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DELTA (Debris Environment Long-Term Analysis)

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In this paper, we present the ESA Debris Environment Long Term Analysis (DELTA) tool, used to analyse the long term propagation and evolution of the future debris environment. DELTA is one of the models that contribute to the IADC studies on long term evolution, which have already been used to derive the mitigation guidelines and have also underlined the need for Active Debris Removal (ADR). DELTA is a three-dimensional, semi-deterministic model, which allows a user to investigate the evolution of the space debris environment and the associated mission collision risks in the low, medium and geosynchronous Earth orbit regions over user defined timespans. DELTA is able to examine the long-term effects of different future traffic profiles and debris mitigation measures, such as passivation and disposal at end-of-life, and also to take into account remediation measures, with the possibility to perform active debris removal in a variety of scenarios with different criteria. DELTA uses an initial space object population as input and forecasts the evolution of all objects larger than a user-defined size. The population is described by representative objects, evolved with a fast analytical orbit propagator which takes into account the main perturbations. The initial population is usually extracted from ESA's MASTER-2009 (Meteoroid and Space Debris Terrestrial Environment Reference) model at a given epoch, and can consider objects down to 1 mm in size. DELTA uses a set of detailed future traffic models for launch, explosion and solid rocket motor firing activity. They are each based on the historical activity of the preceding years. The collision event prediction is done by using a target centred approach, developed to stochastically predict impacts between all objects within the DELTA population. The fragmentation, or break-up, model used is based on the EVOLVE 4.0 (NASA) break-up model. In this paper, we show in detail the architecture of DELTA. Furthermore, we explain its singular way of computing the probability of collision, which is flux-based, different to the majority of the long term evolution tools which use a CUBE method. Finally, some sample results of simulations performed with DELTA are shown in order to display the large range of possible scenarios and applications that such a tool has.

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Spiderman Spacecraft: Tethered Asteroid Hopping in the Main Belt

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The use of the gravity of celestial bodies for gravity assist maneuvers is quite common in astrodynamics. The spacecraft gravitationally interacts with the celestial body in such a way as to provide the desired delta-v to the spacecraft. While this works well for large bodies such as planets, the gravitational attraction of small bodies, such as asteroids, is typically too small to perform such maneuvers.
In this work we analyze a different type of fly-by orbit using a variable length tether. During a fly-by of the spacecraft at an asteroid, a tether is attached to the asteroid which is then reeled out maintaining a tension within the limits of the tether. A dynamical model describing such a tethered flyby is developed. Using this model, it is possible to perform fly-by maneuvers that conceptually are very similar to traditional gravity assist maneuvers. While current tether technology is not yet able to provide the required tension, these fly-bys provide a new class of orbits that can be utilized in future for missions.
We demonstrate a potential use of such tethered fly-by maneuvers for multiple rendezvous orbits. A spacecraft in the main asteroid belt has to visit a long sequence of asteroids via tethered fly-bys. The optimization is performed with the goal of maximizing the number of asteroids visited in a

given time frame. The computation is performed using the PyKEP toolbox developed by the Advanced Concepts Team. Due to its flexible design, this Python based toolbox can easily be to allow for optimization of missions using these novel fly-by dynamics. The global optimization is performed using either evolutionary algorithms or a tree search strategy.

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Probabilistic orbit lifetime assessment with ESA's DRAMA/OSCAR tool

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ESA's space debris mitigation policy has come into force in March 2014 and adopts the space debris mitigation technical requirements from the ECSS adoption notice of ISO 24113. Those requirements include recommendations on the disposal of systems that have reached the end of their useful life, and were driving the development of OSCAR (Orbital SpaceCraft Active Removal). Specifically, OSCAR is the component of DRAMA designed to address disposal manoeuvres at end-of-life and assess the compliance of the later stages of a mission with the mitigation requirements, where mission planners have to decide on their implementation at early stages in the project (typically around SRR/PDR). In its current version, published and freely available within the DRAMA software suite, OSCAR allows for the analysis of the orbit evolution subject to different possible future scenarios for solar and geomagnetic activity, which are the main drivers in the estimation of the residual lifetime for a specific orbit.
Based on standardised and widely accepted methods for the prediction of those scenarios, the remaining orbit lifetime is computed via a semi-analytical propagation taking into account all relevant perturbations.
However, long-term forecasts of the orbit evolution are very sensitive to several quantities, most of them very difficult to forecast, including the solar and geomagnetic activity, the object's cross-section, its attitude state and mass, the drag and solar radiation pressure coefficient of the object, as well as physical model limitations. Moreover, the uncertainties in the injection manoeuvre transferring the spacecraft into its disposal orbit and uncertainty in the disposal epoch cannot be neglected. With OSCAR being used in the compliance verification process with respect to the mitigation requirements, the current approach to also assess the uncertainty associated with a lifetime estimate shall be discussed. By accounting for the various sources of uncertainty, OSCAR will allow for a more probabilistic and thus more realistic estimate, which is beneficial in the compliance analysis.
This paper will give a brief overview on the core functionalities of OSCAR and then focus on the uncertainties considered in the latest update of OSCAR. The propagation of the uncertainties results in a probabilistic assessment of the orbit lifetime. For example, the 25-year-rule compliance can then be based on an assessment in how many cases the orbit lifetime would be below or above 25 years.
Finally, some exemplary results will be provided, addressing different orbital regimes.

Open Source (I) / 130

An Update on NAIF's Package of "SPICE" Astrodynamics Tools

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"SPICE" (*) is an information system, comprising both data and software, providing engineers and scientists with the geometry data needed to help design robotic solar system missions, conduct mission engineering operations, plan observations from instruments, and analyze the data returned from those observations. The SPICE system has been used on the majority of worldwide planetary exploration missions since the time of NASA's Magellan mission to Venus, and it appears to be the ancillary information system of choice for most future solar system exploration missions. Along with its "free" price tag, portability and the absence of licensing and export restrictions, its stable, enduring qualities and substantial user support in terms of training and consultation help make it a popular choice.
A description of SPICE was presented at the original ICATT symposium in July 2001. This presentation will bring attendees up-to-date on the current capabilities of SPICE and plans for its further evolution. Part of this presentation will include a demonstration of a new visualization tool: Cosmographia. A companion presentation will highlight the WebGeocalc geometry engine.
The research described in this publication was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.
(*) Spacecraft, Planet, Instrument, Camera-matrix, Events

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SMART-UQ: Uncertainty Quantification Toolbox for Generalized Intrusive and Non Intrusive Polynomial Algebra

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The paper is presenting a newly developed modular toolbox named Strathclyde Mechanical and Aerospace Research Toolbox for Uncertainty Quantification (SMART-UQ) that implements a collection of intrusive and non intrusive techniques for polynomial approximation and propagation of uncertainties. Non intrusive methods build the polynomial approximation of the uncertain states through sampling of the uncertain parameters space and interpolation. Intrusive methods redefine operators in the states model and perform the states evaluation according to the newly defined operators.
The main advantage of non intrusive methods is their range of applicability since the model is treated as a black box hence no regularity is required. On the other hand, they suffer from the curse of dimensionality when the number of required sample points increases. Intrusive techniques are able to overcome this limitation since they have lower computational cost than their corresponding non intrusive counterpart. Nevertheless, intrusive methods are harder to implement and cannot treat the model as a black box. Moreover intrusive methods are able to propagate nonlinear regions of uncertainties while non intrusive methods rely on hypercubes sampling.
The most widely known intrusive method for uncertainty propagation in orbital dynamics is Taylor Differential Algebra. The same idea has been generalized to Tchebycheff and Newton polynomial basis because of their fast uniform convergence with relaxed continuity and smoothness requirements. However the SMART-UQ toolbox has been designed in a flexible way to allow further extension of the intrusive and non-intrusive methods to other basis.
The Generalized Intrusive Polynomial Expansion (GIPE) approach, implemented in the toolbox and presented here in the paper, expands the uncertain quantities in a polynomial series in the chosen basis and propagates them through the dynamics using a multivariate polynomial algebra. Hence the operations that usually are performed in the space of real numbers are now performed in the algebra of polynomials therefore a polynomial representation of the uncertain states is

available at each integration step. To improve the computational complexity of the method, arithmetic operations are performed in the monomial basis. Therefore a transformation between the chosen basis and the monomial basis is performed after the expansion of the elementary functions.
Non intrusive methods have been implemented for a set of sampling techniques (Halton, Sobol, Latin Hypercube) for interpolation in the complete polynomial basis as well as on sparse grid for a reduced set of basis.
In the paper the different intrusive and non intrusive techniques integrated in SMART-UQ will be presented together with the architectural design of the toolbox. Test cases on propagation of uncertainties in space dynamics with the corresponding intrusive and non intrusive approaches will be discussed in terms of computational cost and accuracy.

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CelestLab: Spaceflight Dynamics Toolbox for Mission Analysis

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CelestLab is a free and open source spaceflight dynamics Scilab toolbox developed by CNES that is particularly suited to mission analysis. The toolbox contains about 250 functions related to coordinate systems (IERS 2010 conventions), orbit propagation, geometry and events, force models, orbit properties and more. Lots of examples, demos, help pages and tutorials are provided which makes it easy for new users to get started.
The first version was put on ATOMS website (website from which Scilab modules can be downloaded) at the end of 2009 (see http://atoms.scilab.org/toolboxes/celestlab). The module is one of the most popular as the number of downloads for all versions is now over 34000. CelestLab is 100% Scilab language, which makes it easy to install or modify as there is no need for compilers for instance. But there are situations where this approach is limiting. This is the case when open source software exists that we would like to use through Scilab. There are at least 2 main advantages to using existing software: less developing and testing effort is required, and also, the features that are made available are guaranteed consistent with the original versions (considered as standards). That's why CelestLabX (CelestLab extension module) has been created. It contains low level interfaces to available software. The last version of CelestLab/CelestLabX provides utilities related to TLEs (interface to C code for the propagation of Two-Line Elements from: http://celestrak.com/publications/AIAA/2006-6753) and to STELA (Semi-analytic Tool for End of Life Analysis, tool used at CNES and elsewhere for orbit long-term propagation in particular, see https://logiciels.cnes.fr/content/stela?language=en). The presentation will show CelestLab contents in more details and will more particularly focus on recent features. Concrete example of mission analyses based on previous work conducted at CNES will also be shown in order to illustrate some typical uses of CelestLab.

Debris, Safety and Awareness (II) / 135

Debris operational support through the web applications

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With the growing number of debris and satellites which ESA operate, the necessity for operators to mitigate collisions with debris is a reality. Meanwhile atmospheric re-entry of objects is also gaining lots of media interest.
In order to enhance ESA's Space Debris Office's (SDO) operational support to missions and to national alert centers, CGI has been developing web based tools which improve decision making speeds, decision quality and end user experience in general. The tools also allow to scale up operational support to more missions, be it ESA's or external ones.
What are being developed are essentially front-ends for displaying data from the SDO's databases or generated from SDO's computations. The display of data are in tables, graphic plots or 3D visualisations and is split in views which are specifically adapted to the role of the user, e.g. satellite operator, debris analyst or manager. Among the multiple functionalities, an important one is the possibility to launch the SDO's avoidance manoeuver optimiser CORAM and to display the resulting trajectory and close approach geometry.
The applications were developed with powerful web technologies, have a common visually appealing look-and-feel and have been developed with continuous user feedback, such that the ergonomics are adapted to the SDO's work flow as well as to its own varied customers. The latest version of the tools will be demoed during the presentation.

Low Thrust (II) / 134

Low-thrust trajectory design with fully analytic three-dimensional spirals

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We present a preliminary low-thrust mission design technique based on the family of generalized logarithmic spirals. This family is an exact and fully analytic solution to a specific tangential thrust profile, for which even the time of flight can be solved in closed-form. The method includes a control parameter that can be adjusted at convenience and the motion is three-dimensional. Advantages of the method are its speed, intuitive use, and the fact that it can reproduce trajectories generated with more sophisticated methods. Thanks to the conservation of two integrals of motion the versatility of the algorithm is improved significantly by introducing coast arcs, decomposing the trajectory in intermediate nodes, and the use of gravity assist maneuvers. Combining these techniques yields a flexible method for which all the computational load reduces to solving two equations with two unknowns. Examples of interplanetary orbit transfers are presented.

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Tools and Techniques Supporting the Operational Collision Avoidance Process at ESOC

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ESA's Space Debris Office provides a service to support operational collision avoidance activities. This support currently covers ESA's missions Cryosat-2, Sentinel-1A, Sentinel-2A, Sentinel-3A,

and the constellation of Swarm-A/B/C in low-Earth orbit (LEO). The support process is provided to third party customers, too.
We provide an overview on tools used in the mission design phase and during the operational phase. In addition, we briefly introduce the process control and data handling in the ESA process. During the mission design phase collision avoidance is studied. Here the focus is at the effect of warning thresholds on the risk reduction and manoeuvre rates. For such analysis ESA's DRAMA tool suite with the module ARES is available.
During operations collision avoidance needs to address conjunction event detection, collision risk assessment, orbit determination, orbit and covariance propagation. ESA's process based on the central tools CRASS and CORAM is implemented following a database-centric approach through a temporary local "mini-catalogue". This catalogue is based on Conjunction Data messages (CDM) and own operational orbits. Each forecasted conjunction event is analysed in an automated way, returning the approach details and an estimate of the associated collision probability. A wide range of contemporary collision risk estimation algorithms with covariance scaling is supported. Identified high-risk conjunction events are further assessed and mission-specific processes are in place for decision-taking and manoeuvre recommendation. CORAM is able to assess optimised manoeuvres considering various constraints.
The database is also used as the backbone for a web-based tool (SCARF), which consists of a visualisation component and a collaboration tool that facilitates both, the status monitoring and task allocation within the support team, as well as the communication with the control team. The web-based solution optimally meets the needs for a concise and easy-to-use way to obtain a situation picture in very short time, and the support of third party missions not operated from ESOC.

Optimization and Dynamics (II) / 138

Understanding concepts of Optimization and Optimal Control with WORHP Lab

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The ESA-NLP solver WORHP is already used in several academic and industrial projects in a wide range of applications, as aerospace, automotive or logistics. Currently over 500 users worldwide code their problem formulations using the the standard interfaces to Fortran, C/C++ and MATLAB.
To simplify the formulation of optimisation problems for demonstration and educational purposes WORHP Lab is developed as a graphical user interface (GUI). With a growing set of applied examples and visualisation techniques it shows the capabilities of the underlying solver WORHP and opens access to more involved concepts like parametric sensitivity analysis using WORHP Zen.
Furthermore, WORHP Lab provides the possibility to solve optimal control problems using our transcription method TransWORHP. Different approaches like full discretisation with grid refinement or multiple shooting are compared easily within this tool. Additionally, optimal control problems can be solved on reduced time horizons to illustrate concepts of nonlinear model predictive control.
WORHP Lab was already employed successfully in several industrial workshops as well as for educational purposes with pupils and students. In this talk we illustrate its features with aerospatial examples of optimisation and optimal control problems.

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Libration Mission Designer

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Since the Apollo mission epoch, the Libration points orbits play an important role in space missions design. Due to their interesting and uncommon size, shape and dynamical features, the Libration points result to be more attractive for many different and challenging space exploration missions, with objectives that ranging from the collection of solar radiation and data observation, to the positioning of new space human outposts, up to the targeting of Mars.
The gravitational forces of the two massive bodies Moon and Earth govern the entire S/C dynamic in the vicinity of the Libration points, resulting in orbital motions which are nominally different from the classical and well known Keplerian motion.
Through the formulation of a complete elliptical three-body problem (E3BP), the paper presents methods for analytical and numerical computing of libration point orbits, investigating in details some station-keeping manoeuvre techniques which allow to constantly maintain a generic S/C within the forecasted and selected orbits around those points, driving the definition of some important design constraints from AOCS and GNC system point of view. In addition, a brief overview of past and future Libration points missions are presented, together with new proposed ones.

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Trajectory generation method for robotic capture of a non-cooperative, tumbling target

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ABSTRACT
Recent studies of the space debris population in Low Earth Orbit (LEO) [**3**] have concluded that certain regions have already reached a critical density of objects, which will eventually lead to a cascading process called the Kessler syndrome [**2**]. Thus, there is a consensus among researchers [**3**] that active debris removal (ADR) is the only viable way of reducing the space debris population and preserving the space environment for future generations.
Upper stages appear to be among the most suitable targets, given their number, mass properties and spatial distribution. However, their non-cooperative nature and possible residual angular momentum makes them difficult targets for ADR missions. Moreover, a lot of research still needs to be performed in order to make ADR missions a reality. Furthermore, up until today the research community has failed to achieve consensus about the most optimal technique to employ, even in case of one particular target (e.g. ESA owned ENVISAT).
Nevertheless, among the proposed ADR technologies, those involving orbital robotics are the most mature ones since they have been successfully tested on-orbit in more than one occasion. Moreover, they are the most versatile ones given that they can be easily adapted for a wide variety of targets and their usage can also be also extended to on-orbit servicing and assembly. However, no robotic spacecraft has ever performed a capture of a completely non-cooperative, tumbling object.
To overcome this limitation, the on-board rendezvous and capture (RVC) control system of the spacecraft should be implemented with a particular attention to the robotic control subsystem that should perform the capture and take over the planning and control of the chaser spacecraft in the moments immediately preceding the physical contact. In fact, given the random character of the contact forces, the attitude and orbit control system (AOCS) of the spacecraft should be deactivated during the capture maneuver to avoid random shocks that could lead to a collision with the target or could damage the manipulator of the spacecraft. Moreover, grasping a target that has a residual angular momentum without considering it in the approach phase (as it was done in the past) could pose difficulties to the AOCS in the post-impact phase and most probably will result in a failed maneuver [**1**].

In this context, the following paper presents a method for trajectory generation of joints of a manipulator mounted on a free-floating base spacecraft, with the scope of using the non-linear dynamics of the base to capture a non-cooperative, tumbling target and minimize the angular momentum management of the whole stack in the post-capture phase. The presented method is based on the Bias Momentum Control (BMC) approach developed by Dimitrov D. N. et al and described in [**1**]. However, with respect to the latter work and existing literature, the novelty of the following work stands in considering an initial non-zero angular momentum of the base spacecraft, a free-floating base, deactivated AOCS of the spacecraft and a total zero angular momentum of the stack.
The configuration of capture problem considered in this paper consist of a 2D problem, with one degree of freedom (DOF) tumbling target (having a known constant angular velocity), a three DOF base spacecraft (having a known initial angular momentum) and four DOF manipulator mounted on top of the base spacecraft.
The trajectory planning method is formalized as an optimization problem performed in two steps. The parameters to be optimized are either joint angle velocities of the manipulator or the angular velocity of the base spacecraft. The imposed kinematic constraints are the end-effector final position and its linear velocity. The inequality constrains are the joint angles limits of the manipulator. The two step process consists of the: **(a)** computation of the initial configuration of the manipulator, so to satisfy the end-effectors final position and initial angular momentum of the base, **(b)** profile of either joint velocities or the base angular velocity in order to satisfy the inequality constrain and reach desired linear velocity of the manipulator immediately before the contact with the target.
The feasibility of the approach is verified with a 2D numerical simulation, implemented in MATLAB/Simulink computational environment, and the comparison of the results between the two sets of parameters to be optimized is assessed.

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Modelling and Simulation of Autonomous Cubesats for Orbital Debris Mitigation

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It is well known that orbital debris about Earth impose increasingly stringent restrictions on the operation and commissioning of both current and future space applications. These orbital debris, which are becoming ever so prevalent, can literally destroy a satellite. Even particles of diminutive stature can result in disastrous ramifications. Many of these endangered satellites, of which humans are reliant upon for supplying the infrastructure necessary to support modern life in the twenty-first century, routinely have to perform avoidance maneuvers in response to ground data indicating that an object is on a trajectory that could pose a threat, negating away precious finite amounts fuel. These orbital debris are vexatious, with respect to not only a spacecraft's integrity, but also its lifespan. It is imperative that a solution be realized. This study aims to demonstrate the feasibility of utilizing modular cubesats to deorbit space debris. A large high fidelity multidisciplinary simulation is constructed with the goal to simulate the cubesat's orbital and attitude dynamics, as well as its autonomous functions. Such autonomous functions will manifest in the development of autonomous control algorithms to execute mitigation procedures,

such as path planning and rendezvous, as efficiently as possible with respect to multiple subsystems' criteria.

Students (I) / 27

Indirect Planetary Capture via Periodic Orbits about Libration Points

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The libration points and periodic orbits possess unique dynamics properties in multibody system, which have been exploited to design low-energy transfer for space mission. In this paper, we investigate the periodic orbits for planetary capture and propose an indirect planetary capture method. In new method, the periodic orbit is consider as a park orbit during the capture, which connects with the interplanetary trajectory and the target orbit by stable and unstable manifolds, respectively, at the corresponding periapsides respective to the planet (see in Fig. 1).
The indirect capture method is researched under the background of Mars. Firstly, the dynamic model of CRTBP is established. The libration points and halo orbits for Sun-Mars system is established. Secondly, the candidate halo orbit for capture is determined according to the periapsis distance of invariant manifolds. The amplitudes of halo orbits that suitable for park orbits are given. L1 and L2 halo orbits provide different capture opportunity. The phase angles of invariant manifolds are discussed. Finally, the efficiency of indirect capture method is compared with the direct periapsis capture. The results show that the cost for indirect capture is approximately constant regardless of the periapsis distance (see in Fig. 2). The indirect capture method could save velocity increment significantly at the cost of long transfer time. The new method for planetary capture can also applied to other planets and provide reference for future exploration mission.

Loitering / Orbiting (I) / 20

Comparison of the Orekit DSST Short-Periodic Motion Model with the GTDS DSST and the F77 DSST Standalone Models

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Development of the DSST started in the mid 1970's at the Computer Sciences Corporation and continued at the Charles Stark Draper Laboratory and the MIT Lincoln Laboratory. These developments employed the non-singular equinoctial elements. The Draper Semi-analytical Satellite Theory used the GTDS orbit determination system as the development platform. However, users external to the Draper Laboratory wanted access to the Semi-analytical Satellite Theory without the 'overhead' of GTDS. The DSST Standalone program was developed in 1983-84. The Standalone included complete models for the mean element motion (based on the conventions then employed in GTDS) and a portion of the short-periodic model. The intent was to provide accuracy for LEO orbits of approximately 200 meters. By 1996, extensive improvements to the GTDS DSST had been made. These included 50 x 50 geopotential fields, solid Earth tides, and J2000 coordinate systems. In 1997, an effort to extensively upgrade the DSST Standalone was undertaken. This effort included improvements to the force modeling and to the maintainability of the source code. While the 1997 upgrade touched large portions of the DSST Standalone source code, testing primarily focused on the mean element equations of motion. In 2010, the first author

presented the paper "Open Source Software Suite For Space Situational Awareness And Space Object Catalog Work" at the ICATT meeting in Madrid. This paper proposed the migration of the DSST Standalone Orbit Propagator from Fortran 77 to an Object-Oriented software platform. In 2011, the implementation of the mean element motion portion of the DSST in the Orekit open source library was initiated. Implementation in the Orekit library involved migration of the DSST to the object-oriented java language and to a different functional decomposition strategy. Resolution of the F77 Standalone DSST code and documentation anomalies was an important product. Orekit DSST mean element predictions were compared with those produced with the F77 DSST Standalone. For several test cases involving several thousand day arcs, the Orekit and F77 mean element histories could not be distinguished. The DSST employs Fourier series for the short-periodic motion in the equinoctial elements. These expressions are closed form for the zonal, lunar-solar, and the tesseral m-daily terms. The Fourier coefficients in these expressions are functions of the slowly-varying mean elements (a, h, k, p, and q) and are slowly varying when plotted over time.1 The Fourier coefficients are computed 'off-grid' via an interpolation process. This Fourier coefficient interpolation must be compatible with both the high order Dormand-Prince and the classical RK integrators employed in Orekit DSST. This paper provides a description of the java classes adopted for the Orekit DSST with emphasis on the short-periodic models and the associated interpolation processes. The Orekit class DSSTForceModel includes the functions getMeanElementRate and getShortPeriodicVariations. This paper provides detailed comparisons of the Fourier coefficients computed by the Orekit DSST, F77 DSST Standalone, and GTDS DSST programs on a perturbation-by-perturbation basis. We also consider the overall accuracy that is possible with the present Orekit DSST implementation.

