeris data directly from SPICE, new launcher models, the GNSS signal in the navigation analysis, etc.).

The main high-level analyses covered by LOTNAV are:

- Production of a continuous trajectory profile compliant with a set of environmental conditions and thrust hypotheses and also consistent with an ulterior use for navigation and guidance purposes.

- Generation of a number of system observables which shall allow performing an ulterior estimation of the spacecraft state as well as a number of environmental parameters.
- Theoretical assessment of the achievable levels of accuracy in the spacecraft state dispersion and knowledge, as well as on the selected environmental parameters.
- Estimation of the spacecraft state vector and the selected environmental parameters together with the required guidance to meet the system goals over a number of cases. Statistical analysis of the results obtained allows validation of the theoretical results.

LOTNAV provides answers to the above aspects in an efficient, modular and integrated fashion. In line with each of the previous functions, a software module was developed:

- Trajectory Reconstruction
- Measurement Generation
- Covariance Analysis
- Simulation
- Support Tools
- Project Management

The Trajectory Reconstruction Module provides with a consistent trajectory definition meeting the finite-thrust mission goals proposed by the user, which will serve for the purposes of the rest of the modules. Then, to analyse the trajectory estimation process, the observables for the system are established and computed within the Measurements Generation Module. Those observables together with the trajectory definition allow carrying out a theoretical assessment by means of Covariance Analysis. Trajectory determination and guidance are dealt together by the Simulation Module, performing a Monte Carlo simulation over the full navigation process thus obtaining empirical statistics of the system knowledge and dispersion in presence of low-thrust guidance.

Summary:

LOTNAV has been for 10 years the ESA reference tool in the design of finite-thrust and ballistic interplanetary spacecraft trajectories and the preliminary assessment of navigation and guidance issues on the computed trajectories. Within DEIMOS Space LOTNAV has recently undergone a considerable update effort to enhance its modularity and flexibility to extend its applicability to new navigation problems. The aim of this paper is then to present LOTNAV capabilities and its most important updated features (i.e. the inclusion of ephemeris data directly from SPICE, new launcher models, the GNSS signal in the navigation analysis, etc.).

Low Thrust #3 / 69

MARS SAMPLE RETURN: MISSION ANALYSIS FOR AN ESA EARTH RETURN ORBITER

Authors: Eric Joffre[1] ; Uwe Derz[2] ; Marie-Claire Perkinson[2] ; Jakob Huesing[3] ; Friederike Beyer[3] ; Jose-Manuel Sanchez Perez[4]

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Bi-lateral discussions between NASA and the European
Space Agency identified the orbiter element as a promising
European-led contribution to a future international Mars Sample
Return campaign. Airbus recently completed the Mars

Sample Return Architecture Assessment Study on behalf of
ESA, with the objective to identify and quantify candidate
mission architectures. The paper describes the mission analysis
that has been conducted to support preliminary system
design, launch mass estimation and mission timeline for the
architectures investigated. It includes the optimisation of interplanetary
transfers, Mars operations including aerobraking
and rendezvous, up to Earth re-entry conditions.

Summary:

Optimization and Dynamics #2 / 49

MODHOC - Multi Objective Direct Hybrid Optimal Control

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MODHOC (Multi Objective Direct Hybrid Optimal Control) is a toolbox for the design, optimisation and trade off study of space systems and missions.
It solves general nonlinear multi phase optimal control problems, automatically computing a well spread set of optimal trade off solutions. In addition, it is able to handle discrete optimisation parameters.
In order to do so, MODHOC combines a direct transcription method based on finite elements, a global multi objective optimisation algorithm combining evolutionary heuristics and mathematical programming.
MODHOC has been applied to a variety of applications: from the optimisation of launch vehicles and their ascent, abort and re entry trajectories, to the design of the optimal deployment of constellations of satellites, to the design of multi target missions.
In this paper, the main elements of MODHOC are described and the application of the software in space and non space related sample problems is demonstrated.

Summary:

Low Thrust #3 / 9

MOLTO-IT: A Multi-Objective Low-Thrust Optimization Tool for Interplanetary Trajectories

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Low-thrust propulsion and gravity assists maneuvers are both well known to provide sig- nificant benefits in terms of required propellant mass for interplanetary trajectories. However, the propellant reduction achieved with low-thrust engines, when compared to their chemical counterparts,

comes at the cost of a higher transfer time.
Therefore, the design and optimization of interplanetary missions has to be treated as a multi-objective optimization problem. The goal is to determine the set of optimal trajec- tories along with the optimal sequence of planetary flybys. For such purpose we present the optimization tool MOLTO-IT (Multi-Objective Low-Thrust Optimizer for Interplanetary Tra- jectories). It is based on a two-step sequential algorithm. In the first step, the trajectory is assumed to be a Generalized Logarithmic Spiral. A heuristic global search algorithm combined with nonlinear programming are in charge of optimizing the set of parameters defining the spirals, as well as the number, sequence and configuration of the gravity assists. As a result the set of Quasi-Pareto-Optimal solutions trading off propellant mass consumed and time of flight are obatined. In the second step, candidate solutions are regarded as initial guesses for a direct collocation method, where the problem is transcribed into a Nonlinear Programming Problem by discretization, considering the full dynamics and the complete set of constraints.
A full overview of the capabilities and features of MOLTO-IT will be given. In particular, the ability to obtain optimal flyby sequences without a-priory knowledge by the user. Addi- tionally, the effectiveness of our methodology to generate not only rapid performance estimates for preliminary trade studies, but also accurate calculations for the detailed design, will be highlighted.

Summary:

Low Thrust #3 / 8

MOLTO-OR: A Multi-Objective Low-Thrust Optimization Tool for Orbit Raising

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Commercial and scientific satellites located in Geostationary Equatorial Orbit (GEO) that are not placed there by the launch vehicle are often injected in a parking orbit. They are transferred therefrom to GEO using their own on-board propulsion system. The classical strategy relies either on Chemical Propulsion (CP) or Electrical Propulsion (EP). The former guarantees very short transfer times, whereas the latter attains propellant savings at the cost of longer on-orbit delivery times. Intermediate performances may be obtained by allowing both propulsion subsystems to coexist on a Combined-Chemical-Electric (CCE) platform.
Therefore, the design and optimization of orbit raising missions has to be treated as a multi-objective optimization problem. The goal is to determine the set of optimal trajectories along with the optimal propulsion subsystem. For such purpose we present the optimization tool MOLTO-OR (Multi-Objective Low-Thrust Optimizer for Orbit Raising). It incorporates models for EP, CP and CCE platforms and realistic effects on the space environment, such as eclipse effects, Earth oblateness perturbations and solar-cell degradation due to passage though the Van-Allen radiation belts. Additionally, complex operational constraints such as slew-rate limitations, avoidance of the Geostationary ring or phasing to a certain orbital slot can be imposed. MOLTO-OR is based on a two-step sequential algorithm. In the first step, the low-thrust control law is derived from a Lyapunov function and the chemical maneuvers are regarded as instantaneous impulses. A heuristic algorithm computes the set of Quasi- Pareto-Optimal solutions trading off propellant mass consumed, time of flight and radiation damage. In the second step, candidate solutions are deemed as initial guesses to solve the Nonlinear Programming Problem resulting from direct transcription of the problem.
A full overview of the capabilities and features of MOLTO-OR will be given. The effec- tiveness of our methodology to generate not only rapid performance estimates for preliminary trade studies, but also accurate calculations for the detailed design, will be highlighted.