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SIRIUS-DV: The new Flight Dynamics algorithms for the future CNES missions

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The SIRIUS project aims to develop a set of Flight Dynamics products that will be used operationally in the control centers of the upcoming CNES missions. It mainly covers three different layers: the mathematical low level libraries (PATRIUS), intended to be used either in an operational environment or in expert studies; the flight dynamics algorithms, implementing the operational functionalities (SIRIUS-DV); and the FD applications, that include the assembly of the algorithms to build stand-alone applications – with dedicated GUI - and the infrastructure services (such as time, messages, logging, . . .) needed in an operational FDS.
Due to its unique architectural conception, SIRIUS provides a higher level of flexibility (so as to be easily adapted to any future mission, almost in a "plug and play" manner) and scalability (the effort to add new functionalities is reduced) with respect to other state-of-the-art systems. The choice of technologies used in the line of products also guarantees its non-obsolescence up to - at least - twenty years from now.
This paper focuses in the second layer, the software applications implementing the flight dynamics algorithms that are divided in several technical domains:

- Conversion: Dates conversion, orbital/attitude parameters conversion.
- Ephemeris Generation: orbit propagation and ephemeris generation.
- Events: events/phenomena calculation.
- Scenario: Functionalities dealing with the data scenario, which represents the whole mission of a given satellite
- Orbitography: measurement treatment, orbit determination, collision risk assessment.
- Orbital Maneuvers: orbital maneuvers computation, station keeping.

- Guidance & Programming: AOCS programming, guidance, constraints checking.
- Mission: reference orbit calculation.
- Monitoring: monitoring of TM parameters, thresholds checking.
- Interfaces: External data retrieval/production.
- Scenario Processings: treatment of the different parts of the data scenario, such as trajectory, attitude, maneuvers, MCI, thrusters, tanks and solar arrays

These algorithms rely on a data model managed by the CNES domain experts and which is updated gradually as the development advances. It contains the definition of all the data that are used in the algorithms, the definition of the interfaces (inputs/outputs) of each algorithm and the software requirements that the implementations must meet. Using this model as input, the implementations of both the data and algorithms interfaces are automatically generated (using a code generator that is also part of the SIRIUS line of products), which serve as starting point for the development carried out by the team.
The SIRIUS-DV applications are developed in Java using an Agile/SCRUM methodology with sprints (realization iterations) lasting four weeks. The functionalities to be developed in a given sprint are presented (at the beginning of each sprint) to the team by the CNES domain experts. During the sprint a constant communication flow is established between both parties in order to ensure the understanding – and hence the quality – of the tasks to be done (the development team being physically located at CNES premises). At the end of each sprint, those functionalities that are finished are presented to the users by the development team, so a fully usable product is available once a month, with increased functionalities over time.
This paper gives a brief description of the development process of SIRIUS-DV and describes the key concepts of this new line of flight dynamics algorithms, the data model and its impact in the developed software and several of the most representative applications, paying special attention to the architectural design of the propagator and its link with the data scenario, since it constitutes the core of the system.

Verification and Validation Methods / 22

Differential Algebra Space Toolbox for Nonlinear Uncertainty Propagation in Space Dynamics

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The problem of uncertainty propagation represents a crucial issue in spaceflight dynamics since all practical systems - from vehicle navigation to orbit determination or target tracking - involve nonlinearities of one kind or another. One topic of recent interest concerns for instance, the space surveillance and the accurate propagation of uncertainties on the initial conditions of resident space objects in order to identify and track them. Another relevant application consists of computing landing dispersion both for reentry missions and for estimating the casualty area of space debris. In addition, within the space mission design process, uncertainty propagation is a fundamental tool to assess the fulfillment of mission requirements and constraints, to evaluate mission performances, to perform sensitivity analyses, and to verify the robustness of guidance

and control laws. However, most spaceflight mechanics problems involve nonlinearities. This observation finds proof even in basic applications. For instance, many problems in celestial mechanics require conversions between different coordinate systems (e.g. the conversion from polar to Cartesian coordinates that form the foundation for the observation models of many sensors). Such transformations are typically nonlinear and therefore entail the problem of applying nonlinear transformations to the estimated statistics.
Although in the recent past, the power of computing hardware has been growing exponentially, giving rise to many new possibilities in the field of numerical mathematics, the vast majority of numerical algorithms developed to tackle the problem of uncertainty propagation, are still largely based on the same pointwise or linear algebra (matrix) techniques developed 50-100 years ago. In particular, two main families of techniques exist: the first one is represented by standard linear methods, where a linear approximation is introduced and used to perform uncertainty analyses. Despite the advantages in terms of computational time, they are typically characterized by a low level of accuracy, since the linear approximation holds only in a restricted region of the problem domain. The second one is represented by standard Monte Carlo techniques, which are able to provide very accurate results, but require large computational times and memory burden. In light of the above, the scientific community has started focusing on the development of new tools, able to improve the approximation of standard linear methods available in the literature or to reduce the computational time required by standard Monte-Carlo simulations.
Differential algebra (DA) perfectly suits with these requirements, providing a method to easily extend the linearization methods and allowing the implementation of efficient arbitrary order methods. The resulting technique performs much faster and yields more accurate results in many mathematical, physical as well as engineering applications. Dinamica, with the support of ESA, has recently completed the implementation of the Differential Algebra Space Toolbox (DAST), a DA-based software tool for the efficient, nonlinear propagation of uncertainties in space dynamics. DAST is divided in three main layers: the Differential Algebra Computational Engine (DACE), the Software Framework (SF), and the Uncertainty Propagation Tool (UPT). DACE provides the user with the tools to perform all the basic DA operations by replacing the operations between single numbers by suitably chosen operations on polynomials. The same sequence of operations coded for floating point numbers can thusly be evaluated using this new meaning of each operator almost without changes to the code. SF is built on DACE and comes with a set of more advanced features (e.g. vectors and matrices of DA, propagation schemes), specific astrodynamic routines (DA Kepler solver and DA Lambert solver) and ready-to-use dynamical models. Finally, UPT allows the user to propagate uncertainties through different techniques (from classical Monte Carlo to range estimation using polynomial bounders or classical linear covariance propagation) and perform statistical analyses.
Although the general mathematical foundation suggests potential applications of the same approach to various fields such as biology, automotive engineering, and finance, this work focuses on the application of DAST to a set of different test cases in the field of astrodynamics and space engineering, with the aim to illustrate the potentials with respect to classical approaches. Thanks to the use of DA techniques, indeed, DAST has been shown to be orders of magnitude more efficient than traditional methods for uncertainty propagation.

Ascent (I) / 23

Design of Optimal Observation Strategy for Re-entry Prediction Improvement of GTOs Upper Stages

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From 2004 up to the date more than 200 launch vehicles operated by five independent nations and two international organizations placed satellites in Geostationary Earth Orbit (GEO). In almost all cases, each successful launch left one or more pieces of debris in Geostationary Transfer Orbits (GTO). Particularly, many of this space debris consist of large spent upper stages of launch vehicles whose atmosphere re-entry might violate the constraint on casualty risk of 1/10000: as of 16 October 2014, it is expected that about 79 spent upper stages operating on GTOs with an inclination lower than 20 degree will enter the Earth atmosphere in the next 200 years. Moreover, the GTOs are highly eccentric orbits with perigee normally at low altitudes (170–650 km) and the apogee near geo-stationary altitude (35,780 km). Thus, space debris in GTOs generally passes through densely populated regions such as Low Earth Orbit (LEO) and GEO regions, being a hazard for the safety of other operating spacecraft. In light of the above, the improvement of re-entry prediction of GTO spent upper stages is a key issue to manage both on-orbit collision risk and on-ground casualty risk. Currently the only public data source available for re-entry prediction of a space object are represented by Two Line Elements (TLEs), provided by the United States Strategic Command (USSTRATCOM). However, this set of data are inaccurate and do not come with uncertainty information, making their use in re-entry prediction and conjunction analyses challenging, especially for the GTO space object. This leads to the need of using the observational data to improve the re-entry prediction. The design of observation strategy for GTO upper stage is not trivial. The detection and tracking of space objects on GTOs might require more than a single sensor in fact, since the distance from the observer has large variation along the orbit; this multiple sensors configuration might involve problems such scheduling or data fusion, making space object observation complex and costly. In addition, design of an optimal observation strategy for improvement of re-entry prediction involves the definition of a high-accuracy orbit determination (OD) algorithm, and the implementation of proper methods for uncertainty mapping. This might require the definition of accurate dynamical models in order to describe the effects of third-body perturbations and the Earth's oblateness and to capture the intricacies of re-entry phase, as well as the use of nonlinear technique for orbit determination. In this paper, a systematic approach to design the observation strategy of spent upper stage moving on GTOs is presented. More specifically, the design is formulated as a multi-objective optimization problem solved by means of a multi-objective genetic algorithm (MGA). This approach allows minimizing both the number of total measurements required to detect the space object and the error on re-entry prediction. Within the optimization process a nonlinear OD algorithm is run to determine the estimates of both initial state and model parameters. The Nonlinear Least Square Filter (NLSF) technique is implemented, exploiting the differential algebra framework to reduce the computational effort related to OD problem solution. Finally, the software tool IRIS is developed to accurately simulate the observation campaigns based on geometry and constraints of existing sensors currently available to European Space Agency (ESA).

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Computer Vision Automated Attitude Estimation from ISAR Images

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Determining the kinematic state of objects in space is a topic of major concern for both researchers and spacecraft operators. The scientists and engineers are, e.g., interested in the influence of the attitude changes on the orbit of the object, be it for long-term propagations of the state vector or for re-entry predictions, driven by the varying geometric cross-section. For the operators the capability becomes highly important in the case of contingency situations, when communications with the satellite might be lost and solutions have to be found.
Different approaches are currently being explored, such as laser ranging, light-curves (variation of the brightness of space objects in the visible region) or Inverse Synthetic Aperture Radar (ISAR) techniques.

Our work focuses on the latter, in which the apparent motion of the object w.r.t. a single radar station is used to determine the geometry and motion of the reflecting object. ISAR images have an imaging plane different than the one adopted by optic cameras which depends on the attitude state of the object, generally the unknown variable. The problem of unknown attitude states during the ISAR image generation is often practically solved by assuming a state. This can therefore limit the application of the results deduce from such images, e.g. to checking the presence of components on an object. However, under some assumptions, the attitude state can be estimated from sequences of images or when the shape of the object is known.
This paper presents the analysis and results of applying computer vision techniques to estimate of the pose of a space object only from ISAR images. More specifically, three scenarios have been evaluated: attitude estimation from a single image when the shape of the object is known; attitude estimation from a sequence of images still with knowledge of the object, and attitude estimation from a sequence of images without any information about the target. For the first two scenarios, the silhouette of the target in the ISAR image is first extracted and then compared against a database of reference images, each of them corresponding to the target under a different orientation, generated synthetically with tools able to simulate (entirely or, at least, partially) the ISAR process. The closest match will correspond to the actual orientation of the object. For the third scenario, point features are detected and tracked during multiple images of the sequence and used to recover simultaneously the shape and orientation of the target.

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An Interactive Trajectory Design Environment Leveraging Dynamical Structures in Multi-Body Regimes

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With the increasing complexity of cislunar and interplanetary missions, as well as the introduction of a wide range of scenarios involving small spacecraft, there is significant motivation to design trajectories that require fewer resources and may be sustainable over longer time intervals. Progress toward such goals is achievable by leveraging the natural dynamical structures available within multi-body regimes to guide the selection of a baseline trajectory. As demonstrated by previous missions, such as ARTEMIS, three-body dynamical structures can provide innovative trajectory design options. Such analysis can be particularly beneficial for upcoming mission concepts including exoplanet observatories, the redirection of asteroids, as well as lunar and interplanetary CubeSat missions. Tools that enable active selection of dynamical structures available in a multi-body regime may also supply the guidance necessary for rapid and well-informed trajectory design in chaotic environments.
The accessibility of regions of interest in a multi-body regime is a challenging metric to represent and the efficient selection of specific solutions is nontrivial for dynamically sensitive environments. In higher-fidelity multi-body models, the generation of a large set of solutions demands significant, and often prohibitive, time and computational resources. However, the Circular Restricted Three-Body Problem (CR3BP) is well known and this model offers a reasonable approximation to the dynamical behavior, e.g., in the Earth-Moon and Sun-Earth systems; periodic and quasi-periodic orbits in these dynamical environments also tend to persist in higher-fidelity models. Associated with these ordered motions are natural manifolds, which are useful in assembling low-cost transfers. In fact, researchers in the astrodynamics community continue to investigate a large number of solutions and techniques that may be useful for orbit design and operation within multi-body dynamical environments.
To incorporate knowledge of the dynamical accessibility of specific regions in the Earth-Moon and Sun-Earth systems into the trajectory design process, Purdue University, in partnership with NASA Goddard Space Flight Center, has developed a graphical and interactive design environment that enables rapid and well-informed construction of complex trajectories that leverage natural solutions. This trajectory design tool is comprised of several modules that offer

guidance into the leveraging of known dynamical structures for the active selection of trajectory arcs. For instance, to support various mission scenarios, an interactive catalog of periodic and quasi-periodic solutions in the CR3BP supplies the capability for straightforward design trades and orbital selection. Pre-computed libration point orbits are also available in some systems, with the option for on-demand manifold generation. Additional modules offer the capability to identify alternative periodic and quasi-periodic orbits using point-and-click selection on a Poincaré map. Incorporating maps (and/or surfaces of section) into the trajectory design process enables detection of additional orbit options and connections, and provides insight into the dynamical sensitivity. This interactive design tool enables rapid and well-informed construction of complex trajectories in a user-friendly environment that offers intuitive access to dynamical systems theory techniques including Poincaré maps. The end-to-end trajectories are then available for transition to higher-fidelity models, for refinement via the addition of various constraints, for input to other tools such as GMAT.

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High-fidelity small body lander simulations

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Recent missions to the small bodies of our Solar system have established a core understanding of the origins, characteristics, and dynamics of these pristine bodies. Furthermore, the feasibility and benefit of including lander platforms capable of performing in-situ (sub-)surface measurements was demonstrated most notably by Rosetta's Philae lander. The inclusion of such landers on small body mission motivates the need for capability to generate high-fidelity simulations of the landers' trajectories. In this work, we discuss and demonstrate the SAL tool, which provides that capability.
Our tool models small body environments at three distinct scales: at the large scale, the coarse shape and gravity field of the target body are represented using a constant-density polyhedron. The intermediate scale consists of rocks and boulders on its surface, which are recreated following statistics observed on asteroid Itokawa. Finally, at the small scale, interactions between the target body surface and a lander are captured using a contact dynamics model based on the normal, friction, and rolling resistance forces and torques.
The major challenge in simulating this environment results from the high resolution of the applied models. The computational load of detecting collisions between a lander and a small body shape model with hundreds of thousands of facets is reduced through the use of bounding boxes and the subdivision of the target body shape into local worlds. By constructing simplified gravity models from the high-resolution shape model, we reduce the cost of gravity field evaluations in exchange for only a small reduction in the gravity field accuracy. Generating the full distribution of millions of surface rocks is computationally unfeasible; instead we apply a procedural generation strategy that creates these surface features "on the go," and only on the active local world when the bounding-box collision module detects an impending collision. This provides a reproducible, low-cost technique for creating rock distributions. Finally, regional variations in the properties of the small body surface are captured by locally varying the coefficients of restitution, friction, and rolling resistance, which govern the energy dissipation during contact interactions.
Using this tool, we may carry out sets of Monte Carlo deployment simulations in which the release conditions, rock field parameters, and surface interaction coefficients are varied. The resulting trajectories allow mission designers to analyze the feasibility of a given deployment strategy, and establish related hardware requirements, such as the on-board battery capacity, which follows from the expected time-to-land. The trajectories' geographical spread provides information on the illumination, thermal, and scientific characteristics of the expected landing site. Finally, our tool may also be used to simulate surface mobility operations, in which a lander uses mobility devices such as reaction wheels to generate impulses, which allow it to travel on and over the small body surface in a series of "hops," enabling a single lander to obtain scientific measurements at multiple sites.

Orbit Determination and Prediction Techniques (I) / 94

Recursive estimation of non-gravitational perturbations from satellite observations

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Thanks to high-fidelity ephemeris and detailed gravitational maps, third-body and non-spherical gravitational perturbations can be modeled with sufficient precision for most applications in low-Earth orbit (LEO). On the contrary, owing to severe uncertainty sources and modeling limitations, mathematical models of the main non-gravitational forces – namely, aerodynamics and solar radiation pressure (SRP) – are generally biased even when advanced formulations are considered.
To date, accurate satellite drag and SRP estimation is only envisaged in challenging missions and the recursive estimation of non-gravitational forces is generally carried out by means of high-sensitivity accelerometers. Nonetheless, unmodeled force estimators using satellite observations only were also proposed. The method of dynamical model compensation (DMC) is arguably the most popular example of this class: first, an underlying parametric model of the unknown perturbation is adopted; then, the parameters of such model are assumed to be first-order Gauss-Markov processes and they are appended to the state vector of a recursive estimator (most often an extended Kalman filter). Provided accurate and sufficiently dense satellite observations, DMC was successfully applied to the estimation of atmospheric force. In that study, no other process noise but the one in the atmospheric force itself was considered.
In the broader context of Bayesian estimation of dynamical systems, sequential Monte Carlo (SMC) algorithms – which include the popular particle filters – are valuable tools to optimally approximate the posterior distribution of hidden Markov processes. Compared to Kalman filtering techniques, particle filters do not require any assumption on neither the linearity of the system nor the nature of the noise. SMC was used to tackle several problems in astrodynamics but, to the best of our knowledge, it was not applied to non-gravitational force estimation, yet.
In this paper, *we propose an SMC algorithm for the recursive inference of non-gravitational perturbations from satellite observations* with no supporting in-situ acceleration measurements. Our approach is conceptually similar to DMC but, on the top of the previously mentioned advantages and drawbacks of SMC, we show that it provides good estimates of the non-gravitational perturbations even when fairly inaccurate measurements and a modest underlying propagator are used. The filter works by updating the empirical distribution of a prescribed number of weighted particles. Each particle consists of one set of orbital elements and some parameters involved in the computation of the forces, e.g., drag and reflectivity coefficients. Weights are assigned to the particles based on the agreement between propagated orbital elements and observations. Secular effects of the non-gravitational perturbations allow "good" particles to emerge when the weights are updated.
Mean orbital elements are exploited as measurements. They can be obtained by either converting GPS states with a contact transformation or using two-line elements (TLE). This feature allows analytical and semi-analytical propagators, e.g., SGP4, to be naturally integrated in the algorithm to propagate particles. For these reasons, this work can be a valuable resource both for space situational awareness applications, e.g., space debris' characterization from TLE, and to enhance short-term trajectory predictions on-board small satellites.

Debris, Safety and Awareness (I) / 4

Conjunction Risk Assessment and Avoidance Maneuver Planning Tools

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The ever increasing number of objects in the near Earth region has been causing growing concerns about the space environment and accordingly about the safety of future space missions. Since most of orbital debris stay in the orbit for years, even a single collision between space objects could seriously increase the debris population, making further collisions more likely.
The German Space Operations Center (GSOC) has been performing collision avoidance operation since 2009. Additional to the operational satellites currently 5 in LEO and 2 in GEO, conjunction monitoring and mitigation for several satellites are supported, which are operated by other organizations or ended the operation phase. The Conjunction Data Message (CDM) provided by the Joint Space Operations Center (JSpOC) is currently the main source for the orbital information of the space objects due to the quality and timeliness of the available information. Conjunction prediction, risk assessment, and avoidance maneuver planning are performed automatically in the operational process.
In the collision avoidance operation, numerous conjunctions are reported daily, which were detected within certain thresholds. Early detection of possible critical conjunctions among all predicted events is therefore important to handle critical situations promptly and efficiently. Additionally, earlier estimation of a possible avoidance maneuver strategy is also important, because the decision of the avoidance maneuver execution is mostly done within one day before the closest approach based on the latest prediction. Especially, the operational satellites TerraSAR-X and TanDEM-X are flying in a very close formation with a minimum distance of ˜300 m, therefore the avoidance maneuver for both satellites shall be taken into account to handle close approaches of each encountering object. The avoidance strategies to meet the control requirements and to optimize the maneuver shall be investigated in the limited time.
The important factors for the criticality assessment are the conjunction geometry and the collision probability. In addition to the geometry and probability analysis at the latest and in the historical prediction, estimation of the future prediction is also necessary to detect the high risk conjunction. Even after a detection of the high probability event, the conjunction shall be carefully analyzed due to the large variety of the orbit prediction accuracy depending on the encountering objects. In case of very large orbit uncertainties, the probability density is spread widely, therefore the cumulative probability that the object comes near the target could become lower as a result. For the avoidance maneuver planning, it is useful for operators to be able to estimate the effect of the different maneuver strategies to the conjunction geometry as well as the collision probability. The maneuver size and epoch shall be then adjusted depending on the desired safety criteria, the time constraints, and the orbit control requirements.
In the paper, algorithms for the conjunction risk assessment and the avoidance maneuver planning are described, followed by a presentation of their application in the automated collision avoidance process in the user-friendly way to facilitate the handling of the critical conjunctions. Finally, the lessons learned through the operation are presented.

Loitering / Orbiting (II) / 8

New tool for finding periodic Halo orbits: the solver of a Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation)

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Halo orbits and other periodic orbits in the restricted circular 3 body problem has always been very well explained in the literature since their discovery by Farquhar in the 60 of last century. However for finding the numerical values of such orbits, the availability of the tools dedicated for such tasks is not obvious. Due to the fact that the differential equations are quite simple, those days most of the time tools used are based on some computer listings written within an US based mathematical framework, which is clearly dedicated for people highly involved in computer and computer language rather than for general purpose Engineers. Hence the approach chosen in the paper is to rely on a tool largely used by Engineers (and not computer guys) for taking advantage of the capabilities of solving dynamic problems: the tool used is a European tool which is a object oriented solver of differential equations which is the cornerstone of the Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation) largely used by engineers. The paper presents the mathematical problem in simple words and the method used to solve it. The major advantages of the approach proposed and used successfully is to benefit of a real simulation framework based on models and on experiments where there are no mixing between the inputs\outputs needs and the real problem being to be solved. Hence the full model can be clearly and explicitly described while the results coming from the experiments can be extensively assessed and analysed with simple monitor outputs.

Satellite Constellations and Formations / 163

ASTOS 8.1 - Mission Performance Analysis, System Concept Analysis and Other New Features

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The ASTOS software was originally designed as a trajectory optimization tool for launchers. This was more than 25 years ago. Since then the software had been continuously developed towards a multi-purpose analysis, simulation and optimization software for nearly all kind of space scenarios. This paper shall present the latest extensions of ASTOS incorporated into release 8.1. The paper will first give an overview on ASTOS and its workflow. Then it will present the new capabilities. Four new major developments have been made for the release 8.1 of ASTOS focussing on new markets and increased usability: 1) An integrated set of mission performance analysis features that cover almost all aspects required to fill a mission analysis report according to DOPS-GS-RM-1002-OPS-OSA. 2) Models and techniques for system concept analysis considering the power system, thermal control system as well as data management and communication systems. 3) A built-in tool to import CAD models, to easily texturize them and to use them for the built-in animation tool. 4) New wizards that ease the set-up of an ASTOS scenario and that help migrating a scenario from one to another application. The mission analysis capabilities of ASTOS cover launch, LEOP, operational phase and post-mission disposal, whereas mission analysis aspects like operational life-time prediction, eclipse analysis, collision avoidance, deorbit and graveyard manoeuvres, station keeping, etc. are considered. The results are automatically written into the mission analysis report template. For the system concept analysis dedicated models for solar arrays, batteries, PCDUs, data storages have been implemented. On the other hand all existing models, e.g. for actuators or sensors have been extended by thermal, data and power system aspects. These individual components can be linked into a system using a node model. In case of the thermal system each node can be linked to a surface element or another node, whereas each connection is characterized by its thermal resistance. For a realistic animation of a scenario it is essential to have a detailed geometry model of the vehicle. These models are typically available as CAD files. Those models lack texture and relevant surface material properties. These features are typically added using expert tools like Maya, Cinema4D or Blender. In order to reduce this effort ASTOS 8.1 provides an easy-to-use texturing tool that comprises a database with typical spacecraft surface materials. Due to the flexibility of ASTOS, it requires several steps to create a scenario in the classical way. This effort is significantly reduced by the new wizards. Based a question tree these wizards either create a new scenario or reduce the effort to migrate a scenario, e.g. from an optimization towards a guidance reference scenario. The paper will detail on each of the above mentioned new functionalities, showing user input and results taken from typical example scenarios.

Environment Modelling / 120

Evaluation of satellite aerodynamic and radiation pressure

acceleration models using accelerometer data

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Models of non-gravitational accelerations, of which satellite aerodynamics and radiation pressure are the most important examples, are critical for many orbit determination and prediction applications. Such models typically consist of three parts, each of which can be implemented at various levels of sophistication, depending on the required accuracy of the application. The first part consists of a model of the environment, such as the density of the atmospheric particles, or the direction and magnitude of the photon flux coming from the Sun and Earth. The second part is a model of the geometry and material properties of the satellite's outer surfaces, while the third part is a representation of the interaction between the particles and the surfaces. In its most simple form, these last two parts combined can be expressed in terms of a constant satellite ballistic coefficient.
Traditionally, the implementation of non-gravitational models in astrodynamics tools is based on a semi-empirical approach, and their assessment is based on an evaluation of tracking data residuals. The accelerometers on the CHAMP, GRACE, GOCE and Swarm satellites, however, measure the sum of the non-gravitational accelerations directly. The combination of these observations with non-gravitational acceleration models has led to the availability of thermospheric data sets with many scientific applications in the field of aeronomy.
In this paper, the experience obtained with the processing of accelerometer data and the use of non-gravitational force models for aeronomy applications is demonstrated, and applied in order to provide useful pointers for the implementation of such models in orbit determination and prediction tools at various levels of complexity and accuracy.

Re-entry and Aero-assisted Maneouvers / 121

Real-Time Atmospheric Entry Trajectory Generation Using Parametric Sensitivities

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The real-time generation of optimal trajectories and controls for nonlinear systems is a technology of interest to many applications. But the online solution of an optimal control problem (OCP) is often not computationally feasible on embedded systems. We present a method to generate a near-optimal control sequence and the corresponding state trajectory based on the parametric sensitivity analysis (PSA) of nonlinear programs (NLPs) which does not require performing the classical gradient based NLP process online and hence reduces the computational load. The OCP is transcribed into a parametric NLP which is solved offline for a nominal set of parameters. Additionally the parametric sensitivities of the optimal solution with respect to different types of perturbations are computed at discrete points along the nominal trajectory. The sensitivities are used online in a Taylor expansion of the nominal solution and an iterative feasibility and optimality restoration to compute a new near-optimal control sequence and trajectory from the disturbed state to the terminal set without resolving a disturbed instance of the original NLP. This process is repeated successively in the neighborhood of the nominal trajectory. The proposed method is demonstrated for the guided, hypersonic entry of a small capsule into the Martian atmosphere. The PSA algorithm is used as feed forward command and trajectory generation to provide the input for a drag-energy tracking controller.

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Coupled dynamics of large space structures in Lagrangian points

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Nowadays interest on large structures, ISS like, to serve for a long time as orbiting outposts place in strategic, possibly long-term stable locations, is increasing. They can serve as a support for far target robotic\manned missions, for planetary tele-operated robotic surface activities, as scientific labs for sample return missions in preserved environment avoiding contamination, for astronauts training, for refueling and maintenance of deep space vessels. Whatever the exploitation is such large structure would undergo numerous docking\undocking activities with a time dependent matrix of inertia; it should require a large lifetime along with orbital stability and, being the structure extended, a strongly coupled attitude\orbital dynamics is expected. Lagrangian points are an evident appealing location for such an infrastructure, offering stable trajectories together with well suited relative positioning with respect to the Sun and the other planets to be considered in the three body system [1].
The investigation of Large Structures coupled dynamics, whenever located in Lagrangian points proximity, is the topic of the paper. The configuration design\6 DOF dynamics coupling is deeply investigated to, eventually, drive the infrastructure system and operational design. Because of the wideness of possible practical applications in the incoming decade, the Earth Moon Lagrangian points system is here considered.
The paper firstly shows the natural periodic orbit-attitude solutions, introducing maps to visually identify the regions where those solutions exist, under the CR3BP approach. The maps are parametrized over the infrastructure inertia properties, and solutions are classified with respect to the number of attitude rotations per orbit. This taxonomy supports preliminary mission design and operations analysis, in verifying the impact of variations in inertia properties (e. g. after docking/undocking of a module) on the attitude and orbital motion. Practical applications of such solutions are discussed, with attention to future missions (e.g. ARM, HERACLES) which could benefit from a full exploitation of the coupled orbit-attitude dynamics. Distant retrograde orbits (DRO) are investigated with greater detail, since their stability properties are appealing for numerous applications. Extension to other orbit classes is then briefly discussed, underlining differences and similarities with the presented results and suggesting new fields of investigation.
The aforementioned discussion introduces then to the rigid body finite extension enhancement in the model, to assess its effect on the orbital motion and analyze modifications in the stability regions. Solar radiation pressure (SRP) disturbance is part of the model enhancing as well, assessing its effects on stability regions as a disturbing action on the whole coupled 6DOF dynamics. Design parameters and drivers are obtained from SRP analysis, with some conclusions on critical aspects and possible requirements for a large structure in the Earth-Moon system.
The need of an active attitude control is preliminarily discussed, with attention devoted to requirements needed to guide the control strategy and system design.
Effects of flexibility in the large infrastructure are then introduced in the model; energy principles and linear modal analysis are the mathematical approaches exploited to assess whether orbital and attitude motions could excite/be excited by small vibrations of flexible structures.
Preliminary considerations are deduced from the classical flexible dumbbell model, and are extended to lumped mass models of given complexity.
[1] B. Hufenbach, K. C. Laurini, N. Satoh, C. Lange, R. Martinez, J. Hill, M. Landgraf, A. Bergamasco, International missions to lunar vicinity and surface - near-term mission scenario of the global space exploration roadmap, 66th International Astronautical Congress, 2015.

Multidisciplinary Design Optimization / 123

An Object-Oriented multidisciplinary simulation framework for space dynamics and space tether simulation

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Tethers have been used in space for a number of different purposes: generation of artificial gravity, formation flying, propulsion, etc. Lately, space situational awareness has fostered the study of tethers in the context of space debris mitigation and removal, as is the case of the promising capabilities of electrodynamic tethers to de-orbit a satellite efficiently. In other approaches, tethers are considered as parts of more complex devices such as harpoons or hooks.
The dynamics of tethered spacecraft involve nonlinear effects with coupling between orbital motion and longitudinal and lateral modes, and show a extremely rich behavior including periodic and unstable motions. Moreover, tether dynamics can be coupled with other spacecraft systems such as AOCS and the power system.
Although different approaches have been proposed to address the simulation of space tethers, there is still a need to simulate the dynamic behavior with the required accuracy and an acceptable time consumption. There is also a need for modeling flexibility for simulating different complex multi-tethered configurations or the coupling with other spacecraft systems.
This work is intended to meet this need by using a new tether dynamical model and a non-casual and object-oriented modelling approach. To this end, the SDG is developing a set of simulation libraries for the multidisciplinary simulation tool EcosimPro. Although the focus of the libraries is the simulation of tether space systems, common space dynamics functionalities such as orbit propagation, satellite attitude dynamics, formation flying, AOCS simulation, etc. are also included in the libraries.
The tether dynamical model, which was presented in previous works, is briefly described. It is based on the discretization of the tether in a number of elastic rods, and allows simulating multi-tethered satellite systems with central and ending masses modeled as point masses or rigid bodies.
After the description of the structure of the libraries, the orbit propagators, and the validation process, several simulation cases are presented to show the capabilities of the simulator for trajectory propagation, AOCS simulation, coupled simulation of satellite motion with other systems, and multi-tethered satellite simulation. The work concludes by presenting a complete de-orbiting mission of an electrodynamic tethered system.

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Tube Dynamics and Low Energy Trajectory from the Earth to the Moon in the Coupled Three-Body System with Perturbations

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We develop a low energy trajectory from the Earth to the Moon by extensively using the framework of tube dynamics. We assume that a spacecraft is under the influence of gravity of the Sun, the Earth as well as the Moon and also that the spacecraft and the planets move in the same plane. In this situation, we model the Sun-Earth-Moon-Spacecraft (S/C) 4 body system as a coupled PRC3B system with perturbations, where the Sun-Earth-S/C 3 body system with the Moon perturbation and the Earth-Moon-S/C 3 body system with the Sun perturbation may be coupled at an appropriate patch point. First, we consider various boundary conditions such that the spacecraft departs from the low Earth orbit (LEO) and arrives at the low lunar orbit (LLO). Then we want to find a low energy trajectory connected at a patch point, so that the trajectories from the LEO to the patch point and from the patch point to the LLO can be connected with

less maneuver. To do this, we compute the Finite Time Lyapunov Exponent to detect stable and unstable invariant manifolds called "tubes" at a section and we construct the family of departure orbits in the Sun-Earth-spacecraft system with the Moon perturbation, which may be outside of the unstable manifolds associated with the Lyapunov orbit around L2. The family of arrival orbits is also obtained to be inside of the stable manifolds in the Earth-Moon-spacecraft system with the Sun perturbation. With a trajectory correction maneuver, then we obtain a low energy trajectory at the patch point. Finally, we will show that the proposed trajectory may be more efficient in the sense that the total maneuver is to be less than other trajectories such as the Hohmann trajectory.

Loitering / Orbiting (II) / 125

Fast Low Earth Orbit Acquisition Plan Optimiser

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Earth observation data is key for a more efficient use of land and natural resources, better land and sea monitoring, more informed political decisions, and better understanding of the weather, climate, and land changes.
Many Earth observation satellites are operated in low Earth orbits which provide a good trade-off between revisit time and spatial resolution. Repeat ground track orbits are of particular interest as they allow the acquisition of the same scene at fixed time intervals. Additionally, either for calibration or nominal operation proposes, many satellites are required to overpass an exact location on Earth.
For missions comprising spacecraft constellations, accurate orbit phasing is also needed. Hence, the satellite manoeuvres should be carefully planned to control the ground track drift so that the desired longitude at ascending node crossing are achieved.
This paper describes a tool and the associated mathematical framework that automatically computes an orbit acquisition plan that minimize the duration of the orbit acquisition phase or the required Delta-V given the spacecraft characteristics and mission constraints. The automatic algorithms are built upon on a perturbation analysis of the nominal orbit and provide the necessary information to perform preliminary analysis of orbit acquisition phases of Earth observation satellites.
Two kinds of simplified orbit acquisition plans can be computed: i) continuous semi-major axis change and ii) impulsive semi-major axis change. In the former, constant Delta-V per time interval is applied to the spacecraft, which is suitable when several small impulsive manoeuvres can be approximated by a continuous manoeuvre and also when there is no detailed information about the manoeuvring capacities and constraints. The latter is based on impulsive semi-major axis changes, which allows a more detailed plan where practical consideration such as the existence of calibration and touch-up manoeuvres are taken into account.
The tool also allows the analysis of several semi-major axis launch dispersions and launch dates providing for each case the resulting acquisition duration, required Delta-V, and mean local solar time drift. In particular, to insert another spacecraft within a constellation, a trade-off needs to be made regarding the requested orbital elements offset at orbit injection (in terms of semi-major axis, inclination or mean local solar time). Taking into account the spacecraft manoeuvre capabilities, the operational constraints of LEOP, the constraints preparing and implementing manoeuvres, launch date constraints and the agreed (or expected) launch dispersions, the tool identifies the consequences of all the possible scenarios (all cases are analysed together), which can then be used to define the orbital offset to be requested.
Some examples are given that illustrate the potential applications of this tool.

Open Source (II) / 126

A Comparative Study of Programming Languages for Next-Generation Astrodynamics Systems

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Due to the computationally intensive nature of astrodynamics tasks, astrodynamicists have relied on compiled programming languages such as Fortran for the development of astrodynamics software. Interpreted languages such as Python on the other hand offer higher flexibility and development speed thereby increasing the productivity of the programmer. While interpreted languages are generally slower than compiled languages recent developments such as JIT (just in time) compilers or transpilers have been able to close this speed gap significantly. Another important factor for the usefulness of a programming language is its wider ecosystem which consists of the available open-source packages and development tools such as integrated development environments or debuggers.
The aim of this study is to identify the most promising programming language for developing next-generation astrodynamics systems and tools. This target language shall offer an acceptable compromise between numerical performance and programmer productivity and possess a mature and sustainable ecosystem.
The study compares three compiled languages and three interpreted languages which were selected based on their popularity within the scientific programming community and technical merit. The three compiled candidate languages are Fortran2008, C++14, and Java 8. Python 3.5, Matlab 2015b, and Julia 0.4 were selected as the interpreted candidate languages. All six languages are assessed and compared to each other based on their features, ease-of-use, and ecosystem. Additionally idiomatic solutions to classical astrodynamics problems are developed in all candidate languages and compared based on their performance and simplicity.

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First results of IOTA (In-Orbit Tumbling Analysis)

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Estimating and predicting the tumbling motion of orbital debris is not an easy task. While in orbit an object experiences perturbations due to external forces in the near-Earth environment. This changes the object's orbit and attitude over time. There are observational techniques that are being used to determine the spin rate of orbital debris such as light curve observation, satellite laser ranging and radar measurements. It is difficult to determine the true attitude motion without combining the results from the observations and comparing them with a model.
IOTA is a prototype software developed under ESA's "Debris Attitude Motion Measurements and Modelling" project by Hyperschall Technologie Göttingen (HTG) to address this issue. The software will provide short-, medium-, and long-term propagation of orbit and attitude motion (six degrees-of-freedom), taking into account all the relevant forces and torques acting on satellites and space debris. External influences included are gravitational forces from the Sun-Earth-Moon system, aerodynamic drag, solar radiation pressure, eddy current damping and momentum transfer

from micrometeroid impacts as well as internal influences such as reaction wheel behaviour, tank sloshing, magnetic torquer activity and thurster firing.
Post-processing of the simulated result will enable generation of synthetic measurements of observation. Combining and comparing the observational with the simulated results increases the accuracy of the attitude motion determination and will lead to better understanding of the attitude evolution.
The software is still under development. In this paper the software implementation of the environmental influences and propagation of the state vector will be discussed and preliminary results with of the attitude motion of ENVISAT have been simulated to provide an example of the capabilities of IOTA.

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Dynamics in the center manifold around equilibrium points in Periodically Perturbed Three-Body Problems

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Please find attached the corresponding abstract.
Best regards,
Bastien Le Bihan.

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A new Mars EDL mission design and simulation tool - MEDLMDST

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Mars entry, descent and landing (EDL) begins when the vehicle reaches Mars atmospheric interface (about 125km altitude) and ends when the supersonic parachute is deployed. EDL phase is considered the critically important sub-phase, and largely determines the success or not of entire Mars EDL. A new Mars EDL mission design and simulation tool, named as MEDLMDST, has been developed at the Space New Technology Laboratory, Nanjing University of Aeronautics and Astronautics. In this paper, the components and architecture of MEDLMDST are firstly introduced. Then, the progress of MEDLMDST and some new results in EDL trajectory optimization, navigation and guidance are reported in detailed.

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Mission Staging Planner

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The Concurrent Design Engineering is that engineering process through which short iterations in preliminary design phases allowed to significantly improve the consistency of data, reducing costs and non-qualities in the results. This concept is applicable on Space Systems engineering due to the design complexity and the long development times and higher costs, typically requested for the development of new missions. Future challenging science and space exploration missions are more demanding and specialists use many different software to make their calculations.
Looking forward for a major leadership of Thales Alenia Space Italy in this direction, based on the experience acquired through several ESA Phase-A mission studies programs, a staging analysis tool for the support of the Mission Analysis and the S/C propulsion and architecture design has been developed. Knowing the mission profiles selected, introducing the S/C mass and propulsion reference parameters, propellant and dry mass optimisation of multi and/or single stage S/C can be performed on the basis of the forecast staging events selected, directly drive the S/C design and size processes, targeting since the beginning of the studies its proper configurations.

Low Thrust (I) / 59

A Sequential Method to Compute Multiobjective Optimal Low-Thrust Earth Orbit Transfers

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A sequential algorithm for optimizing low-thrust Earth orbit transfers in terms of the propellant consumed is proposed. The dynamical model includes the effect of the Earth shadow and the J2 effect of the gravitational potential. The algorithm is based on two steps of growing complexity. In the first step, a near-optimal solution is obtained using simplified dynamics. In the second step, a hybrid approach embedded in a direct collocation scheme is used to consider the optimal coast arcs out of the Earth shadow. This novel approach is a continuation of a previous work in which the minimum time problem was solved. The ultimate goal of the work is to develop a robust and flexible tool that addresses the multi-objective design of low thrust transfers in Earth orbit. Hence, the user will not only obtain just point solutions, but will be able to explore the set of Pareto optimal trajectories for these two objectives.

Debris, Safety and Awareness (III) / 58

Computer Graphics for Space Debris

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The number of resident space objects re-entering the atmosphere is expected to rise with increased space activity over recent years and future projections. Predicting the probable survival and impact location of the medium to large sized re-entering objects becomes important as they can cause on-ground casualties. We present development and application of a new tool for quick estimation of aerodynamics and aerothermodynamics properties of the re-entering objects. The novel method uses primitive geometries to develop a complex object and voxelization (voxels are the 3D equivalent of pixels in 2D images) for computing the shading/visibility factors to quickly estimate the aerodynamic and aerothermodynamics properties of the re-entering object. The tool can be used as a module of object oriented codes to support preliminary End of Life analyses.

Low Thrust (I) / 55

Low Thrust Trajectory Optimization for Autonomous Asteroid Rendezvous Missions

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The demand for deep space missions and the desire to investigate small space objects like asteroids and comets is increasing constantly. Such exploration missions open up the possibility to gain further scientific knowledge about the origin of our solar system as well as to find and eventually mine rare earth elements that are narrow on Earth or even to discover novel resources. To realize such missions, trajectory planning and optimization are of utmost importance. When flying further into outer space, the spatial distances become enormously huge while fuel is very limited and signal runtimes are increasing rapidly. These conditions impose an unmanned and autonomously working spacecraft. The mathematical field of optimization and optimal control provides the foundation for autonomous decisions and facilitates more safety and minimal resource consume. Solutions may additionally be transferred to other, earth-bound applications like e.g. deep sea navigation and autonomous driving with minor additional expenses. The aspects investigated in the present paper focus on the specific challenges of guidance and control regarding the cruise and approach phase of a spacecraft starting in a parking orbit around the Sun and reaching for an asteroid in the main belt. The underlying optimal control problems are solved using so called direct methods also known as transcription techniques. Those transform an infinite-dimensional optimal control problem (OCP) into a finite-dimensional non-linear optimization problem (NLP) via discretization methods. The resulting high dimensional non-linear optimization problems can be solved efficiently by special methods like sequential quadratic programming (SQP) or interior point methods (IP). For solving the problems introduced in this paper the NLP solver WORHP, which stands for 'We Optimize Really Huge Problems', is used, a software routine combining SQP at an outer level and IP to solve underlying quadratic sub problems. Within this paper the transcription is performed using the robust method of full discretization. The trajectory optimization and optimal control problems are modeled and solved using low thrust electric propulsion on the one hand and chemical propulsion on the other hand for comparison. The movement of the spacecraft is described through ordinary differential equations (ODE) considering the gravitational influences of the Sun and the planets Mars, Jupiter and Saturn as well as the different thrust commands. Competitive mission aims like short flight times and low energy consumption can be provided with a weighting factor within the optimization process. The varying challenges of the two propulsion types are analyzed and comparative solutions and results introduced. Several mission trajectories are compared, aiming at different destination asteroids and optimizing with different weighting factors for energy cost and flight time duration in order to investigate the different possibilities of an asteroid rendezvous mission. The results show the huge gain of trajectory optimization as input for on-board autonomous decision making during deep space missions as well as the great increase in possibilities for flight maneuvers by providing solutions for changing and contradictory mission objectives. Furthermore, trajectory optimization can be used to analyze the potentials of different propulsion systems beforehand.

Optimization and Dynamics (I) / 54

WORHP Multi-Core Interface, Parallelisation Approaches for an NLP Solver

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The goal of this talk is to present current research activities aiming at improved efficiency and stability within the ESA NLP-Solver WORHP. It is designed to solve high dimensional sparse non-linear optimisation problems.

The underlying SQP method is inherently sequential, therefore parallelism cannot be exploited straightforwardly.
In order to obtain the best solver performance the parameter configuration should be adapted accordingly for every problem itself. An approach running several solver instances using different parameter sets in parallel has been developed and proven highly beneficial on a given set of problems. The First-Across-The-Line approach stops all instances when the first local optimum has been found, thus improving the solver's speed and stability as well. Furthermore, the approach allows the user to experiment with specialised algorithms within the optimisation as the threads using basic parameter settings serve as safeguards guaranteeing the solver to converge as usual.
In order to improve the solver's efficiency for one special problem, the new operational mode can be used to automatically attune the solver's parameters accordingly. Again the solver is started with several instances at once, but this time the Best-Of-All mode is used in order to obtain the best local optimum and the corresponding parameter settings. Additionally, the mode enables users to perform parameter sweeps to further improve the solver's configuration.
The results presented show the improvement of the solver's performance on the state-of-the-art CUTEst test set that has been solved faster and with more optimal solutions found compared to the traditional single-core approach. The parameter attunement is applied to specific single problems from the test set as well.

Ascent (I) / 57

Modeling and Performance Evaluation of Multistage Launch Vehicles through Firework Algorithm

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Multistage launch vehicles of reduced size, such as "Super Strypi" or "Sword", are currently investigated for the purpose of providing launch opportunities for microsatellites. Currently, microsatellites can be launched according to the time and orbital requirements of a main payload. The limited costs of microsatellites and their capability to be produced and ready for use in short time make them particularly suitable to face an emergency (responsive space), therefore small launch vehicles dedicated to microsatellites would be very useful. On the other hand, in order to reduce the launcher size without increasing too much the launch cost per kg of payload it is necessary to simplify the launch system as much as possible, including the guidance algorithms. A simple open-loop guidance strategy is proposed in this research and applied to the Scout rocket, a micro-launcher used in the past. Aerodynamics and propulsion are modeled with high fidelity through interpolation of available data. In order to simplify the open loop guidance law employed for the first three stages, the aerodynamic angle of attack is assumed constant for each stage. Unlike the original Scout, the terminal optimal ascent path is determined for the upper stage, using a firework algorithm in conjunction with the Euler-Lagrange equations and the Pontryagin minimum principle. Firework algorithms represent a recently-introduced heuristic technique inspired by the firework explosions in the night sky. The concept that underlies this method is relatively simple: a firework explodes in the search space of the unknown parameters, with amplitude and number of sparks determined dynamically. The succeeding iteration preserves the best sparks. With regard to the problem at hand, the unknown parameters are (i) the aerodynamic angles of attack of the first three stages, (ii) the coast time interval, (iii) the initial values of the adjoint variables conjugate to the upper stage dynamics, and (iv) the thrust duration of the upper stage. The numerical results unequivocally prove that the methodology at hand is rather robust, effective, and accurate, and definitely allows evaluating the performance attainable from multistage launch vehicles with accurate aerodynamic and propulsive modeling.

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Use of Ground-Based Space Electro-optical Tracking to support Low Thrust Satellite Orbit Transfer to Geosyncrhonous Orbit

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The INMARSAT-5F2 satellite was launched on February 1, 2015, and on February 18, 2015 initiated its XIPS Ascent Mission (XAM) maneuver phase using Xenon-Ion Propulsion System (XIPS) thruster technology. A set of optical telescopes were enlisted to observe the satellite during the XAM phase for the purpose of assessing the value of those measurements in supporting and meeting orbit determination (OD) requirements with reduced range tracking – reducing the requirement for range measurements will reduce the cost of the mission support. The goal of the analysis was to compare the OD ranging-only solutions – the operational standard – with solutions derived from reduced range tracking plus the addition of Electro-Optical (EO) data. The low thrust maneuver profile was incorporated into the Orbit Determination Tool Kit (ODTK) orbit processing software and corrections to the thrust estimated, along with the orbit parameters. The results indicate that the combined ranging and EO OD and thruster estimation performance is comparable, if not better than, ranging. The assessment was based on the comparison of estimation covariances, state differences between the two tracking cases, cross-over differences between adjacent independent arcs, and a computation of the Mahalanobis Distance (MD) for those differences (the MD metric accounts for state uncertainties in the state differences). Though the radial and in-track performance for the combined data case was marginally worse compared to the range-only solutions, there was significant improvement in the cross-track component for the combined case resulting in an overall reduction in error over range-only. The results demonstrate the value of combining EO and range for tracking low thrust maneuvers, with the promise of reducing costs associated with range-only tracking.

Optimization and Dynamics (I) / 50

Non-Keplerian Trajectory Planning via Heuristic-Guided Objective Reachability Analysis

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In many space exploration scenarios of great interest, such as close-range study of asteroids and comets, spacecraft motion cannot be effectively approximated using Kepler's laws. Furthermore, special dynamical structures such as periodic orbits are not inherently associated with specific science requirements and serve only as a limited framework for facilitating operations. To broaden the mission design domain for pursuing abstract objectives in non-Keplerian systems, we instead formulate a reachability analysis tool that maps a domain of available single-impulse maneuvers onto a set of high-level outcomes. As this process can only be conducted with numerical sampling, heuristics are used to guide iterative refinement of map features or the search for a performance metric's global maximum. The reachability data product can be visualized to aid preliminary mission design or efficiently computed onboard the spacecraft to enable opportunistic and robust online planning. Both modes of use will be demonstrated for planning trajectories for close-range imaging of potential lander deployment sites on the highly irregularly shaped comet 67/P.