Summary:

Re-Entry and Aero-Assisted Manoeuvres #1 / 46

Mission Design for a Retropropulsive Mars Pinpoint Landing

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The execution of precision landing missions applicable for future Mars missions (in particular, Sample Return Missions and Human Missions) is a major technical challenge that will require the adoption of a set of technologies that have not yet been demonstrated in flight.
In the scope of an ongoing ESA activity related to the development of advanced navigation techniques for pinpoint landing on the Moon and at Mars, an effort was directed towards the trajectory definition component of such missions. For the Mars case in particular, it is found that the enhanced control authority permitted by using a retropropulsive phase directly after the reentry phase (e.g. without the use of parachutes) is very likely to improve the ability to land precisely at a pre-defined site. This requires a sequence of design steps not normally associated with the concept of optimization (such as including specific guidance algorithms in the trajectory design loop, both during the entry and the retropropulsive phases), while simultaneously ensuring that there are specific time windows for the ground observations necessary to acquire sufficient knowledge to enable true pinpoint landing (defined as landing well within 100m of a selected site). The present work describes the mission design process which was followed in the scope of the ESA ANPLE activity for the purpose of demonstrating the feasibility of Mars pinpoint landing missions, from interplanetary transfer to touchdown, and considering the specific application to a NASA MER/NASA Phoenix/ESA Exomars EDM-class vehicle and assuming the availability of highly accurate navigation aids (including vision-aided absolute navigation means and the use of orbiting and surface radio beacons).

Summary:

Ascent #1 / 6

Multidisciplinary Modeling and Simulation Framework for Reusable Launch Vehicle System Dynamics and Control

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Future launch vehicle concepts and technologies for expendable and reusable launch vehicles are currently investigated by the DLR research project AKIRA, focusing on vertical takeoff and horizontal landing (VTHL), as well as horizontal takeoff and horizontal landing (HTHL) concepts.

Dedicated developments of multidisciplinary frameworks for launch vehicle modeling and preliminary design optimization have been presented in the relevant literature. Moreover, it is common that these activities are performed by several independent, discipline-specific tools. With such an approach, only a limited amount of interactions of the involved disciplines with the overall system dynamics can be accounted for.

This paper focuses on the multidisciplinary launch vehicle guidance, control, and dynamics modeling framework that has been developed at DLR-SR in support of the aforementioned reusable launch vehicle design activities taking into account highly interconnected disciplines (propulsion, aerodynamics, and GNC, amongst others) and changing environmental conditions. The modeling framework is based on the object-oriented, multi-disciplinary, and equation-based modeling language Modelica.

In this paper, dedicated 3-DoF and 6-DoF models, covering the kinematics and dynamics formulation, environmental effects, aerodynamics, and propulsion models for system dynamics analyses, trajectory simulations and GNC design are presented.

In particular, we showcase the advantages of using nonlinear inverse models obtained automatically by Modelica. This method establishes a direct connection between 3-DoF and intermediate 6-DoF models considering trajectory optimization results provided by the DLR Trajectory Optimization Package 'trajOpt'. With this approach, angular rates and resulting moments can be obtained by the intermediate 6-DoF model for subsequent controllability analyses. The benefits of our modeling framework are discussed in terms of future GNC design and trade-off studies.

Summary:

Interplanetary Flight and Non-Earth Orbits #3 / 43

Near Rectilinear Orbits around the Moon as Operational Orbit for the Future Deep Space Gateway

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A renewed vision to send humans beyond Low Earth Orbit (LEO) has given rise to a whole range of studies proposing different destinations and operational orbits for a new crew-tended space station; referred here as Deep Space Gateway (DSG). Near Rectilinear Halo Orbits (NRHO) have been identified as one of the most promising destinations for a DSG, due to the combination of both dynamical properties and accessibility to potentially water-rich regions in the Moon.

The aim of this paper is then to revise the suitability of NRHOs as long-term destinations for this new space station. The NRHO family indeed appears as a continuation of the classical Halo orbits, and, as such, also allows continuous line of sight with the Earth. The paper revises the formal definition and identification of NRHOs, as in the CR3BP model. Dynamical substitutes of the NRHOs are also refined in the Bi-Circular Model (BCM) by means of a multiple shooting method. key features such as lunar south-pole coverage, station keeping requirements and accessibility of the orbit are then analysed.

In particular, nine different station keeping strategies were identified and implemented. These strategies correspond to variations of three different underlying schemes; the cancellation of unstable dynamical modes, the use of the multiple shooting method and, finally, a weighted single-criteria numerical optimization.

The accessibility to and from the NRHO is discussed by computing direct and invariant manifold lead transfers. Due to the dynamical characteristics of the L2 NRHO family, the optimal strategy to transfer from LEO orbit is to target an auxiliary halo orbit first, for a posterior sub-transfer to a member of the NRHO family. A minimum LEO-to-NHRO transfer cost of 3.68 km/s is achieved following the aforementioned transfer strategy. An annual station keeping budget of only 1 m/s would be expected considering 1 km and 1 cm/s 3σ navigation errors, as computed in a BCM framework.

Summary:

Open Source Tools and Smart Computing #1 / 18

OPEN SOURCE ORBIT DETERMINATION WITH SEMI-ANALYTICAL THEORY

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Space objects catalog maintenance demands an accurate and fast orbit determination (OD) process to cope with the ever increasing number of observed space objects. The development of new methods, that answer the two previous problems, becomes essential.

Presented as an alternative to numerical and analytical methods, the Draper Semi-analytical Satellite Theory (DSST) is an orbit propagator based on a semi-analytical theory allowing to preserve the accuracy of a numerical method while providing the speed of an analytical method. This propagator allows computing the mean elements and the short-period effects separately. We reproduced this architecture at the OD process level in order to be able to return, as desired, the mean elements or the osculating elements. Two major cases of use are thus possible: fast OD for big space objects catalog maintenance and mean elements OD for station keeping issues.

This paper presents the different steps of development of the DSST OD included in the Orekit open-source library. Integrating an orbit propagator into an OD process can be a difficult procedure. Computing and validating derivatives is a critical step, especially with the DSST whose equations are very complex. To cope with this constraint, we used the automatic differentiation (AD) technique. AD has been developed as a mathematical tool to avoid the calculations of the derivatives of long equations. This is equivalent to calculating the derivatives by applying chain rule without expressing the analytical formulas. Thus, AD allows a simpler computation of the derivatives and a simpler validation. AD is also used in Orekit for the propagation of the uncertainties using the Taylor algebra.

Existing OD applications based on semi-analytical theories calculate only the derivatives of the mean elements. However, for higher accuracy or if the force models require further development, adding short-period derivatives improves the results. Therefore, our study implemented the full contribution of the short-period derivatives, for all the force models, in the OD process. Nevertheless, it is still possible to choose between using the mean elements or the osculating elements derivatives for the OD.

This paper will present how the Jacobians of the mean rates are calculated by AD into the DSST-specific force models. It will also present the computation of the state transition matrices during propagation. Both mean elements and short-periodic derivatives are developed. The performance of the DSST OD is demonstrated under Lageos2 and GPS orbit determination conditions.

Keywords: Orbit determination, Automatic differentiation, Semi-analytical theory, DSST, Open-source, Orekit.

Summary:

Low Thrust #2 / 33

OPTELEC - Constrained Low-Thrust Transfer Optimisation Tool applied in Operational Context

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Traditionally, GTO to GEO transfers using chemical propulsion consist in optimising a rather limited number of manoeuvres. The compliance with the satellite platform and operational constraints are then guaranteed by the launch window design. On the contrary, low-thrust transfers of GEO satellites require very long thrust phases. The complex satellite platform and operational constraints

induced by the use of EP and by long transfer durations need to be handled by a dedicated transfer optimisation software. This paper presents the development, validation and various uses of the in-house low-thrust transfer optimisation software, OPTELEC.

The development of OPTELEC was performed to serve a three-fold purpose:
(i) Studies and mission analysis: to be used for system definition, that is transfer ΔV budget and duration estimation, computation of satellite attitude guidance throughout the transfer and optimisation of a selection of injection orbit parameters. Minimum-time and minimum propellant mass low-thrust transfers are covered. Up to three levels of thrust are available (full thrust, reduced thrust and coast arc). The family of injection orbit is not limited to GTO but can typically span from LEO to SSTO.
(ii) Operations: to be used during the Launch and Early Operation Phase (LEOP). This includes the development of high fidelity environment and satellite modelling, along with the satellite platform and operational constraints to be taken into account throughout the optimisation process. Also, real time operations require a fast-running software, without the need of an optimisation technics expertise.
(iii) Versatility: to be developed to handle not only Electric Orbit Raising (EOR) transfers optimisations but also transfers with high thrust propulsion only - using the Liquid Apogee Engine (LAE) - and hybrid transfers including any combination of LAE burns, Reaction Control Thruster (RCT) burns and thrust with EP.

This paper shows how OPTELEC was designed in order to achieve these goals and the applications to a wide variety of missions.

Summary:

Interplanetary Flight and Non-Earth Orbits #2 / 29

On the Design of Transfers to Solar-Sail Displaced Orbits in the Earth-Moon System

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The existence of families of solar-sail displaced libration point orbits in the Earth-Moon system has recently been demonstrated. These families originate from complementing the dynamics of the classical Earth-Moon circular restricted three-body problem with a solar-sail induced acceleration. The addition of this acceleration makes the problem non-autonomous, but by constraining the orbital period in a differential correction scheme, closed orbits can be found that are periodic with the Sun's synodic motion about the Earth-Moon system. These orbits can be catalogued into traditional orbit families such as solar-sail displaced Lyapunov, halo, and vertical Lyapunov orbits where different families can be generated for different solar-sail steering laws. Previous work has furthermore demonstrated the applicability of these orbits for high-latitude observation of the Earth and Moon. To not only demonstrate the existence and applicability of these orbits, but also their accessibility, this paper investigates the design of solar-sail transfers to a subset of solar-sail displaced libration point orbits in the Earth-Moon system.

Initial guesses for the transfers are generated using reverse time propagation of the dynamics starting from a grid of state-vectors along the targeted periodic orbits. The backwards propagated transfers are truncated at close approach to Earth. Furthermore, the control is provided through a locally optimal steering law that maximises the solar-sail acceleration component along the inertial velocity vector. These near-feasible initial guesses are subsequently transferred into a highly constrained 12^{th}-order Gauss-Lobatto collocation scheme to improve their feasibility. Constraints are included that ensure linkage between the start of the transfer and commonly used Earth parking orbits, a

minimum altitude with respect to the Earth and the Moon, as well as a realistic maximum rotation rate of the solar sail of 20 deg per day.

The paper provides sets of feasible trajectories for realistic- near-term solar-sail technology. In particular, transfers to a solar-sail displaced Lyapunov orbit at L_1 and a halo orbit at L_2 are provided as well as a two-spacecraft transfer to a constellation of solar-sail displaced vertical Lyapunov orbits at L_2. This constellation achieves continuous coverage of both the lunar South Pole and the center of the Aitken Basin, while maintaining an uninterrupted communication link with Earth. The Aitken Basin is of great scientific interest as it is believed to hold clues to the history of the Moon and allows access to the deeper layers of the lunar crust. The lunar South Pole is often mentioned as a potential location for a human outpost because it is an area of near-permanent sunlight, providing access to power, and water ice is most likely present in the continuously shaded areas of the crater interior.

For the two-spacecraft transfer to the constellation of vertical Lyapunov orbits at L_2, identical launch conditions for both spacecraft are sought for, such that the constellation can be initiated using a single launch by a Soyuz launch vehicle. The resulting transfers allow two 1160-kg spacecraft to be launched into standard highly elliptical Earth parking orbits from where the solar sail is deployed to transfer the spacecraft to their respective orbits at L_2. These transfers take 53.1 and 67.9 days to complete before the spacecraft enter their respective constellation orbits. These results prove the accessibility of solar-sail displaced libration point orbits in the Earth-Moon system, thereby reaffirming the potential of solar-sail technology to enable novel scientific missions in the Earth-Moon system.

Summary:

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Opening the 7th ICATT

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Opening speech of the 7th ICATT conference by Dr. J. Bals from DLR

Orbit Determination and Prediction Techniques #1 / 21

Operational Orbit Determination for the Eumetsat GEO fleet based on optical observations.

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The Eumetsat GEO fleet has successfully performed its orbital determination using ranging-only data from three different ground-based tracking stations. Data from two different ground stations is used for each of the satellites. For more than 15 years, these results have been used for manoeuvre planning, after manoeuvre calibration and collision risk assessment. Since June 2018, an effort has been undertaken to enhance the orbit determination procedures with optical observations, with the

objective of assessing the suitability of using optical data to perform the operational orbit determination, and determining the suitability of routine orbit determination and manoeuvre calibration using mixed data from ranging stations and telescopes. The optical measurements are provided currently by the Deimos Sky Survey (DeSS) telescopes, using additional sensors (coordinated by Deimos Space) as a backup in case of adverse weather conditions or technical issues. The processing of the measurement data is being performed by two separate teams at Eumetsat and at Deimos. This paper describes the processing chain put in place at Deimos for performing the observations and the processing of measurement data, and summarises the findings related to the aforementioned objectives after several months of routine observations.