Rendezvous & Docking (I) / 53

Fast Pose Tracking of Spacecraft from LiDAR Point Cloud Data

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Reliable relative navigation between spacecraft is a key technology in a variety of mission scenarios. This is especially true for future missions like On-Orbit Servicing (OOS) or Active Debris Removal (ADR), wherein some servicing spacecraft has to approach a possibly uncooperative target satellite/object. To execute such manoeuvres safely and effectively, accurate estimates of the target's relative pose, i.e. its position and attitude, are essential. Since the target must be considered uncooperative, classical sensors like GPS or cooperative aids like actively illuminated navigation markers or passive retroreflectors are not available. However, optical sensors are well suited for providing the necessary data. 3D scanners based on the Light Detection and Ranging (LiDAR) principle, create a three dimensional point cloud of the target by measuring the time-of-flight of laser pulses backscattered from the target's surface. Given such a point cloud, a pose initialization algorithm may calculate a (rough) estimate of the target's position and attitude without any a priori information about its pose. Beginning with this initial estimate, a tracking algorithm can then determine accurate solutions of the target pose continuously, at each time step correcting and updating the solution of the previous step.
In this paper, an advanced pose tracking algorithm is presented which copes with the challenging problem of reconciling two major contradicting requirements in the context of spacecraft relative navigation: Even in the face of noisy and sparse point clouds, pose estimates have to be sufficiently accurate for the spacecraft's Guidance Navigation and Control (GNC) system to ensure a safe and stable approach manoeuvre. At the same time the algorithm has to be capable of providing a reasonably high rate of pose estimates despite of being constrained by strongly limited computational power available in space. This becomes especially clear when considering that uncooperative targets may be tumbling at relatively high angular velocities.
The high-performance algorithm presented in this paper is based on the Iterative Closest Point (ICP) principle, which successively computes the change in current target pose with respect to the solution of the previous time step by solving a local optimization problem to minimize the error between the reference model point cloud and the point cloud scanned by the LiDAR sensor. A core step in solving the optimization problem is the search for the nearest neighbour model point with respect to a given scan point. This is by far the most costly part of the algorithm and highly optimized data structures are used to accelerate the search. After this step, outliers, i.e. invalid scan points must be identified and discarded to not falsify and corrupt the solution. From the resulting valid point-to-point correspondences, the transformation can be calculated that minimizes the error between model and scanned points. The whole process is repeated iteratively, until the solution converges or the residual error is sufficiently small. The resulting change in target pose with respect to the solution of the previous time step is used to update the current absolute target pose relative to the sensor frame.
The algorithm is verified with LiDAR point cloud data obtained during a test campaign with the European Proximity Operations Simulator (EPOS), located at DLR in Oberpfaffenhofen (near Munich), Germany. The tests involved a true-to-scale satellite mockup with representative surface materials, illumination conditions resembling those in Low-Earth-Orbit (LEO), a breadboard version of a LiDAR prototype aiming for space applications and complex relative trajectories with realistically simulated tumbling target motion.
This is followed by a discussion of algorithm test results. Accuracy and performance are presented and their dependence on algorithm parameters is evaluated. An optimal combination of parameter values is determined for this representative test case.
Finally, points of possible further improvement and development are outlined.

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Impact of Solar Spin on Planetary Orbits

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Abstract. This paper examines the possible effect of solar spin in the planetary orbit. We show in our earlier paper that if we incorporate the contribution of spin of the central gravitating body in orbital calculations, a residual slight perturbation on the standard constant areal velocity

should exist. In particular, the second law of planetary motion requires a revision. However, it turns out that the classical result of Kepler is recoverable from our result as a special case. To be able to appreciate the need for the revision suggested by the new perturbation considered here, this paper looks into the genesis of orbital theory. We herein propose to reduce spin theory to non-relativistic regime. In fact, we consider restricted three-body problem. Confining, for the moment, again to the specific context of solar system, our initial calculations show that the transverse component the transverse component of the force field is nonzero, in contrast to the GN-physics (Galilei-newtonian physics) wherein such a component vanishes. In particular, the transverse component of the central force field does vanish if we neglect the spin of the gravitating star. This situation is radically different from that of GN-theory (where linearisation often does result). However, if we set the spin equal to zero, we retrieve the orbit equation of GN-physics. As regards solution, we apply numerical schemes to determine solution of nonlinear orbit equation for Earth. Our results exhibit that the new light on issue in relativistic celestial mechanics and models of planetary motion.

Open Source (I) / 90

STAVOR: a mobile application for the space domain

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Modern scientists and engineers use complex software tools to ease and accelerate their work.
In the space domain, the calculi are too heavy to be computed manually, especially optimization algorithms. Software is therefore mandatory for any realistic task. One issue common to most high level technical software is that validating an operational system takes a long time, and during this delay, the available technologies change very quickly. In the aerospace domain we usually found that the available validated applications are not up to date to the latest technologies and therefore, they don't maximize their capabilities, they become complex and they are not user friendly.
We think that standardized validated and operational computing engines should be combined with some, less validated software but using last technologies in order to complete the market needs and get the best of both worlds. These new tools should be very simple, intuitive and interactive to maximize the help to the user; and should be only used in cases where the rapidity prevails to the accuracy of the results: like in education, preliminary analysis or similar cases.
The new technologies that are interesting for this kind of software are: latest 3D visualization technologies, interactive data plotting, touch screens with the corresponding control gestures, mobile platforms, cross-platform solutions…
At CS Systèmes d'Information, we decided to create one of these tools as an example, using as much as possible these new technologies. The company is the main developer of an open-source space dynamics library called Orekit. This library has been already used operationally by many actors and proved itself to be very powerful and validated; nevertheless, it demands a minimum of programming skills to be used to its full potential. People in the aerospace domain do not always have this knowledge, nor should they have it. By creating this new tool, the main functionalities of the library should be bypassed to an interactive UI, and therefore, erasing the need of informatics knowledge to use it. With this move we reduce its functionality but we increase largely the target users.
The application itself is a mobile platform (Android), using 3D representations with modern cross-platform languages (WebGL) and controlled by an interactive touch-ready UI. It consists of a simple space mission simulator, and three different results visualization screens: two 3D modules for the attitude and orbit representations, and a 2D cartographic view to represent the information over the Earth surface. The main uses for this application are the education, due to simple, interactive and intuitive visualizations and its portability; but also more industry oriented applications like orbits comparison, preliminary mission analysis, information exchanges between coworkers…

This product, called STAVOR, has been conceived as a quick aid application to have in the smartphone or tablet, to coexist with the normally used desktop tools.

Ascent (II) / 164

Wave-Based Attitude Control of Launch vehicle with Structural Flexibility and Fuel Sloshing Dynamics

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ABSTRACT
The dynamics of a typical launch vehicle are non-linear, non-uniform and time variant. In addition, liquid fuel systems introduce sloshing dynamics which can be difficult or impossible to model, especially if required for real time control. It is a difficult problem, with consequences for the stability, trajectory tracking and efficiency of the launcher. Wave based control has been developed to deal with such difficulties in a natural way and has been shown to cope well with under actuated flexible space systems.
The key concept is that the motion of a flexible body can be separated into two notional components, one travelling from the actuator, for example a TVC engine, into the system, the other leaving the system through the actuator. The actuator, simultaneously launches mechanical waves into a system while it absorbs returning waves. When the launching and absorbing is finished, vibrations launcher have been damped and the desired reference motion is left behind.
A wave-based controller is designed for a launcher similar to the European launcher Vega, where the input data to build the launcher model has been taken from publicly available information. In numerical simulations the controller successfully suppresses the disturbance of the launcher body due to structural flexibility and fuel sloshing motion. A major advantage of the strategy is that no direct measurement of the sloshing motion is required. Only measurements of launcher attitude and actuator torque are needed. The work has been carried out as part of ESA Future Launchers Preparatory Programme (FLPP)
Keywords: Spacecraft Dynamics, Launchers, Attitude Control, Flexible Systems, Sloshing, Mechanical Waves
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Coffee break / Poster Session / Booth Exhibition / 179

The Horizon 2020 project ReDSHIFT: Revolutionary Design of Spacecraft through Holistic Integration of Future Technologies

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The ReDSHIFT (Revolutionary Design of Spacecraft through Holistic Integration of Future Technologies) project has been approved by the European Community in the framework of the H2020 Protec 2015 call, focused on passive means to reduce the impact of Space Debris by prevention, mitigation and protection.

ReDSHIFT will address barriers to compliance for spacecraft manufacturers and operators presented currently and in the future by requirements and technologies for de-orbiting and disposal of space objects. In ReDSHIFT these goals will be achieved through a holistic approach that considers, from the outset, opposing and challenging constraints for the safety of humans on ground, when these objects re-enter the atmosphere, designed for demise, and for their survivability in the harsh space environment while on orbit.
Ensuring robustness into the future, ReDSHIFT will take advantage of disruptive opportunities offered by 3D printing to develop highly innovative, low-cost spacecraft solutions, exploiting synergies with electric propulsion, atmospheric and solar radiation pressure drag. Inherent to these solutions will be structures to enhance the spacecraft protection, by fracture along intended breakup planes, and re-entry demise characteristics. These structures will be subjected to functional tests as well as specific hypervelocity impact tests and material demise wind tunnel tests to demonstrate the capabilities of the 3D printed structures. Modern celestial mechanics and astrodynamics tools will be exploited to find "de-orbiting highways", (i.e., fast trajectories to de-orbit) able to meet de-orbit and disposal needs, coupled with the above-mentioned technical solutions.
At the same time, novel and complex technical, economic and legal issues of adapting the technologies to different vehicles, and implementing them widely across low Earth orbit will be tackled through the development of a hierarchical, web-based tool aimed at a variety of space actors. This will provide a complete debris mitigation analysis of a mission, using existing debris evolution models and lessons learned from theoretical and experimental work. It will output safe, scalable and cost-effective satellite and mission designs in response to operational constraints.
Through its activities, ReDSHIFT will recommend new space debris mitigation guidelines taking into account novel spacecraft designs, materials, manufacturing and mission solutions. In the talk, the technical description of the project will be given, along with the first progresses made within the study.

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Navigation Tools at ESOC Mission Analysis Section

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Software able to study and simulate the guidance and navigation process is essential when dealing with complex interplanetary missions. The outputs required from such software are multiple, ranging from the evaluation of stochastic ΔV required to keep the dispersion errors with respect to the nominal trajectory small, to the identification of critical phases of the missions for the guidance and navigation process, to the assessment of the optimal ground station network selection and many others. In the ESOC Mission Analysis section the two most complete tools available in this context are IntNav (Interplanetary Navigation software tool), developed by GMV and LoTNav (Low Thrust Navigation software tool), developed by DEIMOS Space. Both tools are composed of several modules and they share a common structure; the core modules of both software are: measurements generation module, parameters estimation module and guidance module. The first module computes the measurements selected for the orbit determination, given the trajectory of the spacecraft and the measurement schedule (availability of observations and ground stations are always taken into account). The parameters estimation module is used to compute partial derivatives of the estimated states with respect to the selected parameters and also to conduct the covariance analysis for knowledge and dispersion. The guidance module finally is used to determine the stochastic manoeuvres to be executed in the guidance process. Additional modules are also present in both tools (or were added to the core software afterwards), to deal with other functions or analyses to be performed in the navigation context: for example modules for the trajectory computation and sectioning are available in both tools, a reconstruction module is available in IntNav to simulate the trajectory reconstruction of particularly interesting phases in the trajectory (e.g. flybys), and many others are available. A general overview of this structure will be presented, followed by examples obtained with both tools. JUICE moons tour navigation

results obtained with IntNav and Bepi-Colombo interplanetary navigation results obtained with LoTNav will be presented. The major assumptions will be explained and the encountered issues will be highlighted: flybys with high uncertainties and large navigation manoeuvres, parameters selection choices taken after the results evaluation, the non-trivial optimization process for the manoeuvres targeting and other aspects. Final results for both cases will be then detailed.

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Assessing Orbit Determination Requirement with Unscented Transformation: Case Study of a Lunar CubeSat Mission

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The orbit determination (OD) requirement for a lunar CubeSat mission is examined. The motivation for this study is the NASA CubeQuest Challenge, for which the winning 6U CubeSats in the competition are offered a launch on the Exploration Mission (EM) 1 mission as secondary payloads. All such CubeSats will be disposed into a high-energy trajectory that will fly by the moon. Unless high-impulse chemical propulsion system is allowed on the CubeSats, most designs will involve some form of a low-thrust propulsion system to achieve lunar orbit. In order to determine the OD strategy for such a mission, the OD accuracy requirement needs to be understood. One driver for the OD accuracy is its contribution to the delta-V budget and hence the spacecraft's ability to achieve the target lunar orbit. Typically, this type of analysis is done using Monte Carlo simulations, but the large number of cases required to achieve a statistically significant result is often prohibitive.
In this paper, we examine the use of unscented transforms1 to determine the impact of OD accuracy on the delta-V budget. We take a candidate low-thrust propulsion trajectory from EM-1 disposal to lunar orbit and a candidate ground station architecture using one-way Doppler measurements to determine the OD accuracy requirement. This method is not unlike the linear covariance analysis2, however, its use of sigma points extends its usefulness beyond the linear region, especially for the highly nonlinear problem of the low-thrust transfer to the moon.
Two open-source tools from NASA Goddard Space Flight Center are utilized to perform this analysis: the General Mission Analysis Tool (GMAT)3 is used for the low-thrust maneuver planning and the Orbit Determination Toolbox (ODTBX)4 is used for the orbit determination. GMAT and ODTBX are interfaced with a function that generates the sigma points of the unscented transformation from the orbit estimate covariance matrix obtained from the orbit determination. Another separate function combines the sigma points of the delta-V penalty. The efficacy of the unscented transformation method is demonstrated by comparing the results of this technique with the results from a small-scale Monte Carlo simulation.
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Re-entry and Aero-assisted Maneouvers / 114

Aerodynamic categorization of spacecraft in low Earth orbits

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Spacecraft re-entering the Earth atmosphere in an uncontrolled manner may get stabilised by restoring aerodynamic torques, if they have an appropriate shape and mass distribution. While the aerodynamic force (mainly drag) is usually a second-order effect compared to the gravitational acceleration by the Earth at altitudes above 150 km, sometimes the aerodynamic torques can already compete with the Earth gravitation gradient-induced torques at altitudes around 250 km and below. Therefore it is of interest to have an understanding of how to compute the aerodynamic torques in this altitude regime.
The usual approach to compute the aerodynamic coefficients at high altitudes is to construct a surface model of the spacecraft, where the surface is either modelled with plane face elements or discretized into small triangular or quadrangular "panels". The aerodynamic coefficients are then computed for each surface element and summed up with an Integral or Monte-Carlo method, utilizing that the flow around the spacecraft can be considered as free-molecular.
While the Monte-Carlo method can give quite accurate coefficients for a given configuration, it does not allow to parametrize the object of interest. This is different to analytical solutions, where the geometric dimensions and mass distribution appear as explicit parameters, and where the influence of configuration changes on the results are direct. On the other hand, the possibility to get analytical solutions is limited to convex geometric shapes. This can be extended to concave shapes by using some kind of shadowing algorithms, but in this case the additional effort needed to examine the shadowed areas can foil the advantages of the analytical approach compared to the numerical analysis.
In any case, the combination of different methods can give an added value. For basic geometries analytical solutions are known. Comparing these solutions with Monte-Carlo results can serve as a calibration method for the Monte-Carlo method's statistical uncertainties. For more complex geometries the Monte-Carlo method can give a measure of the effect of shadowing and multiple reflections, which cannot be considered exactly, or not at all, in analytical solutions or integral methods.
Due to the special form of the free-molecular gas-surface interaction and its momentum transfer some simplifications are possible especially for the typical high-speed conditions in orbit, which can be used to extend the validity of the analytical solutions or at least extend their approximate range of validity. This will be used in the paper for a simplified categorization of spacecraft without extensive multiple internal reflections w.r.t. to their aerodynamic characteristics.

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Exoatmospheric Guidance Strategy for Small Solid Propelled Launcher

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This paper describes the powered phase exoatmospheric guidance strategy used for the small solid propelled launcher. In the 3rd stage, the closed loop guidance method is used and Alternate Attitude Control Energy Management(AEM) is applied for the remaining energy consumption. For the 4th stage, pointing algorithm is used to get a circular orbit based on orbital parameters equations and the necessary conditions. At last, the robustness and high accuracy of the guidance strategy is validated by Monte Carlo simulation and analyzing the influence of different launcher terminal status parameters.

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Preliminary study on launcher reusability - An illustration of ONERA's knowledge and tools

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Up to the early 80s, access to space was mainly a sovereignty issue. From this period, it is also and even increasingly more a commercial issue. Injecting the maximal payload on the desired orbit with the maximal reliability was the problem given to the engineers. Unfortunately, the cost reduction was not there. This explains why the reusable launcher concept is on the agenda and particularly the re-use of its lower propulsive stage, as it is equipped with the most powerful or the most numerous engines. The number of open issues for such design is impressive: return or not to the launch site, recover of only the engine part of the stage or of the full stage, vertical landing using the launcher propulsion itself or horizontal landing using some aeronautical technologies, re-use or not of the existing propulsive stages . . .
From a design capability point of view, the combined use of an integrated MDO approach (the whole interaction between disciplines being modelled, and a joint geometry/trajectory optimization being performed) and a collaborative environment (allowing exchanges between experts and design space visualization and exploration) enables to explore quickly several design options. This allows to provide the decision maker enlighten clues for selecting promising launch vehicle architectures to comply with reusability issues.
From a decade, ONERA has matured MDO methodologies such as design problem formulation, optimization algorithms, uncertainty propagation, etc. ; and dedicated collaborative environment to deal with such design problems. This paper proposes to illustrate these ONERA abilities on a two-stage reusable launch vehicle design problem with toss back return. In this study, the first LOX-CH4 stage and the second LOX-LH2 (or LOX-CH4) stage are optimized in order to inject a 5 ton payload into GTO orbit. Both ascent and return trajectories are optimized along with the launch vehicle design variables. Several different architectures (use of boosters, reuse of Vinci engine, type of propellant of the second stage) have been optimized in order to select the best configuration for this mission.

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New orbital elements for accurate orbit propagation in the Solar System

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Close encounters with massive bodies, such as planets or Jupiter/Saturn's satellites, make the orbit of any asteroid or spacecraft chaotic. Moreover, in the case of subsequent encounters the Lyapunov time can become very short. Accurate propagation is required in the orbit determination of chaotic bodies, because it mitigates the exponential divergence of nearby orbits. For example, the impact monitoring of natural and artificial objects with the Earth, and the pianification of space missions with several fly-bys, have to be done with mathematical tools that are able to deal with chaos. One of these tools is a reliable and accurate orbit propagator.
We propose new methods to accurately compute elliptic and hyperbolic motion in the Solar System. Our approach roots in the regularization of the two-body equation, which is transformed into a set of linear differential equations with constant coefficients. This result is obtained by introducing a new independent variable (also called fictitious time) and new spatial coordinates in place of the position and velocity.
In the Burdet-Ferrándiz linearization the fictitious time is the true anomaly, and the new state variables are the inverse of the orbital radius, the radial direction and the angular momentum. In this way the motion is decomposed into the radial displacement and the rotation of the radial

unit vector. We show that a new linearization of the two-body equation can be obtained with a similar decomposition when either the eccentric or the hyperbolic anomaly is the independent variable. Then, by applying the variation of parameters we introduce six variables that can be used to describe the perturbed motion of the propagated object. The new quantities, together with the total energy and the physical time, constitute the state vector of the special perturbation methods proposed here (the method that works with negative total energy is described in ref. 1). We also investigate the geometrical and physical meaning of the six parameters: they are all related to an intermediate frame which shares with the local-vertical local-orizontal frame the direction of the angular momentum. This slowly moving frame recalls the ideal frame discovered by the Danish astronomer P. A. Hansen in 1857, which plays a key role in Deprit's (ref. 2) and Peláez's (ref. 3) sets of orbital elements.
The performance of the new formulation has been evaluated for geocentric motion and for trajectories with close encounters. We found a considerable advantage with respect to the traditional integration in Cartesian coordinates.

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Coffee break / Poster Session / Booth Exhibition / 110

The ESPaCE consortium as a European producer of spacecraft and natural moon ephemerides

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The consortium ESPaCE (European Satellite Partnership for Computing Ephemerides) is composed of seven European institutes: IMCCE (Institut de Mécanique Céleste et de Calcul des Ephémérides, Paris Obs.), ROB (Royal Observatory of Belgium), TUB (Technical University of Berlin), ERIC (European Research Infrastructure Consortium formerly known as JIVE : Joint Institute for VLBI in Europe), TUD (Delft University of Technology), French space agency (CNES) in France and German Aerospace Center (DLR) in Germany. The objective of the consortium, initiated under an FP7-European project is to provide new accurate ephemerides of natural satellites and spacecraft. For this goal astrometric data issued from ground-based observations as well as from space observations are being analyzed and reduced. On the other hand emerging technologies, specifically VLBI and interplanetary laser ranging, applied to the positioning of spacecraft are also studied. The ESPaCE project addresses also data related to gravity and shape modeling, control point network and rotational parameters of natural satellites. The accuracy improvement of these ephemerides makes them a powerful tool for the analysis of space missions, the preparation of future missions, or for the determination of planetary physical

parameters. Among relevant sub-products for space missions, we note the delivery of updated ephemerides of the Mars moons Phobos and Deimos derived from data by the Mars Express mission. In addition, the ESPaCE ephemerides of the Galilean moons are regularly updated in the context of the upcoming JUICE mission.

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The true nature of the equilibrium for geostationary objects, applications to the high area-to-mass ratio debris

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The long-term dynamics of the geostationary (GEO) region has been studied both numerically (Chao, 2005; Anselmo and Pardini, 2007) and analytically (Chao and Baker, 1983; Chao, 2006; Valk et al., 2008; Rosengren and Scheeres, 2013), and some of these results contributed to the IADC guidelines for disposal of objects in the GEO region.
In this work, we revisit the dynamics of this region through the application of canonical perturbation theory, and we apply our results to study the peculiar dynamical behavior of high area-to-mass ratio space debris. More specifically, previous works focused on the evolution of objects *around* a nominal solution called the forced equilibrium solution. Here, instead we focus on the nature of the equilibrium solution itself. Thanks to a higher order normal form, we demonstrate that this equilibrium is actually a lower dimensional object containing slow frequencies. This means that even placed at this pseudo-equilibrium, an object will exhibit periodic variations of its elements, which can be large. We give an analytical expression of these variations, valid for long time scales.
To this end, we considered the Hamiltonian of the system accounting for all major perturbations in GEO : the Earth gravitational potential at order and degree 2, the third body perturbations from the Sun and the Moon from Montenbruck and Gill (2000), and the solar radiation pressure.
Using canonical perturbation theory, we perform a rigorous averaging of the 8 degrees of freedom Hamiltonian by the method of normal forms via Lie Series (Hori, 1966; Deprit, 1969). The fast terms are then eliminated by a series of canonical transformation, revealing the long-period evolution of the different elements. This allows us to derive the forced equilibrium of this averaged Hamiltonian which is a lower dimensional object containing 5 slow frequencies defining a quasi-periodic orbit, which shows the actual nature of this pseudo–equilibrium. We obtain through a back-transformation of the canonical transformation made from the forced equilibrium, the analytical time-explicit evolution of all elements at this equilibrium. This analytical result is compared to the numerical integration of the full model before averaging, and gives satisfying accuracy. The long term evolution of the inclination and eccentricity for an object at the equilibrium are particularly analyzed showing strong dependence on the area-to-mass ratio.
We highlight that in addition to the geopotential at order and degree 2, we use a realistic model for the Sun and the Moon from Montenbruck and Gill (2000), where the Moon and the Sun are on elliptical, inclined orbits with a variation of their argument of perigee and right ascension of the ascending node. As noted in Valk et al. (2008), having a fixed Sun-Earth distance in the estimation of solar radiation pressure (an assumption made in previous studies, such as Chao (2006)) would induce spurious long-period terms in eccentricity and inclination evolution. This also ensures that the solar radiation pressure which derives from the position of the Sun is correctly modeled. Another novelty of the approach is that the Hamiltonian is derived in cylindrical coordinates since the geometry of the GEO region is very suitable for this coordinate system, therefore our approach does not need the disturbing function expansions making it simple to develop, and our results are directly translatable in Keplerian elements without singularity. The time-explicit solutions also give direct access to the equilibrium without scanning the whole phase space, and the long-term behavior described by these formulas can be used for disposal studies.

Coffee break / Poster Session / Booth Exhibition / 176

The Fate of Highly Inclined Earth Satellites: From Order to Chaos

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We consider Earth satellite orbits in the region where the perturbing effects due to Earth's oblateness and lunisolar gravitational forces are of comparable order. This range covers the medium-Earth orbits (MEOs) of the Global Navigation Satellite Systems and the geosynchronous orbits of the communication satellites. There is no complete general solution for the long-term behavior of such satellites even in the averaged problem under the quadrupolar approximation; that is, when the disturbing function, consisting of the dominate oblateness term in the geopotential together with the lowest-order term (second harmonic) in the Legendre expansion of both the solar and lunar potentials, is averaged over the periods of the satellite and the disturbing bodies. Under the further approximation that the the lunar orbit lies in the ecliptic, reducing the averaged system to two degrees of freedom, Lidov and Yarskaya (1974, CosRe 12:139) indicate the known integrable cases and investigate the geometrical behavior of the resulting motion. The analogous problem of the dynamics of planetary satellites about an oblate planet perturbed by the Sun, studied for over two centuries beginning with Laplace, was treated recently in great detail by Tremaine et al. (2009, AJ 137:3706) who found all possible equilibrium solutions and determined their stability. But despite these important contributions, we cannot yet construct a complete, logically ordered picture of the global dynamics of gravitationally dominated orbits. It becomes increasingly more difficult when the complexities of the Earth-Moon-Sun system are taken into account, giving rise to secular resonances involving commensurabilities amongst the slow frequencies of orbital precession (Rosengren et al., 2015, MNRAS 449:3522). While the existence of some of these resonances has been known since the early 1960s, the implications of their dynamical effects are still being fully probed and understood (Daquin et al., CeMDA, 2015 doi:10.1007/s10569-015-9665-9). There remain many questions, fundamental and cardinal ones, in this complex subject that are of great practical importance and call for the need to develop new, simple, and reliable models and simulation capabilities.
Here, we will recall a first-order averaged model, based on the Milankovitch vector formulation of perturbation theory, which governs the long-term evolution of orbits subject to the the predominant gravitational forces (Tremaine et al., op. cit.). The averaged equations of motion hold rigorously for all Keplerian orbits with nonzero angular momentum, and, along with their variational equations, will be given in a concise analytical vector form, which also intrinsically account for the Moon's perturbed motion. In Daquin et al. (op. cit.), it was shown that the devious network of lunisolar secular resonances that permeate the phase space of the highly inclined navigation satellites can interact to produce chaotic and diffusive motions. Using the Fast Lyapunov Indicator, they constructed dynamical stability maps that revealed a transition from a mostly stable region at three Earth radii, where regular orbits dominate, to a resonance overlapping and chaotically connected one at five Earth radii. The goal of this paper will be to explore in more detail this transition from order to chaos in Earth satellite orbits using the vector formulation, and to extend the inclination-eccentricity phase space study beyond MEO to seven Earth radii. Emphasis will be placed upon the phase-space structures near secular resonances which are of first importance to the space debris community. We will show that even the simple and deterministic equations of Tremaine et al. (op. cit.) can possess an extraordinarily rich spectrum of dynamical behaviors, answering why even without the destabilizing influence of atmospheric drag many Earth satellites eventually fall down (wt1190f). Further studies in this area may lead to deeper insights in celestial mechanics as well as provide practical results for satellite technology.

Loitering / Orbiting (II) / 82

An implementation of SGP4 in non-singular variables using a functional paradigm

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The SGP4 (Simplified General Perturbations 4) orbit propagator is a widely used tool for the fast, short term propagation of space orbits. The algorithms in which it is based are thoroughly described in the SPACETRACK report #3, as well as in Vallado et al. update. Current implementations of SGP4 are based on Brouwer's gravity solution and Lane atmospheric model, but using Lyddane's modifications for avoiding loss of precision in the evaluation of the periodic corrections, which are, besides, notably simplified for improving evaluation efficiency. Different alternatives in the literature discuss other variable sets, either canonical or not, that can be used in the computation of periodic corrections (see Izsak, Aksnes, Hoots, or Lara).
This work presents a new implementation of the SGP4 algorithm in Scala that offers a choice about the variable set used for the computation of the periodic corrections. Scala is a hybrid functional/object oriented programming language running in the Java Virtual Machine that allows for incorporating functional features in the design. Validation of the new implementations is made by carrying out different tests based on Vallado's results. Finally, applicability for massive data processing tasks like prediction of orbital collision events and performance are discussed.
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Ascent (II) / 83

Launch Vehicle Design and GNC Sizing with ASTOS

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The European Space Agency is currently involved in several activities related to launch vehicle designs (FLPP, Ariane 6, VEGA evolutions, etc). Within these activities ESA has identified the importance of developing a simulation infrastructure capable of supporting the multi-disciplinary design and preliminary GNC design of different launch vehicle configurations. Astos Solutions has developed the Multi-Disciplinary Optimization (MDO) and Launcher GNC Simulation and Sizing Tool (LGSST) under ESA contract. The functionality is integrated in the ASTOS software and is intended to be used from early design phases up to Phase B1 activities. ASTOS shall enable the user to perform detailed vehicle design tasks and assessment of GNC systems, covering all aspects

of rapid configuration and scenario management, sizing of stages, trajectory dependent estimation of structural masses, rigid and flexible body dynamics, navigation, guidance and control, worst case analysis, launch safety analysis, performance analysis and reporting. This paper will present how the workflow for the vehicle optimization and design of launcher GNC algorithms is realized in ASTOS®, DCAP, ODIN and Matlab/Simulink®. The first step comprises the definition and design of the launch vehicle including the structural mass estimation on substructure level as function of the dimensioning load case. The reference solution is used for the GNC design in Matlab/Simulink® and first performance analysis tasks with rigid body dynamics. In a second step ODIN is used for the finite-element model export, which again can be used as input to the mode-shape export computed by DCAP. Alternatively a free-free-beam model can be utilized in combination with spring damper elements between stages and at engine suspensions. The mode shapes are used for linearization of the ASTOS flexible dynamics and for controller design in MATLAB. In a third step Matlab/Simulink® is used for the controller design re-using as much information as possible from ASTOS and DCAP . Finally, the impact of the launch safety analysis on the launch vehicle design is discussed.

Coffee break / Poster Session / Booth Exhibition / 80

Using accurate ephemerides of solar system objects for autonomous navigation

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The future of space missions will be dependent on the improvement of autonomous celestial space navigation methods. Nowadays the deep space exploration uses the starry sky background for attitude determination by imaging and determining the coordinates with an embedded star catalog. Two different approaches are necessary in order to find the position of an interplanetary spacecraft. The first one when you have an approximation of the position using the latest position known, and second, the most complicated when you are lost in space. To achieve these goals we need to have an accurate model of the space environment with usable targets at any time, namely planets and asteroids in the good frame reference. The ephemerides must be computed before the mission with the goal to estimate the number of objects usable for the position determination by the use of a 3D positioning algorithm. In a first step, our method proposes to use the virtual observatory ephemeris to know the number of visible useful objects and to determine the resulting accuracy of the calculated position of the space probe. The next step will be the making of a simulation allowing to determine the parameters to be improved in order to get a more accurate position of the probe.

Optimization and Dynamics (II) / 81

On Ultimately the Most Highly Inclined, the Most Concise Solar Polar Trajectory with Practically the Shortest Period

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This paper presents the extended orbital synthesis results from the author's work in 2009 to achieve ballistic and short period out-of ecliptic trajectories which possess ultimately the most highly and most concise solar polar properties. Those are realized through almost ballistic flight instead of using electric propulsion or solar sail acceleration. The strategy developed utilizes a Jovian gravity assist first, followed by very high speed synchronized multiple polar gravity assists by Earth or Venus. So far, the use of very high speed gravity assist has been conceived not practically useful to control the trajectory energy. However, this paper presents those still effectively contribute to amending the trajectories periods, in other words, to diminishing the size

of them, and lead to acquiring small sized out-of-ecliptic ballistic trajectories. The process simply converts orbital energy associated with highly eccentric ellipses to inclination change. The biggest advantage of this strategy is to reduce propellant mass to be carried drastically, even close to zero, like ballistic flight. While the author's work in 2009 presented the trajectories down to almost one year period, this paper will present the further sequences that make the semi-major axis lower than one AU and lower the perihelion distance closer to the Sun for close-up observation of the Sun.

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Caviar: a Software Package for the Astrometric Reduction of Spacecraft Images

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Caviar is an IDL-based software package for the astrometric reduction of spacecraft images. The software was originally developed at Queen Mary University of London (QMUL) in 2003-4 for the reduction of images from the Imaging Science Subsystem (ISS) of the Cassini spacecraft. It has been used to reduce many thousands of Cassini images since the start of the main mission to Saturn in 2004, and will continue to be used until the end of the mission in 2017 and beyond. Published astrometry using the software includes the discovery of two new satellites of Saturn, Polydeuces and Anthe (Murray et al 2005, Cooper et al 2008) and an analysis of the tidal evolution of the Saturn system (Lainey et al 2015, submitted). The latest version of the software, developed at Institut de Mécanique Céleste et de Calcul des Éphémérides (IMCCE), includes an extensive restructuring of the code and a new graphical user interface. This version is shortly to be made available publicly through the NASA PDS website.

The software is currently capable of performing two primary functions: the correction of errors in the camera pointing direction, and the measurement of the astrometric positions of natural satellites. The camera pointing correction is performed by iteratively matching reference stars to their imaged positions. Currently the Tycho2 and UCAC4 star catalogues are used for reference, with star positions obtained via the Vizier web interface. In common with the estimated star positions, the astrometric positions of unresolved satellites are derived using a centroiding technique based on DAOPHOT 'Find' (Stetson 1987), while the positions of resolved satellites are estimated by iteratively fitting a shape model to the measured position of the satellite limb in the image. Approximate satellite positions are obtained from the latest JPL ephemerides using IDL routines from the SPICE software library (Acton 1996).

Future plans include the addition of options to process data generated by imaging cameras from other spacecraft missions, including Mariner 9, Viking, Voyager, Galileo and New Horizons. A capability for the automatic batch processing of multiple images is also in development. In addition, an interface to the GAIA star catalogue will be added as soon as GAIA data is available.

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Group Targets Tracking Using GM-PHD Filter Combined With Clustering

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Group targets tracking is a more complex problem of multi-target tracking. A modified Gaussian mixture-probability hypothesis density (GM-PHD) filter combined with clustering method is proposed. In the updated process of the GM-PHD filter, the proposed method introduces the dummy measurements generated by the group centers to improve the tracking performance, rather than partitions the measurements set. After estimating the single target statements, their similarities are computed. Then the estimated targets are clustered to achieve the group tracking. Finally, the track points of the group centers in adjacent time are connected to obtain the entire trajectories of the group targets. Simulations show that the proposed method can effectively track the group targets and performs better than extended target GM-PHD (ET-GM-PHD).

Low Thrust (I) / 84

CAMELOT - Computational-Analytical Multi-fidelity Low-thrust Optimisation Toolbox

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In this work a toolbox for the fast preliminary design and optimisation of low-thrust trajectories is presented. The toolbox, called CAMELOT (Computational-Analytical Multi-fidelity Low-thrust Optimisation Toolbox), solves highly complex combinatorial problems to plan multi-target missions characterised by long spirals including different perturbations. CAMELOT implements a novel multi-fidelity approach combining analytical, adaptive surrogate modelling and accurate computational estimations of the mission cost. Decisions are then made by using two optimisation engines included in the toolbox.
The main elements of CAMELOT are: • Fast Analytical Boundary-value Low-thrust Estimation (FABLE) • Multi-Population Adaptive Inflationary Differential Evolution Algorithm with Adaptive Local Restart (MP-AIDEA) • Automatic Incremental Decision Making And Planning (AIDMAP)
FABLE provides accurate cost estimations of orbital transfers using a multi-fidelity analytical and semi-analytical approach. FABLE implements an analytical propagator that includes perturbations due to the J2 zonal harmonic, drag, solar radiation pressure and low-thrust propulsion. The effect of shadow regions is also included. Different control parameterization can be implemented to analytically compute optimal transfers. In order to reduce the computation burden, FABLE can generate surrogate models of the transfers' cost to allow fast evaluation of complex trajectories.
MP-AIDEA is a single objective global optimiser based on the hybridisation of evolutionary computation and mathematical programming. The optimiser has been designed to automatically adapt its input parameters to the specific problem under consideration, in order to avoid tedious manual tuning of the algorithm.
AIDMAP is an incremental decision making algorithm that allows for planning & scheduling of complex tasks. AIDMAP incrementally builds a decision tree from a database of elementary

building blocks. The resulting graph is then evaluated using a set of deterministic or probabilistic heuristics. The deterministic heuristics in AIDMAP are derived from classical Branch & Cut algorithms while the probabilistic heuristics are bio-inspired and mimic the evolution of the slime mould Physarum Polycephalum, a simple organism endowed by nature with a powerful problem-solving heuristic.
CAMELOT has been applied to a variety of applications from the design of interplanetary trajectories to the optimal deorbiting of space debris, from the deployment of constellations to on-orbit servicing.
The paper will present two key applications. One is a multi-fly-by interplanetary mission to the inner part of the solar system to visit the fourteen known Atira asteroids, and search for new ones. CAMELOT was used to generate a globally optimal sequence of asteroids, departure and arrival dates, that allows visiting the maximum number of Atira asteroids in a given time and maximises the chance to discover new ones. The other is a solution to the problem of deorbiting multiple non-cooperative objects from the LEO region. CAMELOT was used to identify the sequence of targets that maximize the number of removed satellites, while minimizing the propellant consumption, using two different strategies: multi-target delivery of de-orbiting kits to perform a controlled re-entry, low-thrust fetch and deorbit with a single towing spacecraft. To speed up the computation, a surrogate model for the ΔV required to realise all possible low-thrust transfers between different targets was used.

Students (I) / 85

A Simulation Tool to Design of Satellite Formations

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EXTENDED ABSTRACT
A simulation and analysis tool for satellite formation is presented. The tool models the relative motion of a chief-deputy satellite formation, offers ways of selecting proper initial conditions for the formation, and helps examine the results obtained from linear and nonlinear models.
Simulation tool is developed in MATLAB environment providing visualization of the formation. User interface of the tool is created by MATLAB GUI, giving a user friendly environment (Figure 1). In this frame, this tool computes the relative dynamics of the deputy satellite with respect to the chief satellite, calculates the required initial conditions for the deputy satellite for keeping the satellites in the desired formation, and offers the required orbital corrections. Nonlinear relative dynamic model and commonly used linear models are also included in this tool, providing an environment to compare the results of the linear and nonlinear formation design approaches.
Figure 1: Main window of the simulation tool
The tool has three main parts: Pre-processing, Processing and Post-processing. In pre-processing part, user must define the chief's orbital parameters, initial conditions and the inputs required for the simulation. The main window for the deputy has two parts to compute its motion using Keplerian relative motion and using orbital parameters. These two main computations run two separate simulations.
After setting up the initial parameters, the processing part is run in MATLAB SIMULINK environment, to compute the relative motion of the deputy generating the outputs needed for the analysis of the formation. The block diagrams parts of the code are given in Figures 2 and 3.
Figure 2: Block diagram of model that use the Keplerian equations of motion
Figure 3: Block diagram of model with orbital elements.
An orbit propagator based on the Keplarian two-body equations of motion are used to simulate the orbital motion of the spacecraft. The chief's position is computed using an orbit propagator including the variation of the mean classical orbital elements. The perturbations due to non-spherical earth (J2), due to moon and sun are included on the computations. The Kepler's equation states that:

(1) where, M is the mean anomaly, E is the eccentric anomaly, is the angular velocity of the orbital mean motion, is the epoch time, and is the mean anomaly at epoch. The differential equations of motion used to obtain relative dynamics of the deputy satellite with respect to chief satellite are given in Equation-2.
(2a)
(2b)
(2c)
In the above equation, are the components of the disturbance vector, and are the control forces, and and are the relative position of the follower satellite with respect to the leader satellite expressed in leader perifocal frame.
Figure 4: Position Vectors and LVLH Axes Frame
The orbital-period commensurability, energy matching condition and initial orbital conditions are the main terms that should be taken into consideration in for developing an optimal formation keeping scheme. The required initial conditions and the required orbital corrections in order to maintain the formation are computed by using this energy matching concept (Alfriend et. al., 2010).
The relative position of the deputy is also expressed using orbital elements. This method, originally suggested by G.W. Hill, and it has been widely employed in the analysis of relative satellite motion. One of the main advantages of the orbital elements approach is to obtain a non-differential relative position equation and incorporate the orbital perturbations. The deputy's relative position defined using orbital elements is given in Equation-3 and various parameters are shown in Figure 5.
(3)
Figure 5: Orbital Elements
The post-processing part is common for both models. It provides to visualize all the parameters computed by the processing part. Some of the results that may be displayed are the relative distance between two satellites, the projection view of the relative motion, the orbital parameters (semimajor axis, inclination, etc.), thrust implementation and its value. Here, the orbital motion in a 2D Earth map and in a 3D Earth model are also used to visualize the motion (Figure 6) Furthermore, it is possible to visualize the motion by running 3D Animation. Here, the 3D motion is given with respect to ECI and ECEF frames, both in space view or satellite view (Figure 7).
Figure 6: 2D Mapping
Figure 7: 3D Animation
In the final manuscript, the details of the simulation tool will be provided. Final manuscript will focus especially on the selection of the initial conditions defined in terms of orbital elements of the chief and deputy satellites. The linear procedures used for determining the initial conditions for relative motion consider near circular orbit and they are valid for orbits with small eccentricities and even for close formations (i.e., the distance between satellites is about 1 km). However, for formations with relative distances greater than 10 km, it is not possible to obtain a stable formation by using the initial conditions computed using linearized schemes. In this case, nonlinear effects should be added to the initial conditions predictions. In the final manuscript, we will present the orbit analysis tool in detail. Also to be presented is a new method that incorporates the nonlinear effects in the selection of initial conditions in terms of orbital parameters. The approach taken will be described in detail, and the success of the approach will be demonstrated through simulations.
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Capitalizing on Relative Motion in Electrostatic Detumble of Axi-Symmetric GEO Objects

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Touchless methods of actuating and detumbling large Earth-orbiting objects is of increasing importance for active debris mitigation strategies. Previously developed are electrostatic detumble dynamics and simulations for deep-space and lead-follower formations. This study investigates the influence of the instantaneous position and relative formation on electrostatic detumble performance. Developed are the mathematical sensitivities to relative position to enable optimal relative guidance studies. The newly developed Linearized Relative Orbit Elements (LROE) formation flying controller is exhibited and applied for formation maintenance. The benefits of formation flying in electrostatic detumble scenarios and the advantages of the LROE controller for electrostatic actuation applications are demonstrated through numerical simulation.

Orbit Determination and Prediction Techniques (I) / 3

Observation of orbital debris with space-based space surveillance constellations

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Since the first orbital launch back in 1957, the population of space debris in orbit around the Earth has steadily risen.
As the orbital debris population grows, the likelihood of catastrophic phenomena like the collision between two orbiting objects increases. In order to limit the proliferation of space debris in orbit, a great number of standards, guidelines and even laws have been put in place since the end of the 90's. In this scenario, a thorough and accurate bookkeeping of space objects is paramount. Space surveillance has thus become our most reliable ally to safeguard space missions from the threat of collisions.
The BAS3E simulator (Banc d´Analyse et de Simulation d'un Système de Surveillance de l'Espace) is a CNES software tool, developed in collaboration with GMV, with the following capabilities: orbit determination of space objects, generation of optimum observation plans, collision forecast, anticipation of dangerous reentries, and detection of debris fragmentation. Furthermore, BAS3E has the capability to simulate observations of space objects obtained by a given sensor network taking into account sensor visibility constraints. Orbit and attitude ephemerides, quality, precision, and usage cost; are some of the parameters that shall be defined for each sensor. Originally conceived for ground-based observations (telescope and radar), BAS3E has been recently enhanced to enable the definition of "orbiting" sensor sites, which allow for the simulation of space-based space surveillance sensors.
Using such space surveillance system simulator, this paper evaluates the feasibility to use on-board sensors for both Low Earth Orbit (LEO) and Geostationary Orbit (GEO) object surveillance. The main goal is to assess the ability of a space-based space surveillance constellation, to detect and catalogue the space debris population on these both orbital regimes. The orbit determination accuracies that can be attained when space objects are tracked by different space-based sensor configurations have also been studied and will be presented in this paper.
Different constellations of equispaced spacecraft following quasi-circular, Sun-synchronous dawn-dusk orbits have been analysed, for which the constellation altitudes and the number of satellites were varied. For simplicity reasons, we assumed that spacecraft followed an attitude profile which

ensured the pointing of the on-board sensor towards the object. This assumption permits the study of the attitude constraints required by each sensor in order to detect, track and catalogue a given space object.

Low Thrust (I) / 7

Low thrust orbit transfer optimiser for a Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation)

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The paper describes the general strategy for electric propulsion orbit transfers, a open source optimiser for orbit transfer based on averaging techniques and the integration of such trajectory optimiser into EcosimPro® ESPSS (European Space Propulsion System Simulation). EcosimPro® is a Physical Simulation Modelling tool that is an object-oriented visual simulation tool capable of solving various kinds of dynamic systems represented by a set of equations, differential equations and discrete events. It can be used to study both transients and steady states. The object oriented tool, with the propulsion libraries ESPSS from ESA for example, allows the user to draw (and to design at the same time) the propulsion system with components of that specific library with tanks, lines, orifices, thrusters, tees. The user enhances the design with components from the thermal library (heaters, thermal conductance, radiators), from the control library (analogue/digital devices), from the electrical library, etc. The use of the new feature included into ESPSS, the satellite library is particularly interesting for orbital manoeuvres because the satellite library includes the flight dynamic (orbit and attitude) capabilities for a full spacecraft including orbital, attitude perturbations and power concerns during Sun's eclipse phases. In order to simulate realistic missions, an optimiser for orbit transfer has been integrated thanks to the design of few new components for interfacing the optimiser and the existing library. Hence the simulations take into account the interactions between the AOCS and the optimal thrust direction wanted to perform the orbit transfer and real strategies for power management during eclipses. Full satellite missions, for example continuous electric orbit transfer from GTO and from super-GTO to GEO are presented and particular behaviours highlighted.

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Finding Continuous Zero Curve of Homtopy Method for Low Thrust Trajectory Optimization

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Homotopy method serves as a a useful tool in solving low thrust optimal orbital transfer problems, for which good initial guesses are difficult to obtain. The principle of homotopy method is to embed a given problem, namely the initial problem, into a family of problems parameterized by a homotopic parameter. The solutions to the embedded problems can be tracked iteratively with the homotopic parameter varying continuously, and the homotopy path emanates from the initial problem and reaches the original problem is called a continuous zero curve. Unfortrantely, it is not easy to find zero curve for many nonlinear problems. In this paper, a newly developed homotopy parameter bounding method is utilized to seek multiple solutions of the initial problem, and the homotopic path is tracked started from each intial solution to check whether it is a zero curve. The process is repeated until a continuous zero curve is found. A three-dimensional low-thrust orbital transfer problem is presented to illustrate the applications of the method.

Rendezvous & Docking (II) / 109

GNC design and validation for rendezvous, detumbling, and de-orbiting of ENVISAT using clamping mechanism

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This paper describes the development and validation of a GNC for active debris removal of the ENVISAT S/C, using clamping mechanism. The design of the GNC is focused on the phases where the choice of clamp mechanism is of particular relevance to the ADR execution. These mission phases include the pre-capture, close range rendezvous with uncooperative target, the post-capture detumbling and stabilization of the composite, and the composite de-orbiting. The derived GNC solutions are based on robust MIMO control, which is especially suited to the considered mission scenario. Namely, the GNC is designed considering S/C dynamics with multiple-bodies attached by clamping mechanisms, with flexible appendages and sloshing effects, and with uncertainties in the mass, centre-of-mass and inertia (MCI) parameters, among others. The results of the GNC design activities are presented, namely in what respects to: the multi-body modelling of the chaser and composite S/C, the LFT modelling of the plant dynamics for GNC synthesis, the GNC trade-offs, the resulting GNC architecture and modes, and, finally, the methodology and results adopted for the Guidance and Control subsystems design. An advanced verification and validation framework for GNC was also developed, and is presented therein, that includes analytical validation of GNC robustness, stemming from the mu -synthesis framework. In addition, numerical validation is performed in a high-fidelity simulator of S/C dynamics, including multi-body dynamics, flexible modes, sloshing, MCI perturbation, environmental disturbances, among others. The validation results are obtained for the nominal scenario, as well as for alternative definitions of ENVISAT rotational motions and of relative sensor technology. Finally, Monte-Carlo campaign is adopted in complement to the analytical validation, allowing for the consolidated derivation of a set of requirements and recommendations for future missions, especially those within the Cleanspace initiative.

Interplanetary Flight and Non-Earth Orbits (II) / 102

Extended Tisserand graph and multiple lunar swing-by design with Sun perturbation

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The use of multiple lunar swing-bys to pump up the hyperbolic escape velocity of interplanetary trajectories has been proposed in literature and repeatedly put into practice in real missions (*Kawaguchi et. al., 1995, Dunham et. al., 2007*). JAXA's technology demonstrator mission DESTINY (*Kawakatsu et. al., 2013*), with its new main mission objective to fly by asteroid Phaethon, is planning to make use of a similar strategy to obtain the required escape velocity after a low-thrust spiralling phase from its launcher injection orbit.
This paper presents a systematic approach to design multiple lunar swing-by sequences that can be applied for this purpose. The Sun third-body perturbation plays an important role essentially providing free Δv between lunar swing-bys. As a first step, an extension of the classical Tisserand graph in perigee-apogee radius is presented, in which the potential gains by solar perturbation

between flybys can be estimated. Secondly, following an approach proposed by *Lantoine & McElrath (2014)*, a database of Moon-to-Moon transfers is generated with a continuation method. A simplified planar circular restricted three-body problem is assumed. The families of transfers are stored parametrised as a function of initial Sun-Earth-Moon angle, lunar hyperbolic escape velocity modulus, and direction. In addition to the families calculated by *Lantoine & McElrath*, new families with multiple revolutions and families with energies close to libration point orbits are found and generated. The database can be accessed and transfers retrieved to quickly generate sequences in a similar fashion to a multiple swing-by classic Lambert problem solver, but including the effect of the Sun third body perturbation.
Two particular practical examples are presented: a sequence to be used for the DESTINY mission to escape the Earth-Moon system and initiate the transfer to asteroid Phaethon, and a solution to obtain a transfer to the Earth-Moon L_2 point.