Each spacecraft in the fleet is observed with optical sensors at least twice every week (in two observation slots), with each slot spanning at least 15 minutes, and with a minimum separation of two hours between slots. A software processing chain based around several Deimos tools has been put in place to plan the observations (SHUX), perform the observations (ITOX), automatically process and resolve the images (TRAX), split tracks (TRACA), and finally perform the orbit determination (TRADE).

The orbit determination is automatically performed weekly with the TRADE by means of a Batch Least Squares approach with a two-week rolling window. In absence of manoeuvres, this allows determining the solar radiation pressure coefficient while maintaining consistency with the previously computed orbits. When a manoeuvre is scheduled, optical observations are taken as soon as possible after the manoeuvre itself. In this case, the paper shows that the orbit determination with optical information only provides results comparable with the nominal range-only orbits.

Finally, the TRADE tool is modified to process range-only measurements from ground-based ranging stations along with optical measurements. A comparison of the quality of the solutions obtained with different combinations of measurements from different sensors is provided.

Summary:

Interplanetary Flight and Non-Earth Orbits #2 / 23

Optimized transfers between Earth-Moon invariant manifolds

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The tentative position of the next habitable space station could be a southern L2 Near Rectilinear Halo Orbit (NRHO) of the Earth Moon System. To bring crew and cargo to the station, a safe and efficient rendezvous methodology has to be established. However, a significant body of work remains to be done on the design of the rendezvous procedure between halo orbits. Given fixed start and end halo orbits, direct transfers (such as Lambert arcs) between the two can produce simple strategies with short transfer time at the cost of relatively high velocity increment Δv, measured in m/s [1].

When longer transfer time is allowed, as with some cargo, lower energy transfers, taking advantage the natural dynamics of the Earth-Moon system, can be used. The most commonly used topological objects are the stable and unstable manifold of a given orbit. These trajectories can lead a spacecraft very far from the original orbit at very low maneuver cost. One strategy is to insert into the unstable manifold of the initial orbit, making the spacecraft leave the starting orbit. After an optimized flight time T_F, a maneuver Δv is performed to insert into the stable manifold of the final orbit, and thus converging to the final orbit [2]. This procedure, however, requires the existence and explicit construction of physical intersections of the stable and unstable manifolds of the original orbits.

In this article, a method is presented to generate the manifolds along with an approximation of the set of their intersections by triangulating the surface and applying a modified Moeller's method

[3]. The most promising points are then further refined into true intersections and the lowest Δv is chosen. The article considers transfer between regular halo to halo, NRHOs to NRHOs and halo to NRHOs. Analysis shows that for several configurations the maneuvers cost can be sizably reduced as compared to fully optimized Lambert arcs.

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Summary:

Loitering, Orbiting #1 / 0

POINCARE: A MULTI-BODY, MULTI-SYSTEM TRAJECTORY DESIGN TOOL

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Poincare is a modular trajectory design tool based on a catalog of three-body science orbits and a differential corrector to compute connecting transfer arcs between orbits in multi-body systems. Poincare attempts to offer a unified approach, i.e., an "all-in-one" integrated search within one interface and setup in MONTE (JPL's signature astrodynamic computing platform) The Science Orbit Design Tool – first module –facilitates rapid and well-informed decisions regarding the selection of periodic orbits for a particular mission and enables the simultaneous study of various orbit alternatives. The second module – the Reference Trajectory Design Tool – allows the user to calculate optimal transfer paths from a departure orbit to a science orbit via dynamical systems structures, resulting in an end-to-end reference trajectory.

In the 1960's, the application of insight from the circular restricted three-body problem (CR3BP) moved into the 'space age' when a mission to the Lagrange points was considered for NASA's Apollo program. Since then, many of the structures that emerge in the CR3BP have been more actively exploited in trajectory design. Consequently, successful missions to the vicinity of the Lagrange points have since been launched, including the International Sun-Earth Explorer-3 (ISEE-3), the Solar Heliospheric Observatory (SOHO), the Advanced Composition Explorer (ACE), and the Microwave Anisotropy Probe (MAP). Parallel to the development of these mission concepts, the possibility of applying dynamical systems techniques to the design of these types of trajectories was also being considered. In fact, in the 1960's, Conley had investigated low energy transfer orbits to the Moon using dynamical system techniques. In the 1990's, the use of invariant manifolds in the design process to construct pathways between the Earth and the Sun-Earth libration points was finally applied in an actual trajectory: the trajectory supporting the Genesis mission.

Summary:

Poincare is a modular trajectory design tool based on a cat- alog of three-body science orbits and a differential corrector to compute connecting transfer arcs between orbits in multi- body systems. Poincare attempts to offer a unified approach, i.e., an "all-in-one" integrated search within one interface and setup in MONTE (JPL's signature astrodynamic computing platform [1].) The Science Orbit Design Tool – first module – facilitates rapid and well-informed decisions regarding the se- lection of periodic orbits for a particular mission and enables the simultaneous study of various orbit alternatives. The second module – the Reference Trajectory Design Tool – allows the user to calculate optimal transfer paths from a departure orbit to a science orbit via dynamical systems structures, re- sulting in an end-to-end reference trajectory.

Tutorial #2 / 71

PTScientists: Mission to the Moon and the return to Apollo 17

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PTScientists is bringing down the cost of exploration and building reusable space infrastructures so more people benefit from access to space - because space belongs to everyone.

ALINA is designed to be compatible with all major commercial launch vehicles (including SpaceX's Falcon 9, and ISRO's PSLV), allowing invaluable flexibility when contracting a launch. This means that customers can choose their launch vehicle based on price and availability, rather than being restricted by type.

Once inserted into an Earth-orbit, ALINA will use its own propulsion to transport the craft 380,000 km to the Moon, and into lunar orbit. Payload racks on either side of the ALINA vehicle can be used to house orbital payload (if a lunar landing is not required), and ALINA can also dispatch CubeSats into lunar orbit.
From lunar orbit, ALINA will perform an accurate soft-landing on the Moon at a designated landing site. With each flight, ALINA can deliver up to 100 kg to the surface of the Moon. For Mission to the Moon this includes carrying two Audi lunar quattro rovers.

On-orbit servicing and proximity #1 / 7

RICADOS - Rendezvous, Inspection, CApturing and Detumbling by Orbital Servicing

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Safe de-orbiting and increasing the life-time of satellites by on-orbit servicing (OOS) will be of high importance in future spaceflight. The rendezvous and docking/berthing (RvD/B) phase is one of the most complex and critical parts of on-orbit servicing and debris removal missions. Several missions and developments have been started like the Restore-L mission of NASA, the RSGS (Robotic Servicing of Geosynchronous Satellites) program of DARPA, the Mission Extension Vehicle (MEV) of Orbital ATK and the ESA Clean Space Initiative (e.Deorbit). Robotic servicing will be of importance also in human spaceflight since rendezvous and docking technology generally plays a major role in all assembly, service and maintenance tasks.
All these missions require robust and reliable guidance, navigation and control (GNC) systems for rendezvous and robotic systems for berthing and maintenance tasks. In a recently started project called RICADOS (= Rendezvous, Inspection, CApturing and Detumbling by Orbital Servicing) the German Aerospace Center (DLR) develops a new on-board inspection, rendezvous and robotic system as well as a ground segment for on-orbit servicing including telepresence capability.
The paper presents the current status of the project and the end-to-end testing environment: The space segment is simulated using two robotic hardware-in-the-loop test facilities at the German Aerospace Center: the European Proximity Operations Simulator (EPOS 2.0) at DLR-German Space Operations Center, where the inspection and rendezvous is tested and demonstrated, and the OOS-Simulator (OOS-Sim) at DLR-Robotics and Mechatronics Center, where the capturing and detumbling are performed. The robots' motion is generated by a numerical satellite simulator in software based on orbit and attitude dynamics for service and target satellite, simulation of actuators and of

the satellites' environment. The communication path from space to ground and vice-versa is simulated such that different scenarios can be tested: Different channel parameters such as telemetry and tele-command data loss, jitter and delay can be chosen for realistic tests. The ground segment is established as for a real on-orbit servicing mission with dedicated consoles (standard satellite console, rendezvous console and robotic console). In a multi-mission control room, which is used for real missions at the same time, the operators can train and collect experience how to run a real on-orbit servicing mission including the robotic capture via telepresence.
The paper also presents a first reference scenario based on DLR's satellite strategy: DLR will launch several compact satellites like Eu:CROPIS in Low Earth Orbit in the next years. The reference scenario of RICADOS foresees a service satellite with rendezvous and berthing payload which is able to perform service tasks for a fleet of compact satellites in neighboring orbits. This paper describes the entire RICADOS-concept for the selected reference scenario, from inspection and rendezvous towards final capturing and detumbling of the target satellite, the ground contact concept for the mission, and presents results of the latest simulations and tests.

Summary:

On-orbit servicing and proximity #1 / 25

ROSSONERO: a tool for preliminary rendezvous mission design in the restricted three-body problem

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Extended abstract

The present international cooperation scenario for robotic and human space exploration is focusing on mission architectures that revolve around building and exploiting a crew-tended cis-lunar space station, known as Lunar Orbital Platform-Gateway. Candidate orbits for this vehicle are the near rectilinear halo orbits (NRHO). Therefore, the capability to inject in NRHO and perform rendezvous and docking or berthing maneuvers with a station in NRHO is key to many future exploration missions.

The aim of ROSSONERO (Rendezvous Operations Simulation Software on Near Rectilinear Orbit) is to provide a tool for preliminary design and assessment of rendezvous trajectory in NRHO and, more in general, in restricted three-body problem scenarios. The tool is developed in MATLAB\Simulink and dynamics simulation is based on the equations of relative motion proposed in 1. Unlike other sets of equations proposed in the past for relative motion in the restricted three-body problem, the equations describe the dynamics of chaser spacecraft with respect to a target in the local-vertical local-horizon (LVLH), a local frame centered on the target generally adopted for rendezvous analysis [2]. For completeness, in last version of the tool, the rotational dynamic has been added based on the equation proposed in [4].

ROSSONERO rendezvous mission description is based on the definition of a set of waypoints in the LVLH, that the chaser must reach during its approach to the target. Two types of maneuver can be used for transferring from a waypoint to the next one: impulsive or continuous thrust. Maneuvers computation is performed by integration of linear equations of relative motion derived in 1 and in [3]. In these References, two different sets of linear equations are presented, obtained by linearizing the exact relative dynamics and using two different assumptions for the primary bodies motion: elliptic and circular restricted three-body problem. ROSSONERO allows the user to choose between these two sets for maneuver computation at each transfer arc. Overall mission analysis is then performed by means of key performance indexes such as maneuver execution error and propellant consumption. Plots showing the trajectory in the LVLH frame and control evolution are provided as well. An example of the output provided by the tool is shown in Figure 1 and in Figure 2 (see attached document).

Acknowledgments

This work was partially supported by the European Space Agency under contract No. 000121575/17/NL/hh. The view expressed herein can in no way be taken to reflect the official opinion of the European Space Agency.

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Summary:

Clean Space and Environment Modelling #2 / 67

Rate and collision probability of tethers and sails against debris or spacecraft

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Clean Space and Environment Modelling #2 / 45

Rate and collision probability of tethers and sails against debris or spacecraft

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Sails and electrodynamic tethers have been proposed as passive devices to deorbit dead satellites. Their implementation in satellites would diminish very much their deorbiting time, typically down to a few years, as opposed to several decades. However, they would also increase very much the collision cross section of the said satellites, which would therefore increase the probability of collision per unit time. This situation calls for a computation of probabilities and rates related to collision between sails and tethers with debris or satellites, in order to check that endowing satellites with sails or tethers is indeed advantageous.

The work done on probability of collision between spherical objects in orbit by various authors is extended here to the case of one spherical object and one flat object of circular or rectangular

shape. The former is a model for spacecraft or debris, while the latter is a model for a sail or a tether. The work presented here is almost always analytical, that is, formulae are given. Two kinds of calculations are presented.

The first is the computation of the collision rate when the flux of one object (typically debris) with respect to the other object is known. Formulae are given both for a sail with random attitude and for a sail with fixed attitude. In the case of the tether formulae are given both for a tape tether and a tether of round cross-section. This information is important when planning a mission.

The second kind is the computation of the collision probability for a particular pair of objects whose probability density functions of the positions are known. This information is necessary to decide if an evasive maneuver is going to be performed or not.

All the work presented here is analytical except for the case of sail-sphere collision probability, which requires the integration of a bivariate Gaussian over a parallelogram. In order to compute it we have developed a program in which the fastest available algorithm to integrate a bivariate Gaussian is embedded. We have made this program available to the public as an app.

Summary:

Clean Space and Environment Modelling #1 / 20

ReDSHIFT software tool for the design and computation of mission end-of-life disposal

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One of the deliverables of the ReDSHIFT H2020 project will be a software tool available to the scientific community and the public via a web-based interface. The ReDSHIFT software is thought as a tool for spacecraft operators, space agencies and research institution to design the end-of-life of any Earth-based mission and to study the interaction with the space debris environment.
In this talk the general description of the tool will be given together with a detailed description of the modules currently in the more advanced development state: the disposal module, via impulsive manoeuvres (manoeuvre module) and solar and drag sails (solar dynamics module), the spacecraft population interaction module and the re-entry module.
Given the initial orbit of the spacecraft and the spacecraft characteristics in terms of its cross area and mass, the optimal options for end-of-life disposal are given and compared; namely end-of-life disposal via one or a sequence of manoeuvres, end-of-life disposal through the use of a solar/drag sail or end-of-life through a hybrid sail + manoeuvre approach. This module is based on a study of the natural orbit evolution in the low to medium and geostationary regions that was performed to identify long-term stable orbits or resonance conditions to be used as graveyard or re-entry trajectories. The optimal manoeuvre to reach such re-entry or graveyard conditions is calculated. Moreover, the re-entry can be enhanced through a sail. In this case, different strategies for sail attitude control were previously compared and selected.
The optimal disposal by this module is passed to the space environment module so that the effect

of this disposal on the space debris environment is calculated. This is done based on precomputed long-term simulations of the whole space debris environment, under different scenarios, to be used for the computation of the collision risk for the spacecraft in the disposal phase.
In the case the disposal trajectory is a re-entry one, the condition of the orbit at 120 km are used to verify the demisability of the spacecraft. This is done, by default, using some predetermined spacecraft configuration but the external user can also load a preferred configuration.
All the modules are interfaced and linked, in a carefully triggered processing chain, through the openSF simulation framework properly configured and customised to adhere to the needs of the ReDSHIFT SW tool.
It is the aim of the ReDSHIFT tool to contribute in a proactive way to the mitigation of space debris problem via passive end-of-life mitigation.
The research leading to these results has received funding from the Horizon 2020 Program of the European Union's Framework Programme for Research and Innovation (H2020-PROTEC-2015) under REA grant agreement n. [687500]- ReDSHIFT.