Verification and Validation Methods / 103

ROSPA, cross validation of the platform-art and ORBIT test facilities for contact dynamic scenario setup and study

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The set-up and use of ground validation testing facilities from the early phases of the missions can provide a very valuable feedback to the equipment and technologies being developed. The validation activities of the own ground testing facilities are key to the usability and confidence of the results obtained from them. GMV's platform-art dynamic test bench has already been validated (for navigation purposes based on optical cameras) with flight data coming from PRISMA mission through the PRISMA-HARVD experiment. In addition, platform-art dynamic test bench is currently being extended thanks to ESA loan of several new devices, including a new high-span KUKA robotic arm, which will extend the functionality of the current test bench. On the other hand, ESA has established an air-bearing facility known as ORBIT (Orbit Robotics Bench for Integrated Technology). The facility located within ESTEC's Automation and Robotic laboratories provides several air-bearing platforms which can move frictionless on a 45 m^2 flat floor.
The in-space Robotic Servicing Physical Assessment (ROSPA) is a study with the purpose of recreating and studying the dynamics during/after contact between target and chaser in a rendezvous and capture mission. The data of the experiments run in the platform-art (GMV) and in ORBIT (ESA) will be used for a cross-validation of the facilities.
Two different scenarios have been setup in both facilities: simple contact and gripping scenario. In the simple contact scenario the Mitsubishi PA10 robotic arm approaches the specifically designed mock-up mounted on the air-bearing (or on the KUKA robotic arm, in platform-art) and touches it through a compliance device and a load cell to measure the contact forces and torques. In the gripping scenario the compliance device is replaced by a gripping device, which after an open loop trajectory attempts the gripping of a Launch Adapter Ring (LAR) mock-up.
In the scope of the activity also another functionality of the platform-art facility is exploited: the space-like environment simulation. This functionality has been developed and demonstrated in the frames of previous collaborations between ESA and GMV, for projects such as NEOGNC (Interplanetary mission, MarcoPolo-R) and ANDROID (Active Debris removal mission). The output of this activity is a data base of representative images taken during an open loop sequence of the robotic arm approach and the gripping with the LAR of a mock-up of the TANGO spacecraft (the 2nd spacecraft of the Swedish PRISMA mission). The representativeness of the images in such a close range scenario is meant in terms of illumination conditions and disturbances recreated in laboratory like fuel on lens, micro pieces of MLI floating and thruster plume in the Camera FoV.

Rendezvous & Docking (II) / 100

GNCDE as DD&VV environment for ADR missions GNC

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GNCDE is an integrated GNC development and verification environment, developed by GMV in the frame of an ESA-GMV co-funded activity. It contains templates of four different adaptable scenarios (Rendezvous and Docking, 3-axis stabilization, Formation Flying, Launchers), a complete set of libraries for sensors, actuators, DKE and GNC blocks, and a set of tools to ease the design and analysis of the mission (e.g. guidance trajectories design, control and estimation synthesis, covariance analysis, Monte Carlo campaign, Statistical analysis, Autocoding, 3D visualization through direct connection with tools like Celestia, etc.). GNCDE has been already successfully used to design the GNC of different rendezvous missions (such as Advanced Re-entry Vehicle and Mars Sample Return Orbiter) and it is the current development environment for the formation flying software of PROBA-3 phase CDE.
This paper will focus on the utilization of GNCDE for assessing GNC concepts of two different ADR scenarios, both aimed at the post-life disposal of ENVISAT:
1) Design, development, verification and validation of the GNC for RDV and de-orbiting phases of E-Deorbit mission, currently in phase B1. E-Deorbit is so far the most advanced ESA activity with the objective of de-orbiting ENVISAT. It is unique in its operational complexity and requires a high reliable and strongly validated GNC design.
2) Quick preliminary feasibility evaluation from GNC point of view of PRIDE vehicle used as active debris removal spacecraft. PRIDE is the ESA program aimed at developing a reusable robotic spacecraft with different in-orbit servicing capabilities, among which the possibility to serve as an ADR vehicle.
The high flexibility of GNCDE has permitted to adapt very quickly the rendezvous and docking template (originally used for an ATV-ISS docking scenario) to the two different ADR scenarios, parametrizing it opportunely to include configuration, initial orbital and attitude data of both ENVISAT and the chaser spacecraft. Sensors and actuators parameters have been also modified to take into account the typical accuracies and errors in the two cases. The rendezvous trajectories have been tailored to these scenarios and the GNC laws adapted to their specific needs.
In the case of E-Deorbit, the work to be done has a long schedule aiming at a fully validated GNC and the design is still on-going. The paper will present the process which is being followed for GNC DD&VV of this specific scenario, how this process is supported by the GNCDE environment and the available preliminary results. In the case of PRIDE scenario, the study preliminary indicates that the vehicle could be suitable for an ENVISAT ADR mission. Using the link between GNCDE and Celestia, a video showing the capture phase, including synchronization between PRIDE and ENVISAT, has been also set up.

Coffee break / Poster Session / Booth Exhibition / 101

The NAROO project for overcoming past, current, and future ephemeris errors

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Accurate orbit determination require a large amount of observations dispatched over a large time span to allow for the best precision and extrapolation. This latter is of high importance in the context of future space missions.
In practice, most orbital models of the Solar System objects are fitted to data covering typically about one century. Even if the conditions are required for precise dynamical modeling, we emphasize an important caveat : ephemerides can sometimes be significantly biased while their

extrapolation quickly diverges. Even though this could be due to various reasons, we found that an important one consists in the imprecision of past observations that are introduced in the adjustments. These observations were processed a long time ago with inaccurate star catalogs and with inaccurate methods compared to recent ones. No real efforts have been attempted to reanalyze these data a new time, considering the amount of time, mean and energy required. Using photographic plates of planetary satellites, we demonstrate that a new reduction of old observations can improve significantly the ephemerides. In this framework and with support of the Gaia mission, the NAROO project has been initiated at Paris Observatory with the primary aim to reprocess the old astrometric observations with the best instrumental, algorithmic and numerical techniques. We discuss the impact of the project on future planetary and satellite ephemerides.

Multidisciplinary Design Optimization / 106

A Self-Boundary Fall Free Algorithm for 2D Open Dimension Rectangle Packing Problem of Satellite Module

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Layout design of spacecraft module belongs to scheme design problem, which has been proved to be NP hard. This problem has not only computing complexity but also engineering complexity, and it is more difficult to tackle the challenge of practical application in engineering. In practical engineering, the dimension of satellite configuration is usually unknown and needs to be optimized (generally minimized) as well, while the dimensions of satellite components are known. Assumed that the material densities of all modules are same, the layout design of satellite module can be simplified to a 2D open dimension rectangle packing problem when the satellite configuration is cuboid. In this problem, given a set of rectangles with known dimensions, the arrangement of these rectangles should be determined without overlapping and inside a predefined enveloping rectangle. This paper proposed a self-boundary fall free genetic algorithm (GA&SBFFA) for the open dimension rectangle packing problem, which is used to optimally arrange the rectangles densely and minimize the area of enveloping rectangle. Meanwhile, the shape of the enveloping rectangle is maintained as square as possible so as to satisfy the static equilibrium requirement in some complex system design, e.g. satellite. The main procedure of SBFFA is as follows. First, the dimension of enveloping rectangle is determined by the packing items, which is different from the methods published previously. Next, the information of feasible space where the next item can be placed is recorded, which includes the widths of the feasible spaces and the coordinates of the feasible points where the bottom left vertex of the next item can be placed. Finally, the next item is placed according to the principle of minimal potential energy. Based on SBFFA, the minimum enveloping rectangle space is calculated with given sequence of packing items. GA is used to solve the packing optimization problem by searching the optimal item sequence. Two experiments are used to testify the proposed method and the efficacy is demonstrated. The computational expenses were reduced to about 30 seconds when there are 50 items, which is much less than the reported methods. Keywords: Packing problem; Layout optimization; Area minimization.
Layout design of spacecraft module belongs to scheme design problem, which has been proved to be NP hard. This problem has not only computing complexity but also engineering complexity, and it is more difficult to tackle the challenge of practical application in engineering. In practical engineering, the dimension of satellite configuration is usually unknown and needs to be optimized (generally minimized) as well, while the dimensions of satellite components are known. Assumed that the material densities of all modules are same, the layout design of satellite module can be simplified to a 2D open dimension rectangle packing problem when the satellite configuration is cuboid. In this problem, given a set of rectangles with known dimensions, the arrangement of these rectangles should be determined without overlapping and inside a predefined enveloping rectangle. This paper proposed a self-boundary fall free genetic algorithm (GA&SBFFA) for the open dimension rectangle packing problem, which is used to optimally arrange the rectangles densely and minimize the area of enveloping rectangle. Meanwhile, the shape of the enveloping rectangle is maintained as square as possible so as to satisfy the static equilibrium requirement in some complex system design, e.g. satellite. The main procedure of SBFFA is as follows. First, the dimension of enveloping rectangle is determined by the packing items, which is different from the

methods published previously. Next, the information of feasible space where the next item can be placed is recorded, which includes the widths of the feasible space and the coordinates of the feasible points where the bottom left vertex of the next item can be placed. Finally, the next item is placed according to the principle of minimal potential energy. Based on SBFFA, the minimum enveloping rectangle space is calculated with given sequence of packing items. GA is used to solve the packing optimization problem by searching the optimal item sequence. Two experiments are used to testify the proposed method and the efficacy is demonstrated. The computational expenses were reduced to about 30 seconds when there are 50 items, which is much less than the reported methods. Keywords: Packing problem; Layout optimization; Area minimization.

Environment Modelling / 107

The SPENVIS Next Generation

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ESA's Space Environment Information System (SPENVIS) is an on-line resource for evaluating the space environment and its effects on spacecraft components and astronauts. The SPENVIS system provides access to a large number of space environment models and related analysis tools, allowing the users to combine and chain results between different models. SPENVIS has a long and acclaimed history. Since its first development at the Belgian Institute for Space Aeronomy (BIRA-IASB) in 1996, it has been successfully operational for more than fifteen years. As a result, SPENVIS has established a mature user community from all over the globe that is using the system for various purposes including mission analysis and planning, education and scientific research. Recently, a new system known as SPENVIS Next Generation has been developed under the ESA/GSTP-5 programme. The key objective was to upgrade SPENVIS into a new web-based service-oriented distributed framework supporting plug-in of models related to the hazardous space environment, and including both a user-friendly interface for rapid analysis and a machine-to-machine interface for interoperability with other software tools. The purpose of this talk is to introduce the new SPENVIS system and its capabilities.

Orbit Determination and Prediction Techniques (II) / 104

From Simulation to Reality: Cataloguing of Objects in from Ground-based Optical Observations

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This paper presents the approach for building a catalogue of Earth-orbiting objects from optical observations in surveillance mode. The cataloguing is based on the DEIMOS CORTO tool (CORrelation Tool) which is described in this paper. CORTO has evolved from of a simulated cataloguing system to a cataloguing software suite intended to process data from real observations. Thus, it merges the knowledge derived from simulation experience and the main constraints

imposed by real observation activities. The main differences among these two approaches are highlighted in the paper. CORTO processes observations from several sensors in a sequential way. For each incoming observation, it attempts to correlate it to an existing object in the catalogue. If such correlation is possible, an orbit determination process is performed on the incoming measurement in the basis of a-priori state vector and covariance of the object. If such correlation is not possible, an initial orbit determination is carried out, and a new object associated to that measurement is created. CORTO allows a cold start of the cataloguing (i.e, it does not need any external catalogue to start) and thus, can maintain orbital information of object not included in public TLE file. The main results from DEIMOS cataloguing experience are summarised, describing the observation strategy and the measurement distribution considered necessary for achieving a proper cataloguing capability. This summary highlights the main difficulties that can be found, in the correlation activities which impose a several-step approach to correlation in order to avoid miss-correlation of objects. The approach undertaken in the CORTO software is based on a three step process: firstly, a correlation in the basis of comparison of observation with expected visibility periods and rough observation angles is carried out for every object. A second orbit determination compatibility cross-check based on the filtering residuals is performed later. Finally, a procedure for removing false objects (i.e, objects created by spurious measurements), and/or to remove objects which are observed sparsely is performed asynchronously. The system is intended to run in a mostly automated way, but allows an operator to assess the correlations performed automatically by the system, and to correct them if necessary. In addition to this, it is possible to correct errors related to manoeuvring objects. If the operator knows with certainty that a manoeuvre has taken place, information regarding that manoeuvre can be loaded into the catalogue. The operator can also infer when an impulsive manoeuvre has happened with support from the system. Finally, the system allows the user to perform an iterative process in order to estimate the area-to-mass ratios associated to each of the objects in the catalogue. The CORTO cataloguing system is accompanied by a set of auxiliary tools, also described in the paper, which complete the capabilities of the system to ensure the proper cataloguing process. These tools include: CALMA for calibration of observation stations (used to qualify a number of observatories), CORTOEditor, to support operator for operational maintenance of the catalogue, and CHOCO which optionally allows correlating the observed objects with the TLE data. This tool serves to assign the international ID to the CORTO objects, but is not mandatory for successful correlation of objects within CORTO. The catalogue is finally made available through a restricted web system (CAWEB) that supports the monitoring of the catalogue. The paper presents the main results from an observational campaign executed in October 2014 focused on the cataloguing of high altitude objects. The campaign lasted 9 consecutive observing nights, providing more than 200.000 observations from three surveillance and a tracking telescopes located in Spain. Those observations are used to feed-up the CORTO cataloguing system, and have allowed creating a catalogue of objects which are observable from southern Europe. In particular GEO ring longitudes covering Europe are well represented. About 300 objects are systematically observed during several nights, eventually reaching accurate orbits. The achievable accuracy of the observed orbits can reach values around 10-100 meters. Object manoeuvres are also observable. Example cases of observed manoeuvres are reported.

Debris, Safety and Awareness (II) / 105

Fragmentation Event Model and Assessment Tool (FREMAT) supporting in-orbit fragmentation analysis

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The future sustainability of the near Earth environment requires continuing efforts to increase our knowledge of the current and future debris population. Possible on-orbit fragmentation events are a major concern nowadays. The Fragmentation Event Model and Assessment Tool (FREMAT) project for ESA was carried out with the objectives of simulating on-orbit fragmentations, assessing their impact on the space population and evaluating the capability of identification of

fragmentation events from existing surveillance networks. In the frame of the FREMAT activity, the implementation of several algorithms related to on-orbit fragmentation events was carried out. FREMAT encompasses three individual tools: Fragmentation Event Generator (FREG), Impact of Fragmentation Events on Spatial density Tool (IFEST) and (Simulation of On-Orbit Fragmentation Tool) SOFT. Fragmentation Event Generator (FREG) has been conceived to simulate fragmentation events (explosion and collisions). A breakup model based on recent models was the baseline for this tool .We have enhanced the baseline NASA break up model, in order to ensure the consistency of mass and momentum in the created fragment clouds. Its output is one or two clouds of fragments that can later be fed into IFEST or SOFT, or to any other propagator. The second tool, IFEST (Impact of Fragmentation Events on Spatial density Tool) allows the evaluation of the impact of on-orbit fragmentations in the space debris population. This tool employs a fast semianalytic propagator for computing the long-term evolution of the clouds of fragments (up to hundreds of years) obtained from FREG, and computes the spatial density caused by those fragments as well as the percent increase in the background spatial density obtained from MASTER. The computation of the spatial density within this tool is validated against results provided by Esa's POEM tool. Finally, the third tool, SOFT (Simulation of On-Orbit Fragmentation Tool), has been created to simulate the determination of the type of fragmentation and the objects involved in a fragmentation event when a space surveillance network detects a number of unexpected new objects and a fragmentation event is considered a possible cause. It can process a cloud from FREG, and clouds from other sources can be adapted to be processed by SOFT. Uncertainties in the knowledge of the orbits of the fragments and the presence of foreign objects is also considered. . The tool determines the type of fragmentation, calculates the time and location of the event and identifies the parent objects. This paper presents a description of the algorithms implemented in this toolkit, a brief description of each tool and a brief summary of their main functionalities. Furthermore, study cases are presented including parametric analysis by means of introducing variations in the input parameters of the fragmentation model. Short and long-term evolution of the clouds are studied, as well as the feasibility of determining the location and time of the fragmentation event. Additionally, the influence on the increase of collision risk is assessed.

Loitering / Orbiting (I) / 39

An efficient code to solve the Kepler equation for elliptic and hyperbolic orbits

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The Kepler equation for the elliptical motion, $y - e \sin y - x = 0$, involves a nonlinear function depending on three parameters: the eccentric anomaly $y =$ E, the eccentricity e and the mean anomaly $x =$ M. For given e and x values the numerical solution of the Kepler equation becomes one of the goals of orbit propagation to provide the position of the object orbiting around a body for some specific time (see references [1–6]). In this paper, a new approach for solving Kepler equation for elliptical and hyperbolic orbits is developed. This new approach takes advantage of the very good behavior of the Laguerre method [7] when the initial seed is close to the looked for solution and also of the existence of symbolic manipulators which facilitates the obtention of polinomial approximations. The central idea is to provide an initial seed as good as we can to the modified Newton-Raphson method, because when the initial guess is close to the solution, the algorithm is fast, reliable and very stable. To determine a good initial seed the domain of the equation is discretized in several intervals and for each one of these intervals a fifth degree interpolating polynomial is introduced. The six coefficients of the polynomial are obtained by requiring six conditions at both ends of the corresponding interval. Thus the real function and the polynomial have equal values at both ends of the interval. Similarly relations are imposed for the two first derivatives. Consequently, given e and $x =$ M, selecting the interval $[x_i, x_{i+1}]$ in such a way that M ϵ $[x_i, x_{i+1}]$ and using the corresponding polynomial $p_i(x)$, we determine the starter value $y_o = E_o$. However, the Kepler equation has a singular behavior when M is small and e close to unity (singular corner). In this case, the exact solution of the equation has to be described in a different way to guarantee the enough accuracy to be part of the seed used to start the numerical method. In order to do that, an asymptotic expansion in power of the small parameter

$\varepsilon = 1 - e$ is developed. In most of the cases, the seed generated by the Space Dynamics Group at UPM(SDG-code) leads to reach machine error accuracy with the modified Newton-Raphson methods with no iterations or just one iteration. The final algorithm is very stable and reliable. This approach improves the computational time compared with other methods currently in use. The advantage of our approach is its applicability to other problems as for example the Lambert problem for low thrust trajectories.

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Debris, Safety and Awareness (III) / 38

Space dynamics software ELECTRA

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The objective of this article is to present the current state of the CNES flight dynamics software ELECTRA maintained by Capgemini and some of the main principles used for validating its scientific and computational facets in an industrial context.

First of all, we will describe the development history. Some pieces of software have been created by industry. Other parts have been developed initially by research teams, based on the ELECTRA method the CNES developed, followed by an industrialization process to meet industrial standards and code quality requirements. Following the prototype phase, the development of the operational tool ELECTRA was decided by CNES Steering Committee of Safeguard Working Group in July 2006 and started in 2007. At the beginning, ELECTRA was implemented for internal CNES safety needs, but it was soon provided to space operators to assess victim risk associated with their operations and comply with the French Space Operations Act. Since December 2010, ELECTRA has been deployed and used operationally to monitor the risk associated with each launch from Guyana Space Centre in place of the existing "SUZHANE" tool. ELECTRA was written in FORTRAN. In 2015, it was decided to port ELECTRA in JAVA.

Secondly, we will describe the functional coverage of ELECTRA. The ELECTRA software computes the risk of making a victim during atmospheric reentries, with or without taking into account protection coefficients in several contexts (Random (or Uncontrolled) reentry, Controlled reentry, launch failure, Reentry on final orbit). ELECTRA computes two complementary estimations of the risk, the probability of causing at least one victim and the expected value of the number of victims, taking into account fragmentation of the spatial vehicle after atmosphere re-entry. The risk computation is done by assessment of fragment impact location and probability of occurrence and consideration of population distribution and habitat protection.

Third, a description of the underlying architecture of the Fortran and JAVA software components will be proposed: different libraries, the role of the different libraries, how each library has been constructed to meet needs of modularity and re-use. We will also point out some limitations of the existing architecture, and pit-falls which should be avoided in JAVA future developments.

Then, we will explain the methods and principles used for validating the new version of ELECTRA (JAVA). Two different and complementary approaches are used: non-regression based on comparison with the previous version developed in Fortran language, and "from scratch" validation in case of major evolutions or implementation of new functionalities. Using examples, we will present the mechanisms, tools and documents used for following the validation of the tool through different versions, and some principles and feed-back of validating new functionalities.
As a conclusion, we will present what seems to us to be the key topics for developing and maintaining the flight dynamics software ELECTRA in terms of architecture and validation, and how this could be taken into account in developing new products.

Open Source (II) / 33

An open-source, modular software architecture for astrodynamics simulation

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Open-source software plays an increasingly important role in astrodynamics research. In this paper, we describe a new open-source, modular software architecture for astrodynamics simulation. We describe the rationale behind the setup of the software architecture and provide a snapshot of an implementation in C++ using generic programming concepts. The implementation, dubbed "astropnp", includes a few different libraries that provide the basic building blocks to construct astrodynamics simulations. We highlight how this architecture lends itself to "plug-and-play" use and provide insight into a simulation use-case for spacecraft rendezvous.
Our paper describes an attempt towards building an architecture for astrodynamics simulations based on the notion of "micro-services" (Kosar, 2012). The concept of micro-services stems from the idea of building up software using "services" that are small, highly decoupled and focussed on executing small, isolated tasks. Micro-services architectures also feature services that can be swapped out easily. In the most general sense, these services are orchestrated by language-agnostic Application Program Interfaces (APIs). Implementing an architecture of this nature affords the software a great deal of versatility and modularity.
We survey the use of the micro-services approach towards the implementation of software for, but not limited to, mission analysis, interplanetary trajectory design, rendezvous & docking and formation flying. In addition, we present a case study based on astropnp for spacecraft rendezvous using robust, nonlinear, feedback control. The services for this case study include a generic mathematics library, a generic astrodynamics library and a library with functions for proximity operations in space (in addition to standard libraries for linear algebra, storage etc.). The case study is conducted within the context of a mission concept for Active Debris Removal (ADR), as part of the Stardust network (EU FP7 Marie Curie Initial Training Network).
The software architecture presented here follows on from previous work to establish the TU Delft Astrodynamics Toolbox (Tudat) (Kumar, et al., 2012). Our study includes an overview of existing open-source software tools for astrodynamics, including Tudat, General Mission Analysis Tool (GMAT), Java Astrodynamics Toolkit (JAT), Orekit and PyKEP. Based on this overview, we comment on how these tools might be employed within our modular software architecture. We present a roadmap to improve upon the architecture and generate a tasklist to raise the readiness level of astropnp in anticipation of future wide-spread use.

Rendezvous & Docking (I) / 32

Experimental evaluation of Model Predictive and Inverse Dynamics Control for spacecraft proximity and docking maneuvers

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An experimental campaign has been conducted to evaluate the performance of close proximity and docking maneuvers of two different guidance and control algorithms based on Model Predictive Control and on Inverse Dynamics Control. The metrics of performance includes fuel usage, time to target, computational burden and constrain handling.
The experiments have been conducted on two ˜10 kg Spacecraft Simulators that float via air-pads over a 4-by-4 meter polished granite monolith surface recreating a reduced gravity and a quasi-friction-less motion in two translational and one rotational degrees-of-freedom (planar motion). By using eight cold-gas thrusters and a reaction wheel, the Spacecraft Simulators are capable of autonomous motion over the floating surface. An onboard tank of compressed air (propellant), a power system and on-board computer give full autonomy to the Spacecraft Simulators. All the required processing (sensor readings, communications, navigation, guidance and control, and actuator commanding) is handled on-board in real-time. The experimental set-up will be described in detail. The navigation problem has been considered solved with the Spacecraft Simulators sensing their position and attitude by using an overhead optical positioning system (VICON) augmented by an on-board Fiber Optics Gyroscope. The accuracy of the Spacecraft Simulator position and attitude knowledge (as well as the knowledge of the target state) can be artificially deteriorated to simulate real sensor limitations and constraints (e.g precision and field-of-view of a RADAR system).
With the Model Predictive Control framework, a cost function (e.g fuel consumption) subject to system dynamics and constraints (e.g. maximum available control actuation level and obstacle avoidance) is minimized over a discretized time period with a finite prediction horizon. By solving the optimization problem, a control input for each discrete time sequence is generated.
In the Inverse Dynamics Control approach the trajectory of the spacecraft is simplified to a known function depending on a set of parameters (e.g. a polynomial). The desired initial and final conditions are then imposed on that trajectory and the parameters that are left unset are then optimized to meet other constrains whilst minimizing a cost function (e.g. fuel consumption). The control input to follow the prescribed trajectory is then generated from the prescribed trajectory and applied to the system.
Both guidance and control approaches eventually reduce to two different non-linear optimization problems that need to be solved periodically in order to provide the control inputs. In this study, the open-source Interior Point OPTimizer (IPOPT) software package has been used.
The set of test scenarios that have been conducted are designed to represent a wide set of rendezvous and proximity operations scenarios (unconstrained and constrained, cooperative and uncooperative docking and proximity operations with or without obstacle avoidance). The goal has been to define a set of tests that can be used to benchmark different guidance algorithms so that a meaningful comparison of different approaches can be made.