Summary:

Optimization and Dynamics #2 / 36

Revisiting Design Aspects of a QP Solver for WORHP

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SQP methods for nonlinear programming rely on a quadratic programming solver for computing a search direction in each major iteration. From the start, the large scale NLP solver WORHP has been using the interior point method QPSOL within its SQP framework, which was developed specifically for WORHP. Experience from usage of WORHP in many areas and development of features like sensitivity analysis and feasibility refinement raised interest in a reworked, extended interface between WORHP and its QP solver. Furthermore, additional concepts like multiple centrality correctors seemed promising for improving the overall performance.

Hence, a revised QP solver was designed and implemented. Mehrotra's algorithm that was implemented in QPSOL was extended by Gondzio's multiple centrality correctors and weighting of corrector steps was added. Special care was taken to handle the very general NLP formulation of WORHP efficiently, yielding a very general problem formulation for standalone quadratic programming as well. A clear interface was implemented for retrieving sensitivity derivatives from the quadratic solver directly, allowing WORHP Zen and feasibility refinement procedures easy access to them.

The talk deals with these algorithmic and interface aspects for the development of the new solver within WORHP. Numerical results on the CUTEst test set for nonlinear programming are presented to show the performance improvements over the previous method.

Summary:

Low Thrust #1 / 35

SOLAR-SAIL TRANSFERS FROM INVARIANT OBJECTS TO L5 PERIODIC ORBITS

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The continuing development of solar-sail technology in combination with the rising interest in a mission to the Sun-Earth L_5 point for heliophysics and the search for Trojan asteroids, raises the question of using solar sailing as the primary propulsion method to enable such a mission. This paper therefore investigates a range of solar-sail transfers to the L_5 point, departing from different invariant objects in the neighbourhood of Earth: natural and solar-sail displaced equilibrium points, families of periodic orbits and their associated stable invariant manifolds. Also the arrival conditions are varied to be either natural or solar-sail displaced periodic orbits around the L_5 point. The transfers are obtained using a hybridisation of different trajectory design techniques. First, a multi-objective genetic algorithm is applied to obtain near-feasible initial guesses, which are transformed into feasible transfers using a differential correction method. Through a continuation on the fixed time of flight, the differential corrector is subsequently used to reduce the transfer time. As the differential corrector implements a stepwise constant control of the solar-sail attitude, a pseudospectral optimisation method is finally taken at hand to obtain a smooth, continuous control profile, to, if possible, further reduce the transfer time. This approach results in fast solar-sail transfers of 396 to 1194 days, depending on the departure and arrival configuration and the assumed solar-sail technology. These results can serve as preliminary design solutions for a mission to the Sun-Earth L_5 point.

Please find an extended abstract in the attached pdf-file

Summary:

On-orbit servicing and proximity #1 / 22

Safe natural far Rendezvous approaches for Cislunar Near Rectilinear Halo Orbits in the Ephemeris model

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In the context of future Human Spaceflight exploration missions, Rendezvous and Docking (RVD) operational activities are mandatory and critical for the assembly and maintenance of cislunar structures 1. The Orion spacecraft is expected to handle cargo delivery and crew exchange missions that will all require RVD. Despite extensive experience in Low Earth and Low Lunar Orbits, no operational RVD has yet been performed in the vicinity of the Lagrangian points.

The scope of this research is to investigate the specifics of orbits of interest for RVD in the cislunar realm and to propose innovative strategies and trajectory designs to safely perform these kinds of operations. With a focus on Near Rectilinear Halo Orbits (NRHO), previous work has investigated close rendezvous relative dynamics using linear and non-linear targeting algorithms [2]. Current research focuses on far rendezvous approaches and the investigation of passively safe drift trajectories in the ephemeris model. The goal is to exhibit phasing orbit requirements, given a prescribed target orbit, that ensure safe free motion and natural approach of a spacecraft near the target while exhibiting low cost transfer capabilties.

Ephemeris representations of NRHOs were derived using time-varying multiple shooting and adaptive Long-horizon targeting algorithms, resulting in orbit maintenance budgets comparable to those found in the literature [3]. The choice of the Long Horizon parameter appeared to be closely related to the properties of the NRHO and the benefits of an adaptive parameterization of the algorithm was

discussed. Simulations showed significant drift and overlapping properties for phasing and target orbits of interest. This motivated the research of safe free drift trajectories with NRHO-like motion in the Ephemeris model, using impact prediction strategies derived from debris avoidance analysis. Such strategies include a local-plane crossing algorithm to perform a quick wide search for potential overlapping sections, later refined by means of screening volumes with probability of collision computations using covariance analysis.

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Summary:

Loitering, Orbiting #2 / 60

Semi-analytical Framework for Precise Relative Motion in Low Earth Orbits

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Intelligent Tools and Assistants #1 / 30

Semi-analytical Framework for Precise Relative Motion in Low Earth Orbits

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Several multi-satellite mission architectures in Low Earth Orbits (LEO) as formation-flying, spacecraft clusters and active debris removal ask for accurate modeling of the relative motion between objects in neighboring orbits. The closer the region of interaction, the higher the level of autonomy the Guidance Navigation and Control (GNC) system may require to accomplish the mission's tasks. Hence, a precise semi-analytical framework reveals a convenient tool to support the development of efficient relative GNC algorithms.

In this context, Orbital Elements (OEs) based parameterizations are often exploited since, from the one hand, the linearization with respect to the orbit of the chief satellite remains accurate enough for quite large relative distances. From the other hand, the development of guidance algorithms can exploit the planetary variational equations to find the most efficient locations of the orbit correction maneuvers. Among the various parametrizations proposed in the literature, here the Relative Orbit Elements (ROEs) inherited from the co-location of geostationary satellites and afterwards adapted to the formation-flying field are employed. These are non-singular for small eccentricities, allow easy inclusion of the concept of passive safety of the formation through a certain relative eccentricity/inclination vector separation, and support a straightforward geometrical interpretation of how the geometry of the relative orbits changes under the effect of impulsive maneuvers.

The inclusion of the effects of the first terms of the geopotential (e.g., till 6th order and degree) modeling the non-homogenous Earth mass distribution can be conveniently accommodated in the ROEs

framework. First, short- and long-periodic terms are removed through an analytical transformation combining a Lie-series based approach, closed form in eccentricity, and the Kaula method. Afterwards, the ROEs secular variations, due to J_2, J_2^2, J_4 and J_6 terms, are recovered through the first order expansion of the time derivatives of such mean set with respect to the chief's orbit.

Differential aerodynamic drag is the second dominant perturbation in the LEO region, with an effect proportional to the differential ballistic coefficient between the considered spacecraft. As for OE-based formulations, the proposed techniques so far either exploit an engineer approach or a physical one. The first method relies on a general empirical formulation of the differential drag acceleration to include the mean effects produced on the ROEs. Whereas the latter methodology directly expands the time derivatives of the averaged OEs subject to a drag acceleration with an exponential density model. Both approaches require the inclusion of additional parameters to the ROE state variables and the investigation of their physical meaning, as well as their relationship, is beneficial to understand their effect on the ROEs' evolution.

In addition to the description of the aforementioned building blocks, the paper focuses on their interfacing, to set-up a fully consistent framework. The resulting performances in terms of achievable propagation accuracy are critically presented together with the framework's validity ranges.

Summary:

Interplanetary Flight and Non-Earth Orbits #3 / 38

Semianalytical Design of Libration Point Formations

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See the pdf file attached.

Summary:

A novel semianalytical technique is proposed for the libration point formation design. It is based on the use of Lindstedt-Poincare (LP) series that approximate the center manifold in the vicinity of the libration point. Any performance factor can be constructed by symbolically manipulating the LP series. For a number of typical formation-keeping objectives, the optimal design parameters are first obtained analytically in a low-order approximation (n=1,2) and then exploited as an initial guess in the numerical optimization procedure for the 15th-order approximation model. The proposed technique is robust, constructive, and versatile: while avoiding the necessity of numerical integration in the highly unstable dynamical environment, we effectively use the full hierarchy of center manifold approximations. The numerical optimization for the high-order approximation model is still low-dimensional and alleviated by a good initial guess obtained from low-order approximation models.

Satellite Constellations and Formations #1 / 26

Stochastic Constellation Replenishment Planner

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In the frame of the European GNSS Evolution Programme (funded by ESA), Deimos Space studied, designed and developed a stochastic simulator with the objective of computing and trading-off different constellation replenishment plans, that are able to guarantee a given service availability of satellite constellations, for any satellite constellation (GNSS, Satcom, etc).
During the first phase of the study, a set of replenishment strategy candidates have been designed, characterised in terms of launchers, spacecraft types, launch scenarios and transfer to the Constellation final slot. A comprehensive trade-off definition has been performed and a sub-set of candidates have been selected as the most promising ones to be further analysed by means of the Replenishment Planner. To perform the replenishment strategy candidate characterization, different methods have been used, spanning from bibliography research to low-thrust transfer optimization using an improved Edelbaum method which takes into account Earth eclipses.
In a second phase of the study, the Replenishment Planner has been thus designed and implemented, based on a stochastic approach in order to calculate the probability of having a certain number of satellites in each plane of a user-given constellation. These statistics combined with satellite and launcher reliability figures, an input provided by system studies, give the key figure to be maintained by the simulator: the service availability.
The Replenishment Planner is able to maintain the system service availability above a user defined threshold by means of different launch strategies, and using the selected strategy candidates as "building blocks". The simulator enables flexibility to accept as inputs generic satellites types, configurable launchers and transfer strategies (direct injection, electric thrust transfer to one or several planes, and staggered separation to inject satellites into different orbital planes at different altitudes), and a wide set of conditions and constraints. As launch strategies, the simulator is able to trigger corrective launches to replace failed satellites, preventive launches to anticipate the decrease of the service availability, preventive launches to avoid the loss of spare satellite capability, or a combination of the previous. The simulation can also consider satellite decommissioning.
The tool is able to run simulations in the order of one hundred years, considering the decay of the current satellites and the replacement with given future satellites, using different launchers, computing relocation of spare satellites and considering temporary outages.
Study cases have been run to validate the novel approach implemented in the Constellation Replenishment Planner, and the preliminary analyses yield promising results.

Summary:

In the frame of the European GNSS Evolution Programme (funded by ESA), Deimos Space studied, designed and developed a stochastic simulator with the objective of computing and trading-off different constellation replenishment plans, that are able to guarantee a given service availability of satellite constellations, for any satellite constellation (GNSS, Satcom, etc).

Ascent #1 / 58

Stochastic Constellation Replenishment Planner

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Open Source Tools and Smart Computing #2 / 63

TUDAT: the open-source astrodynamics toolbox of Delft University of Technology

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Open Source Tools and Smart Computing #1 / 39

TUDAT: the open-source astrodynamics toolbox of Delft University of Technology

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TUDAT (TU Delft Astrodynamics Toolbox) is an open-source, general astrodynamics toolbox, with a focus on numerical state propagation, that has been under development by staff and students of the Astrodynamics and Space Missions (AS) section of Delft University of Technology (TUD). Since early 2015, the software has been hosted as a github repository and is freely available under the BSD-2 license. Feature documentation and numerous tutorials can be found at tudat.tudelft.nl. For the last several years, TUDAT has become a key part of the Space Exploration MSc curriculum at TUD. Moreover, TUDAT has been used in dozens of MSc projects, 5 PhD projects, and has contributed to over 10 peer-reviewed publications.
TUDAT functionality falls into the following broad categories: numerical propagation of dynamics, precise orbit determination, and mission design & optimization. Key aspects of the TUDAT software are: modular software design, rigorous unit testing, the use of modern computing paradigms, and the use of numerous external libraries.
TUDAT has recently been extended with an input layer that makes use of JSON (JavaScript Object Notation) files. This extension currently allows users to propagate orbits numerically using the full power of the TUDAT C++ software. Although the JSON interface does not yet encompass the full TUDAT functionality, there are no fundamental obstacles to fully extending the interface.

Summary:

Clean Space and Environment Modelling #1 / 4

Tiangon-1 re-entry follow-up by CNES

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As French National Space Agency, CNES is in charge of monitoring safety requirements for people and property related to space operation as defined in French Space Operation Act (FSOA). To evaluate these requirements, and in particular to be able to assess the compliance with safety threshold, CNES has developed its own tools, DEBRISK and ELECTRA. DEBRISK software computes the ablation of the satellite and its components all along the reentry trajectory. This object-oriented code combines models for aerodynamic, aerothermodynamic and heat transfer. It supplies a list of surviving objects with their physical on-ground characteristics. This list is then used as an input for the ELECTRA software. ELECTRA software is designed to estimate human casualty risk during launch and controlled or uncontrolled re-entry. Using Monte-Carlo simulations, dispersion of various parameters as for example characteristics of surviving fragments and population grids, Electra computes the probability of incurring at least one victim on ground, the expected value of the number of victims and the impact risk per country. Among the four computation modes of Electra, the RF mode (final re-entry) is dedicated to the computation of risk a few hours or days before the un-controlled reentry of a space object.