Orbit Determination and Prediction Techniques (I) / 31

Processing Two Line Element sets to facilitate re-entry prediction of spent rocket bodies from the geostationary transfer orbit

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Upper stages of rockets are large objects, which contain components that are known to be able to survive atmospheric re-entry. Such surviving material, for example propellant tanks, will impact Earth's surface and might cause ground casualties. Predicting the satellite re-entry, and thus also impact location is notoriously difficult; re-entry prediction is associated with uncertainty in the order of 10% of the remaining lifetime in-orbit. This makes managing the ground casualty risk, by issuing actionable impact warnings, challenging. Thus, the risk posed by spacecraft re-entries will be reduced if the accuracy with which such events can be predicted is improved.
At present, Two Line Element sets (TLEs) are the only publicly available data that can be used for re-entry prediction of a space object. However, there is a number of factors that, if unaddressed, could reduce the accuracy of re-entry prediction based on TLEs:

1. The quality of TLEs of an object is not homogeneous; sometimes TLEs of low quality or even belonging to a different object are published.
2. Occasionally, the object or its orbit can be altered by collisions, fragmentations or space weather phenomena. Such space events render the TLEs of the object from before the event inapplicable to its new, changed state.
3. TLEs do not provide information on space object parameters, such as ballistic coefficient (BC) or solar radiation pressure coefficient (SRPC). TLEs only include the B* parameter that accounts for combined atmospheric drag and solar radiation pressure forces, not BC and SRPC individually.
4. TLEs can only be propagated using the SGP4/SDP4 propagator. However, this propagator is based on the Brouwer theory and, therefore, only models the largest perturbations affecting a satellite. The many assumptions of the theory can severely limit the accuracy of the resulting propagation and thus of the re-entry prediction.
5. TLEs are not supplied with uncertainty information, e.g. a covariance matrix. It is thus challenging to estimate the accuracy with which the re-entry is predicted based on these ephemerides.

In order to overcome these difficulties in TLE-based re-entry prediction, a multi-step procedure is proposed. The first step consists of analysing TLEs, with the goal of identifying outliers, space events, and dynamical phases of re-entry, where the drag is relatively high or low. The filtered TLEs are then used to estimate the unknown spacecraft BC and SRPC. The last step consists of performing an orbit determination in which the TLEs (and derived osculating elements) are used as pseudo-observations.
This paper presents the approach adopted to process the TLEs to improve the accuracy of re-entry prediction. This processing is based on methods previously employed to detect space weather events, which slide a window through the orbital elements contained in the TLEs or derived quantities. Details of the algorithm, which enables TLEs of varying quality and generated in different phases of re-entry to be analysed using the same method, are given. Then, the method used to distinguish between space events and outlying TLEs is described. The trade-off between the number of false positives and negatives, i.e. incorrectly identified and missed outliers, is emphasised. The results of applying the TLE processing methodology to several example rocket bodies are presented in detail and discussed in the context of the accuracy of the resulting re-entry prediction.

Debris, Safety and Awareness (I) / 30

Innovative Method for the Computation of Safety Re-entry Area Based on the Probability of Uncertainities in the Input Parameters

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The risk reduction measures required for the reentry of a spacecraft at its end of life are regulated in Europe by requirements documented in Space Agencies' instructions and guidelines. In particular, according to the French law, "the operator responsible of a spacecraft controlled reentry shall identify and compute the impact zones of the spacecraft and its fragments for all controlled reentry on the Earth with a probability respectively of 99% and 99,999% taking into account the uncertainties associated to the parameters of the reentry trajectories". According to European Space Agency guidelines the Safety Re-entry Area (SRA) delimits the area where the debris should be enclosed with a probability of 99,999%. The computation of SRA is required for a significant number of space missions like spacecraft in low Earth orbits at its end of life and last stages of launchers that shall be controlled to a destructive reentry. This information is crucial to get safety clearance. A similar box, relative to smaller probability level (99%), is required to implement the procedures of warning and alerting the maritime and aeronautic traffic authorities of the concerned countries.
The dynamics of a space object like a spacecraft or a rocket stage entering the atmosphere is quite complicated and quite sensitive to a number of parameters linked to the fragmentation process, to the trajectory models and to the initial conditions of the arc hitting the atmosphere. The computation of the SRA must take into account the uncertainties of those parameters with a satisfactory accuracy or sufficient level of conservatism to ensure that debris will not fall outside the SRA with a probability larger than 0.001%. There are two main modeling aspects for the computation of the SRA: 1) the characterization of the fragmentation and explosion process and of the properties of surviving fragments 2) the computation of the impact points from the trajectory propagation taking into account dynamics model and initial conditions uncertainties. The problem is complicated due to the extremely low probability of interest, which makes quite difficult and inaccurate to use classic statistical techniques and requires to rely on specific extrapolation of the results by fitting distributions tails. A Montecarlo analysis may be performed to estimate the footprint of fragments impacts. The limitation of this method is the number of dispersed fragments impact points to be simulated by the Montecarlo analysis in order to measure the size of the footprint box associated to a low probability (e.g. 0.001%) of occurrence. The number of samples generated as output of the Montecarlo is constrained by computational time and particular statistical tools are used to estimate the quantile of interests when the number of outputs samples are smaller than required.
This paper describes an innovative method to compute the SRA, considering that the input models and its uncertainties are well defined. The method focuses on the statistical distribution of the uncertainties of necessary input parameters contrary to classical methods that generates a large number of impact points with Montecarlo simulation and processes the outputs of this computation.
As a different approach, the innovative method described in this paper processes only sets of the input dispersions associated to a given probability and, consequently, does not require generating Montecarlo simulations and processing the statistics of the output. The probability of interest is computed integrating the multivariate density function of the input parameters and, then, an optimization process is used to find the output worst case among a reduced set of inputs corresponding to a given probability. Three advantages of extreme importance can be recognized: the probability is computed constraining the input dispersions, that are directly associated to the causes driving the phenomena; a large amount of computational time is saved since a Montecarlo simulation is not required; the level of probability can be arbitrarily small because the computational time is not quite sensitive to the probability level to be achieved. The drawback of the method is related to the simplification that is introduced in the computation of the overall input probability. This simplification leads, in some cases, to an overestimation of the size of the box providing a conservative solution to the problem.
The method is applied to an example of the SRA computation for the destructive shallow re-entry of a large space vehicle in the South Pacific Ocean. When a vehicle performs a shallow re-entry, it travels on a final orbit whose perigee radius is larger than the Earth radius and it impacts the atmosphere with a flight path angle shallower than usual. Consequently, the spacecraft is exposed for a long period to particular aero-thermo-dynamic conditions, which enlarge significantly the dispersion of the ground impact area of the surviving fragments and makes particularly challenging

the estimation of the SRA.
The paper presents the results of the computation of the shallow re-entry SRA using the classical Montecarlo approach and this innovative approach. The results are compared highlighting advantages and drawbacks in terms of accuracy, level of conservatism and computational time.
The method described in this paper is suitable for many future applications taking advantage of its computational speed and reliability: the destructive controlled re-entry of large structures, including in particular the International Space Station (ISS) and the ISS visiting vehicles at its End of Life (EoL); the destructive re-entry of large uncooperative satellites orbiting LEO and MEO as conclusive event of the Active Debris Removal (ADR) technology; the destructive controlled re-entry of last stages of launchers.

Rendezvous & Docking (II) / 37

Asteroid proximity GNC assessment through High-fidelity Asteroid Deflection Evaluation Software (HADES)

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This work presents the general architecture and capabilities of the HADES software developed at Deimos Space SLU. Detailed models about the close proximity environment about Near Earth asteroids and the involved operations are often required during preliminary assessment of mission requirements especially under the presence of uncertainties. It is vital to assess the compliance to the mission requirements in terms of safety and illumination. These play an important role in the selection of control techniques and operational orbits. The developed software deals with the high-fidelity modelling of spacecraft operations at irregular shape asteroids. The first version of HADES which includes the main GNC functionalities has been developed and tested. The spacecraft dynamics considers all the possible perturbations, i.e. third body effect from the Sun, the SRP and irregular gravity field of the rotating asteroid. The software uses both spherical harmonics and actual asteroid's shapes. In the first case the coefficients can be given from actual data or they are calculated on the size of user-defined ellipsoid; in the second case the gravity field is reconstructed from the asteroid tetrahedral mesh. The software can handle any operational orbit, with particular care paid to inertial and body fixed hovering. One important aspect when designing proximity operations is to evaluate how the different control techniques and on-board instruments affect the performance of the system. Different control techniques based on both continuous and discrete methods have been considered and implemented. The manoeuvre execution itself can be affected by errors in the magnitude and in direction. The spacecraft orbit determination is performed through a performance models or by on-board measurements, a navigation camera and a LIDAR, which are processed by an Unscented H-infinity Filter (UHF). The latter was selected for its ability to deal with unfiltered biases and non-Gaussian distribution of the measurements. The visibility and illumination condition are considered for the image processing, with the measurements affected by the attitude and pointing errors. HADES can employ different levels of accuracy between the assumed environment knowledge and the model used in the controller and in the UHF. For instance the gravity could be modelled from the shape, but the correction manoeuvre and the trajectory estimate could be calculated by a limited number of spherical harmonics. Also the shape model could be known with a certain error. HADES comes with a Monte Carlo (MC) module which allows drawing more noticeable statistical parameters, such as the control budget, accuracy of the estimation and control systems, or the occurrence of failures when the controller cannot maintain the orbit. Different examples applied to the case of the asteroid Dydimos will be shown for inertial and body-fixed hovering, with some MC analyses for the station keeping of the Asteroid Impact Mission (AIM) for different levels of system and dynamics uncertainties. Finally the ongoing activity on the asteroid deflection by low push methods, i.e. laser ablation, ion-beam shepherd and gravity tractor will be shown. The present tool was developed within the European Commission funded Stardust project under the Marie Curie scheme.

Environment Modelling / 36

Application of the Attitude Analysis of Dynamics and Disturbances Tool in EUMETSAT's study on thruster's allocation and momentum management for meteorological spacecrafts

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The AADDTool - Analysis of Attitude Disturbances and Dynamics Tool – was developed to analyse the disturbances impacts on the dynamics of the spacecraft, start tracker blinding, wheels momentum unloading schemes, and solar power supply, for long term missions in meteorological missions for EUMETSAT. The configuration and setup of the mission is supported by a dedicated graphical interface, for fast configuration and seamless interaction with the simulator.
The paper presents its main features: it combines tiled 3D models for a cylindrical spacecraft (e.g. MSG) or any spacecraft with a central body and solar panels (e.g. MTG and METOP), with accurate models of space environment torques in line with ECSS standard. These were validated with independent tools or available flight/sensor data, and their tile-by-tile analysis of the disturbances allows the user to refine the contributions and take into consideration better approximations for shadowing. The implementation relies on developed libraries implemented in Matlab/Simulink, with a modular architecture to enable a modular design and progressive sophistication of the tool. All modules were validated unitarily using dedicated test setups with reference data (from independent tools, ground and flight data).
Foreseen continuous upgrading activity brought into AADDTool attitude dynamics propagation, spun guidance schemes, guidance programming, and the implementation of an elliptical field-of-view for the Star-Tracker analysis. Additional off-loading schemes were added for performance assessment of improved laws for wheels momentum management such as maximization of time between off-loading or scheduled off-loading with defined set points of the wheels speeds (or momenta).
For closed loop analysis and study of thrust firing parameterisation, the control loop MetOp was also implemented, including a pulse frequency modulator, dedicated to the thrusters' triggering management where the required torque is achieved by modulating the inhibition duration following a period of constant actuation. Before the study, implementations in the tool of two of the MetOp safe modes were successfully validated in terms of propellant mass and number of firing pulses, against simulated telemetry from the real flight software for Sun Safe Mode, and against data coming from LEOP telemetry (flight data) for Earth Safe mode.
The tool has been used for cases studies in LEO and GEO, to size disturbances and momentum management. The upgraded off-loading schemes have been compared with previous results in terms of offloading frequency and propellant consumption. Some results are presented.
The most recent study was for the MetOp scenario: analyse the impact of re-tuning the default thrusters grouping numbers according to mission phases or scenarios for improvement of propellant consumption, varying the lifecycle, solar activity, and initial conditions. The study used the on-board closed loop for two operating controlled modes: Earth pointing (FAM2) and Sun pointing (PRO). For this study, the MetOp spacecraft was modelled with a tiled 3D-model using a parallelepiped for main body and rotating solar array, used to perform a tile-by-tile contribution of the disturbance torques in LEO. The impact of the closed loop settings were analysed, and the main conclusions are summarized in terms of overall mass consumption, and firing and attitude histories.

Rendezvous & Docking (I) / 35

JOSCAR/JDRAGON: Tools for Maneuver Strategy Computation Developed in Java and using PATRIUS

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JOSCAR/JDRAGON are both new tools of maneuvers strategies computation, developed internally in CNES (Centre National d'Etudes Spatiales, French Space Agency) at the Orbital Maneuvers Office (DCT/SB/MO).
Thus, thanks to the decision, some years ago, to use Java technology, existing Fortran tools and libraries are going to be rewritten specially in order to use the powerful CNES PATRIUS library. In that frame, JOSCAR/JDRAGON tools have been rewritten in Java even if they are always based on the same basic principles of the initial OSCAR/DRAGON Fortran versions which were intensively used for the Automated Transfer Vehicle (ATV) and still today, for the operational design of the LEOP, phasing and rendezvous scenarios for GALILEO missions. This paper will describe the methods implemented as well as the software functionalities, pointing out the differences between JAVA versus FORTRAN version, the first one taking advantage of some PATRIUS new functionalities as well as almost 20 years usage feedback.
Thus, JDRAGON is able of computing a near-optimal mission plan, using initial conditions for target and chaser spacecraft's, an amount of maneuvers to be optimized respecting some constraints of application and rendezvous conditions. It is based on a robust and fast method, which requires calling a numerical propagator iteratively. For this purpose, JPSIMU has been also developed based on its PSIMU predecessor which is the heart of numerous CNES flight dynamics tools (as in ATV-CC or GALILEO FDS ones). At a higher level, JOSCAR, which uses JDRAGON as a kernel, allows to perform End-to-End Monte-Carlo simulations, necessary to test robustness of the computed strategies for mission analysis purpose.
Above the new design of these tools, thanks to the Java object approach, their validation is also a big challenge. Thematic validations have been performed with not easy comparisons with the Fortran version, given the differences in their corresponding flight dynamics libraries. Special efforts have been put into performance, looking for optimal tools settings, aiming at having both fast computations and satisfactory accurate results. Concerning the quality of the code, Eclipse environment analysis tools have been used in order to be compliant with CNES coding standard rules.
At last, in order to deal with the considerable input/output data generated, a Graphical User Interface has been also developed using GENIUS (a higher level CNES JAVA toolkit based on Swing) which allows using these tools in a more friendly way on many different Operating systems from Windows to Linux.

Optimization and Dynamics (I) / 34

Analytical Approximation for the Multiple Revolution Lambert's Problem

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An approximate analytical solution of the multiple revolution Lambert's problem is presented. The solution is obtained based on a variation of parameters approach and offers remarkable accuracy near the minimum energy condition. Consequently, the method is useful for rapidly obtaining low delta-V solutions for interplanetary trajectory optimization. In addition, the method can be employed to provide a first guess solution for enhancing the convergence speed of an accurate numerical Lambert solver.

Re-entry and Aero-assisted Maneuvers / 60

PETbox: Flight Qualified Tools for Atmospheric Flight

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The Planetary Entry Toolbox (PETbox) is a set of multiple modules developed by DEIMOS Space S.L.U. to support Mission Engineering and Flight Mechanics in the area of Atmospheric Flight.
PETbox has been intensively and successfully used in multiple ESA projects, EU projects and private initiatives covering a very wide range of vehicles (launchers, lifting bodies, capsules, UAVs, winged bodies, hypersonic transport vehicles, space debris. . .) in multiple environments (Earth, Mars, Titan) and in multiple flight phases (launch, coasting, entry, descent, landing, sustained flight). The set of modules that composes PETbox allows a critical range of multiple analyses, a full "Mission Engineering process" that supports engineers at different levels, from Pre-Phase A studies to Post Flight Analyses. The core module of PETbox is endosim (endoatmospheric simulator) which is the simulation framework used by the Atmospheric Flight Competence Center (AFCC) of DEIMOS Space. The toolbox is live and continuously evolving according to the improvements and modifications implemented daily and that currently integrates more than 50 years of engineering work of the AFCC team of DEIMOS Space.
The applications range is wide, covering vehicle design (shape design, configuration design, system specifications, MDO. . .), aerothermodynamics (computations, inspection, analyses, support to databases refinements. . .), flying qualities (trim, stability, controllability, GNC specifications, . . .), trajectories (modeling, end to end simulation and optimization, analyses, flight predictions. . .), guidance (design, prototypes, functional validation, . . .), sizing conditions (performance and margins verification, specifications for system and subsystems, correlations analyses, . . .), safety aspects (nominal and off-nominal footprints, survivability and risk analysis of debris, separation analyses. . .), visibility aspects (with fixed or mobile ground stations, with GPS, between spacecrafts. . .), post flight analyses (trajectory reconstruction, data fusion, analyses, . . .), etc.
Practical example of use and key applications in multiple projects will be presented with special emphasis on the use of PETbox in the current ExoMars program (2016 and 2018 missions) and in the recent Intermediate eXperimental Vehicle (IXV) that successfully flew on February 11th, 2015. IXV has represented a unique opportunity to increase the TRL level not only of re-entry technologies but it also marked a key milestone in the overall validation of the design methodology and tools implemented in the areas of Mission Analysis and Flight Mechanics; it confirmed the robustness of the approach and the maturity of PETbox which is now Flight Qualified and ready for future challenges in the European re-entry technology roadmap.

Verification and Validation Methods / 61

Dynamic Test Facilities as Ultimate Ground Validation Step for Space Robotics and GNC Systems

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Ground testing of space technologies, and in particular space robotics and Guidance Navigation and Control (GNC) ones, is crucial in order to de-risk space missions that heavily rely on them. In this context, this paper describes the role of the platform-art© robotic facility located at GMV premises in Tres Cantos (Madrid, Spain) as last on-ground validation step within the development and validation of a whole range of different space technologies.

The platform-art© hardware in-the-loop facility can simulate the dynamics of any satellite system: the paper will present how the facility has been used in a number of activities related to a different space scenarios such as Lunar descent and landing, on-orbit servicing (including contact), active debris removal (e.g. ANDROID mission) and Rendez-Vous (e.g. Mars Sample Return capture mission). The modularity of the facility will also be described, which allows for fast re-adaptation of the setup for different scenarios, also enabled by the different mock-ups that are available at platform-art© (with different scale factors and fabrication precision depending on the specific test needs. Lessons learnt and graphical results from different ESA projects that have used the platform-art© hardware in-the-loop facility.
It will also be presented, complementary to the dynamic test facility, an integrated Design, Development, Verification and Validation environment covering from Model-in-the-Loop (e.g. Matlab/Simulink algo-rithms) till real-time Processor-in-the-Loop (with autocoded or hand-made GNC produced SW) and later on extended through the use of dynamic test facilities with real dynamic and air-to-air HW-in-the-Loop (sensors, robotic manipulators, . . .) stimulation.

Interplanetary Flight and Non-Earth Orbits (I) / 62

NEO Threat Mitigation Software Tools

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Nowadays, there are a number of institutions worldwide that contribute to the discovery, tracking, identification, cataloguing and risk characterisation of asteroids in general, and NEOs in particular. However, there is no currently an integrated set of tools that cover, in a complete manner, the assessment of the impact risk mitigation actions that can be taken to prevent the impact of a NEO on Earth and to allow helping the dimensioning of space missions to address such problem. In that context the following set of utilities have been developed within the European Commission funded NEOShield project to allow covering the abovementioned activities: - NEO Impact Risk Assessment Tool (NIRAT) - NEO Deflection Evaluation Tool (NEODET) - Risk Mitigation Strategies Evaluation Tool (RIMISET).
NIRAT, the first tool, allows evaluating, for possible impactors, the projection of the b-plane dispersion at the dates of possible impact and also the presence of keyholes that would enable future collision opportunities. This tool allows characterizing the impact probability for the different opportunities and, together with the knowledge of the asteroid features, the evaluation of the risk. This tool resembles current performances achieved by NEODyS and Sentry, but does not intend to represent the same level of accuracy in the obtained results. The services provided by this tool are required by the next other tools.
The second tool, NEODET, allows assessing the required optimal change in asteroid velocity (modulus and direction) at any given instant prior to the possible impact epoch that would allow shifting the dispersion ellipse out of the contact with the Earth. This would represent the effect of impulsive mitigation options (one or several impacts). It also allows evaluating the accumulated effect that slow-push techniques (e.g. gravity tractor) would impose on the asteroid orbit to achieve optimal deflection by those other means.
Finally, the RIMISET tool allows evaluating how some of the most relevant impulsive and slow-push mitigation techniques would meet the required changes in asteroid state to obtain the searched for deflection and the requirements that this could impose on the design of the mitigation mission. Following mitigation methods are included: explosive, kinetic impact, gravity tractor and ion beam shepherd. Each technological solution is simulated to allow ascertaining the efficiency in achieving the deflection goal by any of the proposed means (impact, explosive, gravity tractor and possible combinations of those). Ultimately, it shall serve to dimension the required mitigation space systems and solutions.
Validation cases have been executed over the now no-threat cases of asteroids 2011 AG5 and 2007 VK184.

Multidisciplinary Design Optimization / 63

Trajectory and Systems Design for Low-Thrust Interplanetary Missions via Multi-Objective Hybrid Optimal Control

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Preliminary design of interplanetary missions is a highly complex process. The designer must choose the launch date, flight time, propulsive maneuvers, and possibly a sequence of planetary flybys as well as altitudes and velocity vectors for each of those flybys. Low-thrust missions add an additional degree of complexity because the designer must also choose a time history of control variables, i.e. thrust magnitude and direction, which define the trajectory. For some types of missions, such as missions to asteroids and comets, the designer is also responsible for choosing the destination because the customer is interested in a population of bodies rather than a specific body. In addition, low-thrust trajectory design is tightly coupled with spacecraft systems design because the propulsion and power capabilities of the spacecraft strongly drive the available trajectory options. Therefore, the choice of system parameters is an integral part of the low-thrust preliminary design problem. Furthermore, a mission designer does not work in isolation - the customer who commissions the work, usually a scientist, does not just want a point solution even if that solution is globally optimal in propellant use, time, or some other metric. Rather, an exploration of the trade space between several metrics of the customer's choice is the true goal of preliminary mission design.
The method of multi-objective hybrid optimal control (MOHOCP) will be presented in this work. In MOHOCP, the mission design problem is separated into two nested loops – an "outer-loop" which chooses the sequence of bodies to be visited and also the systems parameters of the spacecraft, and an "inner-loop" which searches for the globally optimal trajectory corresponding to each candidate outer-loop solution. A "cap and optimize" approach is used where one of the outer-loop's objective functions becomes the objective of the inner-loop solver and the other outer-loop objective functions are represented in the inner-loop as constraints or problem assumptions. In this work a version of Nondominated Sorting Genetic Algorithm II (NSGA-II) is used as the multi-objective outer-loop solver and Monotonic Basin Hopping (MBH) coupled with a nonlinear programming (NLP) solver is used to solve the inner-loop.
The algorithm presented here is the core of NASA Goddard's Evolutionary Mission Trajectory Generator (EMTG), an open-source tool for preliminary design of interplanetary missions. In this work the MOHOCP algorithm will be described in detail and a worked example of a combined interplanetary trajectory and systems design problem will be presented.