One particular case of use of ELECTRA and DEBRISK occurred in 2018, when the Chinese space station Tiangon-1 re-entered the atmosphere and felt in the South Pacific Ocean on April 2nd. With

8,5 tons, this uncontrolled reentry was of large interest, nevertheless not concerned by the French Space Operation Act. In the frame of this high risk reentry, CNES made use of its flight dynamics technical expertise, the observation means activated through national and international cooperation and a variety of tools, both operational and expertise tools, as ELECTRA and DEBRISK, to predict the reentry date and location as well as the on-ground casualty risk evolution inferred by this high risk reentry.

First, this paper presents how the Tiangon-1 space station was modelled and how the ablation phenomena, induced by the atmospheric reentry, was computed using DEBRISK tool. Second, this paper deals with the risk computation using ELECTRA tool via its final reentry (RF) mode. The orbital parameters used as inputs for these tools were provided by OCC division in CNES who's in charge of operational orbit determination and to operationally estimate the reentry point and date of reentries considered as high risk. The evolution of the orbit and the reentry prediction is presented in parallel with the evolution of the risk assessment.

Summary:

This paper describes the prediction and risk estimation performed in CNES for the re-entry of the Chinese Space station Tiangon-1 in April 2018 using flight dynamics technical expertise, observation means and CNES tools, as DEBRISK (ablation computation) and ELECTRA (risk computation).

Interplanetary Flight and Non-Earth Orbits #3 / 41

Trajectory Design in High-Fidelity Models

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The design of space missions is generally driven by severe requirements on the Delta-v budget. Navigation is also becoming more and more challenging, asking for the satisfaction of stringent conditions characterized by unprecedented accuracy. As a consequence, an increased complexity in the trajectory design is needed, ultimately leading to employing high-fidelity models already in the early stages of trajectory design.

Flying in highly nonlinear gravity fields allows exploiting unique features, such as libration point orbits, ballistic captures, and low-energy transfers. These features are achieved by exploiting the sensitivity in initial conditions of highly nonlinear environments, and open up new scenarios for spacecraft characterized by very limited thrust authority.

In this talk, the tools developed at Politecnico di Milano for high-fidelity trajectory design will be presented. These include ULTIMAT (Ultra Low Thrust Interplanetary Mission Analysis Tool) for design and feasibility assessment of limited control authority missions, GRATIS (GRAvity TIdal Slide) for the computation of ballistic capture orbits, DIRETTO (DIREct collocation Tool for Trajectory Optimization) and LT2O (Low-Thrust Trajectory Optimizer) for the direct and indirect optimization of space trajectories in multi-body models, respectively.

Summary:

Low Thrust #1 / 54

Trajectory Optimization of a Low-Thrust Geostationary Orbit Insertion Maneuver for Spacecraft Total Ionizing Dose Reduction

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We consider the problem of optimal low-thrust spacecraft geostationary orbit (GEO) insertion from initial circular orbit with 800 km height and 51.6 degrees inclination. Minimal time for electric propulsion insertion of considered nuclear powered heavy spacecraft 1 is about 117 days. Significant amount of this time (~90 days) the spacecraft spends in regions with harsh space radiation environment inside the Van Allen radiation belts. To reduce an absorbed total ionizing dose (TID) for onboard electronic systems we propose a method of changing shape of the insertion trajectory and examine efficiency of this method. The main idea of the proposed method is to consider TID as a part of state variables set and to add the equation for TID change over the time to the equations of motion. Then if we add to the low-thrust optimal time GEO insertion problem 2 with the new set of state variables a condition of fixed TID at the end of transfer, one could obtain trajectories with lower final TID values. The obtained optimal control problem was solved using the maximum principle for one orbit time-averaged equations of motion. For numerical solution of corresponding boundary value problems we used the numerical continuation method with the predictor-corrector scheme. Numerical integration of considered ODEs was performed using the DOP853 integrator code.

The dose calculation for obtained trajectories we performed with CmdLineAe9Ap9 [3] software and Python scripts using AE8/AP8 MIN, AE8/AP8 MAX and AE9/AP9 electron and proton flux models of the Van Allen Radiation belts. In order to tackle discontinuities in right parts of ODEs we constructed a two-step spline approximation scheme of dose rate function. The first step is cubic smoothing spline approximation for noise level reduction of the CmdLineAe9Ap9 dose rate data. The second step consists of high-order (11th or higher) usual spline interpolation of the smoothing spline values. High-order order spline is needed to meet smoothness conditions of the right parts for Dormand-Prince ODE integration method.

We managed to reduce the final TID values by 25-38% (depending on the flux model) of final TID value on the minimal time GEO insertion trajectory. The duration of GEO insertion transfer is increased by no more than 7% of the minimal insertion duration. The additional required characteristic velocity for obtained trajectories is 320-560 m/s (depending on the flux model) with respect to the minimal time insertion trajectory.

References
1 Legostaev V. P. et al., Prospects for and Efficiency in Application of Space Nuclear Power Plants and Nuclear Electrorocket Propulsion Systems, Space Engineering and Technology, 2013, No. 1., P. 4-15. (in Russian).
2 Petukhov V. G. Optimization of Multi-Orbit Transfers Between Noncoplanar Elliptic Orbits, Cosmic Research, 2004, Vol. 42, No. 3, P. 250-268.
[3] https://www.vdl.afrl.af.mil/programs/ae9ap9/downloads.php

Summary:

Tutorial #3 / 73

Trajectory and multi-disciplinary design optimisation

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optimization methods and related modelling related to spacecraft design

Open Source Tools and Smart Computing #2 / 64

Visual Orbit design using the open-source VOD tool

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Open Source Tools and Smart Computing #1 / 42

Visual Orbit design using the open-source VOD tool

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While modern trajectory calculation and optimization tools are very effective, sometimes it is still useful to manually find an orbit for e.g. a necessary initial guess, or simply in order to understand the size and shape of the orbit.

A simple Visual Orbit Design (VOD) tool was created that allows changing orbit parameters such as apogee, perigee, delta-V given at a point, etc. while immediately seeing the result of the change within a 3D visualization tool. A slide-bar allows changing the time to see if a target such as the Moon is reached or not. The human brain, fed by a clear 3D overview, is very quick in understanding what changes need to be applied to achieve convergence.

Some examples of complex trajectories such as a Weak Stability Boundary transfer to the Moon and a Halo orbit are shown. It is shown that by 1) proper preparation of the trajectory, e.g. where/when to give a delta-V, when to stop a propagation, and 2) the correct selection of the view, and 3) the correct parameters to be changed, these complex trajectories can be found in a simple way.

The tool is open source and is shown as an HTML based use interface within the Systems Tool Kit tool

Summary:

Using the human brain and good 3D visualization, complex orbits can be found quickly using a simple Visual Orbit Design tool, and proper understanding of the orbit concept and an optimal view to show the orbit.