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Techniques for assessing space object cataloguing performance during design of surveillance systems

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In order to guarantee safe operation of satellites, space object catalogues must be build-up and maintained. The catalogues should be complete, i.e. contain sufficiently accurate and frequently updated orbital states for all required objects. In theory, completeness of the catalogue is achieved by designing the radar in a way that a major fraction of the object population is considered detectable, i.e. covered by the sensor's field-of-regard and within the sensor sensitivity. However, complete coverage does not necessarily guarantee a proper catalogue build-up, yet. If an object is observed once, it must be re-observed in order to verify its existence and improve the accuracy

of the determined state. In a next step, individual observation tracks are combined with each other to further improve the accuracy. Consequently, tracks must be associated to each other, i.e. tested if they originate from the same object or not.
The success rate of the association is dependent on the quality of the tracks, the re-observation time and the re-observation geometry. For surveillance radars, the association performance must be considered as a critical design parameter and can be optimized along with the detection rate during the design process. We outline the underlying techniques and present a simulation-based framework for assessing the surveillance system design in terms of association performance and achievable accuracy.
In the tool framework phased-array radars are defined with a detection figure-of-merit (i.e. ratio of detectable object size at a certain distance), a field-of-view, a pointing direction, and a noise estimate. Then, observations are generated with a realistic object population, e.g., based on ESA's MASTER model. The resulting tracks are associated to each other using covariance-based distance metrics. We address several difficulties which arise during the association, e.g. proper treatment of state uncertainties and robust initial orbit determination. The association performance is analysed for different orbital heights and re-observation conditions for a specific radar design concept. Additionally, the typical resulting orbital state accuracies are presented for the initial orbits as well as for the improved ones.

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Applicability of COBRA concept to de-tumbling space debris objects

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COBRA is a contactless concept for de-tumbling and controlling the attitude of a target space debris object that exploits the torques generated on the target by the plume impingement of a thruster facing the target to impart torques on the target. The control strategy for de-tumbling is based on a switching strategy for the de-tumbling thruster and a pointing strategy for aiming the de-tumbling thruster at a specific region of the target. This control strategy has been developed in a previous study of the concept. This article discusses several refinementsfurther developments of the original strategy and examines the general applicability of the COBRA concept, mainly in Active Debris removal missions in line with Cleanspace. The applicability of COBRA concept is investigated by examining several scenarios, namely, de-tumbling and attitude control of debris objects of varying configurations in terms of object geometry and mass parameters and in various initial rotation states. The main targets examined in this study are Envisat and Cryosat PROBA 2.
Simulation results described in a previous article have shown that the simple pointing and switching strategy for COBRA can successfully de-tumble a large space object such as Envisat in a relatively short time and using only a modest amount of ΔV, namely from an initial rotation rate of 5°/s to 0.5°/s in under one orbit. These simulations assumed a relatively favourable rotation state of Envisat, which allowed pointing the thruster at the Solar panel. The normal of the Solar panel was perpendicular to the rotation axis, such that a large torque could be generated.
In the current article, new simulations will be presented for less favourable rotation states of Envisat. In addition simulations will be presented for a different target, PROBA 2. PROBA 2 is a roughly cube-shaped object, which means that there is no particular geometry (location plus orientation) of the thruster with respect to target that generates high torques. Previous results have indicated that the ΔV required ΔV required could be improved, in part by updating the thruster layout of the chaser and in part by updating the control strategy. The thruster layout is updated to include thrusters in the direction exactly opposite to the de-tumbling thruster to limit cosine losses. The control strategy is improved in a number of ways.
Previous results indicated that the ΔV overhead (that is, the ΔV not directly contributing to de-tumbling the debris) was higher than expected based on the ΔV required to compensate for the activation of the de-tumbling thruster. A part of the overhead can be explained by the cosine

losses, but a substantial fraction was due to attitude control. The pointing strategy and the attitude control of the chaser are improved to reduce this overhead. In addition some effort is spent in improving the pointing strategy of the chaser. The current strategy uses a very simple model to predict the torques imparted on the target. The prediction model is improved such that the torque imparted on the target is closer to the desired torque.

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Monte Carlo Simulation of a Triple Flyby Capture at Jupiter Using Paramat

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In 2014, Thinking Systems began work on a parallel processing tool that incorporates the numerical engine from the General Mission Analysis Tool (GMAT) into a system, Paramat, designed to use the processing capabilities of modern, multi-core computer platforms. This paper opens with a brief description of recent changes to Paramat. Paramat is then used to model an orbital capture at Jupiter that uses gravity assists at Callisto, Io, and Ganymede to reduce the orbital insertion costs for the capture. That mission segment is presented as a baseline trajectory for Monte Carlo analysis of the costs of the insertion sequence. Perturbations are applied to the nominal capture trajectory and to parameters related to the course correction maneuvers in order evaluate the maneuver contingency costs for the capture. Analysis of these data provide insight into the total delta-V costs and margins for the capture phase of the mission, along with an estimate of the orbit determination requirements for each phase of the capture trajectory.

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ATHENA: Astrodynamics Toolbox for High-fidelity Error-propagation and Navigation

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This work presents the general architecture and capabilities of the ATHENA software package. ATHENA is a toolbox for uncertainty propagation, gudance, navigation and control of single and multiple spacecraft with distributed architecture. ATHENA implements advanced state estimation and filtering techniques and a high fidelity dynamical model. ATHENA allows for the analysis of the coupled orbital and attitude dynamics during close proximity operations and docking with cooperative and non cooperative targets.
The toolbox implements high fidelity models for both asteroids and Earth orbiting objects. The dynamic model includes third body effects, solar radiation pressure and the irregular gravity field of Earth, Moon and asteroids. The toolbox includes spherical harmonics gravity models, tetrahedral models from radar observations, and finite volume models. In the first case, the harmonic coefficients are from actual data or are calculated from the inertia matrix of a user-defined ellipsoid; in the last case, the gravity field is reconstructed from point masses. Given an initial distribution of point masses of arbitrary value contained within the original shape, their position and mass is optimised in order to match the original centre of mass and moments of inertia.
ATHENA allows for the design and analysis of multi-sensor fault tollerant autonomous navigation systems with sensor fusion across multiple platforms with resource sharing. The paper will present an example of autonomous navigation of a formation of spacecraft flying in the proximity of an asteroid. Multiple measurements collected by on-board cameras, attitude sensors and

LIDAR are data fused to estimate the state of each spacecraft with respect to the asteroid. Inter-spacecraft links are used to combine measurements and improve resilience and accuracy. It will be shown that different combinations of measurements can be constructed to improve the navigation performance. The overall effect is a system more robust in the presence of failures.
ATHENA includes models of binary systems and the coupled orbital and rotational motion of the two bodies around their centre of mass. The paper will present and example of navigation and control of a single spacecraft placed at a stable libration point.
The toolbox can simulate close proximity operations and autonomous rendezvous and docking (R\&D) with non-coopeative targets on elliptical orbits. In particular, ATHENA can generate optimal scheduled plans and control profiles to rendezvous with multiple targets or to dock multiple spacecraft with a single target. This capability will be shown within an orbit servicing scenario. In this demonstration scenario, the target spacecraft is formed by a set of active payload modules (APM) connected to the spacecraft bus via a standardized interface. ATHENA generates a plan and corresponding guidance for a swarm of small satellites that collaborative (as a colony) re-configure the target spacecraft by removing and replacing APMs.

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Debris cloud analytical propagation for a space environmental index

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Environmental indices for spacecraft are meant to rank objects in orbit depending on their effect on the space environment in case of fragmentation. These indices are usually designed to identify potential candidates for active debris removal and several authors have proposed formulations of environmental indices, which take into account different aspect of the debris environment (e.g. background density, spacecraft mass, orbital altitude). The index proposed in this work focusses on the assessment of the severity of the breakup of a spacecraft estimating the resulting collision probability for operational spacecraft.
First, a grid in semi-major axis and inclination is defined to map the possible initial conditions of the breakup. For each point in the grid, the NASA breakup model is applied to generate the fragment cloud (considering all objects larger than 1 cm). The case of catastrophic collision is considered to model the effect of the spacecraft mass. The spatial density of the cloud is analytically propagated for 25 years by applying the continuity equation to model the effect of atmospheric drag. An analytical approach is also used to compute the collision probability between the fragments in the cloud and target spacecraft crossing it. In this application, the target spacecraft represent the fleet of operational satellites on which the consequences of the breakup are evaluated. For computational reasons, not all operational satellites are individually propagated, but their population is sampled, considering the distribution of cross-sectional area, to define a small number (e.g. 10) of representative objects. The value of the environmental index for a specific initial condition is obtained by summing the resulting cumulative collision probability of all representative objects.
In this way, the variation of the environmental index with semi-major axis and inclination is obtained and stored in the so-called "reference layer", which refers to a fixed value of fragmenting mass. The dependence on the mass is introduced by rescaling the "reference layer" following the power law that, in the NASA breakup model, relates the number of produced fragments and the fragmenting mass. One interesting aspect of the proposed index is that the computational effort is required only for the generation of the "reference layer", which needs to be recomputed only in case of a significant variation in the distribution of active satellites. Given the "reference layer" and the dependence on the mass, the value of the index for any spacecraft can be obtained by a simple interpolation.
Only a few key parameters (i.e. mass, semi-major axis, inclination) are required to characterise the criticality of a mission. The index can be applied to spacecraft already in orbit (as it was

done using the data available in DISCOS) and to future missions, to support their mission design and their licensing process. For example, the environmental index can be employed to assess possible waivers to mitigation requirements for spacecraft, considering the estimated severity of their breakup.

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SNAPPShot: Suite for the Numerical Analysis of Planetary Protection

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When interplanetary missions depart from the Earth, the rocket bodies used for their launch may be inserted into a resonant orbit with the Earth or into trajectories that may cross other orbits. A suite of numerical tools has been developed as part of a European Space Agency contract to study the compliance of the launcher with planetary protection requirements.
For each mission, the launcher injection dispersion, the uncertainty in the spacecraft physical parameters, or potential spacecraft failures are treated with a Monte Carlo (MC) analysis. The uncertainty in the velocity vector components with respect to the nominal manoeuvre is defined and the distribution of velocity is sampled to obtain a discrete number of initial conditions. If required, the dispersion of other parameters (such as the area-to-mass ratio) can be included. Each initial condition is propagated through a high accuracy propagator, which describes the evolution of the launcher trajectory in Cartesian coordinates with respect to the solar system barycentre. Once each trajectory is computed, it is analysed to detect if it enters a planet's sphere of influence. If so, the b-plane representation is used to characterise the close encounter.
The suite includes several Runge-Kutta propagators (both with adaptive step size strategy and regularised step size), providing also the possibility of dense output and the inclusion of user-provided events in the propagation (e.g. the propagation can be stopped if an impact is registered). Different options in the ephemeris routines (including the NASA SPICE toolkit) are available, which were validated against the data on the JPL Horizon system. In addition, SNAPPShot provides a tool for the analysis of the b-plane to distinguish between conditions of impact, gravitational focussing, and resonances. Fly-bys of different bodies and subsequent close encounters can be handled. The tool can identify the most critical close encounter, which will be used for the statistics generated from the MC analysis. The number of required MC runs to validate the requirements with a given confidence level is automatically estimated Wilson confidence intervals. If impacts are detected, the number of MC runs is automatically incremented. All codes in the suite have been written in Fortran 90, exploiting the advantages of modern Fortran, such as array operations, dynamic memory allocation, modules and procedure pointers. The possibility of using parallel computations in the MC analysis will be discussed. As an example, an analysis for planetary protection compliance verification for the launcher of the BepiColombo mission requires four hours of computational time on a PC to run a MC analysis with more than 50000 runs.

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From end-of-life to impact on ground: An overview of ESA's tools and techniques to predicted re-entries from the operational orbit down to the Earth's surface

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Currently, there are about 17 000 tracked objects in Earth orbit, out of which approximately 7500 are expected to have a remaining orbital lifetime of less than 100 years. Out of those 7500, about 1250 have a mass of more than 1 kg, but the vast majority are smaller pieces of space debris.
Once an object in Earth orbit has reached the end of its operational life, or in case of a space debris object after its genesis, it enters into the re-entry prediction system of ESA's Space Debris Office. This system automatically predicts the remaining orbit lifetime; In case of a short remaining orbit lifetime it automatically predicts the impact location, risk and associated uncertainties; In case of a high risk re-entry event it enables the in-depth analysis of the affected regions and atmospheric break-up of the object; And tools are available for post-event processing of the observational data. The results of this analysis chain are provided to the relevant actors, e.g. national alert centres or operators, either automatically or on-demand.
In this paper we present the status of the re-entry prediction system, and its orbit determination, orbit propagation, environment forecasting, and risk assessment methodologies related to the orbital lifetime, re-entry location, and atmospheric break-up predictions. Uncertainties depending on the orbit regions, object type, and step in the re-entry prediction system are derived and used to tune the service to provide the best possible results over the entire population. In the last step of the system, these automatically generated results are complemented with an operator review of the available data to provide re-entry and break-up prediction for individual objects, occasionally complemented by processing dedicated observations of the re-entry object. Post-mortem analyses, e.g. after a confirmed re-entry, are performed for selected objects in order to explain potential observations of the re-entry event and retrieved samples on-ground. The entire data collection is a relevant source which serves as input for break-up modelling tools, and improvement of the entire prediction service.

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Robust Control Design Methodology of the 6DoF Flight-Formation of PROBA-3

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PROBA-3 is one of ESA's technology demonstrators, which consists of two spacecraft flying in close formation in a highly elliptic orbit. Its aim is to create a Sun coronagraph instrument with the optical payload and the external occulter disc mounted on different spacecraft. One of its main challenges is to develop the GNC that maintains a flight-formation during the apogee arc with very stringent attitude and relative position requirements. The flight-formation consists of aligning both spacecraft with the Sun vector, while maintaining a fixed distance with sub-millimetre precision. In addition, various technology demonstration manoeuvres will be performed for virtual structure simulation (rotating the formation, while maintaining a fixed distance), and virtual telescope focusing (resizing the formation, while keeping the Sun vector aligned).
The control problem has been posed as a coupled 6DoF multi-input multi-output (MIMO) design task, which simultaneously takes care of both the attitude and the position. To meet the design specifications, state-of-the-art H_∞- and μ-synthesis techniques have been utilized to obtain a robust controller. In addition, in-house developed simulation tools have been employed for validation purposes in a non-linear environment.

In this paper the step-by-step design techniques and accompanying tools used by SENER are presented, starting with the formulation of the control problem, working towards the justification of the designed controller, and resulting in a fully validated product both in frequency and time domain for integration with the full PROBA-3 system.

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The Pointing Error Engineering Tool (PEET): From Prototype to Release Version

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Recent standardization efforts in Europe have led to the publication of the ECSS Control Performance Standard E-ST-60-10C and the ESA Pointing Error Engineering Handbook ESSB-HB-E-003, which are instrumental in defining a clear pointing error engineering methodology for ESA projects.
The necessity for such methodology gains in importance as current and future ESA missions, especially scientific and laser-communication missions, put more and more stringent pointing performance requirement on the spacecraft systems. As a consequence even small performance changes may not only lead to additional iterations of the design, but potentially result in prominent mission-critical changes in subsystems and equipment selection. The rapidly growing system complexity for such kind of missions complicates the situation even more.
Hence a systematic and user-friendly software tool that is capable of automating the pointing performance management process based on the standardized methodology and which replaces the conventional manual computation is largely beneficial and necessary. To complement and support dissemination of this pointing error engineering methodology, a software prototype called Pointing Error Engineering Tool (PEET) was developed and released under the ESA Software Community License.
PEET is designed as an extension to MATLAB and completely runs inside the MATLAB environment to exploit its computational features. It provides a graphical user interface implemented in Java suitable for building pointing systems and analysing pointing errors. The core of the PEET software which contains the mathematical algorithms for error computation is implemented using MATLAB classes.
The prototype software released in the end of 2012 (with several updates until 2015) was restricted to the "simplified statistical method" described in the ESA Pointing Error Engineering Handbook, i.e. results are based on the assumption that the central limit theorem is applicable and the total error follows a (nearly) Gaussian distribution. This assumption may lead to significant deviations from the actual result, in case dominant non-Gaussian error contributions exist.
This paper focuses on the improvements in the ongoing development of the prototype to a release version. The key update supersedes the mentioned restriction on Gaussian distributions and implements the so-called "advanced statistical method" described in the ESA Pointing Error Engineering Handbook. In particular, this implies that probability density functions of (arbitrarily correlated) random variable error contributions - rather than only fundamental statistical moments – are taken into account for an accurate level of confidence evaluation of requirements. The benefits of this approach will be highlighted by comparison of the results obtained with both methods.
In addition, a compliant generalization of the "statistical interpretation" concept in the ECSS Control Performance Standard is introduced which allows more flexibility in the definition of requirements and can help to avoid over-conservative budgets. By supporting script-based execution from MATLAB, the release version of PEET also directly supports its integration in a tool-chain, e.g. for optimization runs or parameter trade-off studies.

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ESA's Asteroid Impact Mission: Mission Analysis and Payload Operations state of the art

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The Asteroid Impact Mission (AIM) is an ESA mission whose goal is the exploration and study of binary asteroid 65803 Didymos, which is expected to transit close to the Earth (less than 0.1 AU) in late 2022. AIM is planned to be the first spacecraft to rendezvous with a binary asteroid: its mission objectives include the highly relevant scientific return of the exploration as well as innovative technological demonstrations. In addition, AIM is part of a joint collaboration with NASA in the AIDA (Asteroid Impact & Deflection Assessment) mission. The primary goal of AIDA is to assess the feasibility of deflecting the heliocentric path of a Near Earth Asteroid (NEA) binary system, by impacting on the surface of the smaller (or secondary) asteroid of the couple. To this aim, AIDA includes the kinetic impactor, DART (Double Asteroid Redirection Test) by NASA and the observer, AIM (Asteroid Impact Mission) by ESA. The work presented in this paper has been performed by the authors under ESA contract within the phase A/B1 design of AIM mission. The paper presents some updates on the ongoing design of the mission. Each phase of the operative life of AIM spacecraft is detailed with information and results on the solutions adopted for Mission Analysis design and on the strategies to suitably operate payloads. The selected interplanetary trajectory is presented, including the available launch window to reach Didymos on time. Suitable transfer solutions are selected based on Δv constraints imposed by the launcher and further requirements imposed by spacecraft design. More in detail, AIM is planned to be launched in late 2020 and to arrive at Didymos system in middle 2022. As the spacecraft approaches the asteroid system, it will go through far- and close- approaching maneuvers. The far-approaching maneuver is presented in detail: the final Δv to stop AIM at Didymos is split into five smaller maneuvers, performed at one week distance between each other, to decrease the overall maneuver cost and to allow for precise tracking and rendezvous with the binary system. Close proximity operations at the asteroid are then described. During this phase, AIM mission analysis is driven mainly by observational requirements coming from scientific payload on board, to study the asteroid system before and after DART impact (expected for late 2022), such to accomplish mission objectives. Observation stations are selected for AIM spacecraft to study Didymos by operating scientific payloads. Close proximity operations include the release of a lander on the surface of the smaller asteroid of the couple (secondary) and the release of a network of cubesat opportunity payloads (COPINS). The deployment strategies are described from the operational and maneuvering point of view. In addition, payload operations include technological demonstration of deep space laser communications, low frequency radar tomography of the smaller asteroid of the couple (called Didymoon) and high frequency radar subsurface investigation. The Earth-Spacecraft-Sun-Asteroid geometry is presented into detail during all mission phases. The paper includes the analysis of coverage and illumination conditions during all phases of the missions, to provide inputs to the planning of scientific payload operations and ground segment operations. Both AIM ground and asteroid coverage is analyzed. Constraints imposed by natural illumination of the asteroids are highlighted, to identify poles visibility and to assess visible latitude bands, during the mission time line and payload operations at asteroid. The results and analyses presented here are part of the phase A design of the AIM spacecraft. The project is currently ongoing and the mission analysis will be further iterated and refined through the design phase.

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MP2OC: Multi Phase Multi Purpose Optimal Control Toolbox

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The paper will present a Multi Phase Multi Purpose Optimal Control (MP2OC) code developed at Strathclyde University in the framework of the UKSA National Space Technology Programe (NSTP-2). The grant received for the development of the tool was part of a bigger project aiming at building an integrated design platform for quickly assessing design parameters based on the simulated performance and operations for future space access vehicles. The trajectory module here presented optimises the entire mission, from ascent to insertion into final orbit and from de-orbiting to final landing. The future development of the tool will also take into account model and operational uncertainties and evaluate the robustness of the descent performance against them.
The design of spaceplane mission delivering a payload to a certain orbit and reentering is in itself a multiphase problem. Moreover the atmospheric trajectories, can be further divided in multiple segments. Among segments, the elements defining the problem can differ though disciplinary models (e.g., propulsion modes for a hybrid engine, or in a multi-stage propulsion system), problem objectives and constraints, and level of fidelity needed within the models. To allow such a flexibility in the design, the simulation needs to been structured in multiple phases, with interchangeable disciplinary models. Hence the tool has been designed to allow for each phase, to define the set of models used and all the variables involved in the parameterisation of the controls and propagation of the trajectory. The continuity between phases is guaranteed at convergence of the optimization process by the optimiser and the formulation through constraints functions. Different optimisation algorithms have been included and tested to tackle a variety of problem definitions – from single to multi-objective, deterministic and stochastic, from constrained to unconstrained – and to offer both global exploration capabilities, and local refinement. Fixed and adaptive step size integration techniques and interpolation methods are also included. The platform has been entirely developed in Matlab and it has been interfaced with C disciplinary code for testing purposes. The tool has been designed in such a way that can be easily extended in the future to integrate different optimization/integration techniques.
Finally the possibility of approximating disciplinary models through Kriging metamodeling technique has also been integrated in the platform. A preprocessing is performed before optimisation to evaluate the model in a number of sampling points and build the surrogate The use of surrogate is particularly useful when expensive disciplinary models are involved in the design. The possibility of developing an automatic procedure for multifidelity optimisation is part of the future development for the tool.
The toolbox capabilities and architecture will be presented together with a real application where the trajectory control is optimised, based on different mission objectives and constraints, for the ascent and descent mission segments of a conceptual single stage to orbit vehicle, to a circular low Earth orbits from fixed take-off and landing site.

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Coupling High Fidelity Body Modeling with Non-Keplerian Dynamics to Design AIM-MASCOT-2 Landing Trajectories on Didymos Binary Asteroid

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The Asteroid Impact Mission (AIM) is a mission by ESA, planned to be the first to rendezvous with a binary asteroid. AIM mission objectives includes both scientific investigations and technological demonstrations. The mission is part of the Asteroid Impact & Deflection Assessment (AIDA),

a joint cooperation between ESA and NASA, devoted to assess the effectiveness in deflecting the heliocentric path of a threatening Near Earth Asteroid (NEA) for planetary defense purpose. The target of the mission is near-Earth binary asteroid 65803 Didymos. The goal of AIDA is to study the effects of a kinetic impact on the surface of the smaller secondary asteroid of Didymos couple, informally called Didymoon. To this purpose, together with AIM, the AIDA mission includes DART (Double Asteroid Redirection Test), the kinetic impactor, designed by NASA. The primary objectives of AIM include the detailed study and characterization of the binary couple. Among these, the internal composition of the smaller asteroid will be determined by means of low frequency radar tomography. Similarly to what done with the CONSERT instrument, on board the ESA's Rosetta mission, the radar will include a lander-orbiter architecture to host both transmitters and receivers. Rosetta mission highlighted the challenges of designing close proximity trajectories and to land a probe on the surface of an extremely irregular body such as comet 67P/Churyumov-Gerasimenko, whose shape and mass distribution were completely unknown and unexpected during the mission design phase. In that case, the Philae lander release was challenged by the highly perturbed dynamical environment in the proximity of the comet and its very low and irregular gravity field. In analogy with the Rosetta mission, AIM will deploy a small and passive probe (MASCOT-2, with clear heritage from MASCOT, Hayabusa 2 mission) that will reach the surface of a largely unknown object after a purely ballistic descent. MASCOT-2 lander do not feature any anchoring mechanism and this makes the release even more challenging since Didymos system's gravity field is expected to be weaker, with an escape velocity from Didymoon's surface of about 4-6 cm/s, being the asteroids estimated to be nearly two orders of magnitude less massive than comet 67P/Churyumov-Gerasimenko. In addition, the presence of two gravitational attractors makes the gravity field in the close proximity of the couple highly unstable and chaotic. The paper proposes an effective strategy for MASCOT-2 release, beneficial for the mission analysis and operations design points of view. The AIM scenraio is presented as a perfect case of study, but the methodology applies to any asteroid/small body scenario. In particular, the landing trajectory and dynamics of MASCOT-2 is studied during close-proximity operations using the highest up-to-date fidelity model of Didymos. The paper presents some updates on the work the authors are currently performing during the phase A/B1 design of AIM, under ESA contract, in consortium with OHB System AG, and Spin.Works. From the orbital mechanics point of view, the binary system is naturally modeled as a three-body system and solutions are studied within the frame of the Restricted Three-Body Problem modeling. Shape-based models are used to model the gravitational contribution of the two asteroids refined models are built by combining them to reproduce the gravity field in the proximity of the binary couple. The purpose of the design strategy is to take advantage of the presence of two gravitational attractors to find effective landing solutions. The increased complexity because of the two gravity sources is here read as a potential opportunity to be exploited through the three-body problem modeling, which opens to a variety of dynamical solutions not available whenever a single attractor is dealt with. Three-body solutions are computed for Didymos binary system and suitable trajectories to land MASCOT-2 on the surface of the secondary are selected. More in detail, the motion close to the Lagrangian points is exploited: stable manifolds associated to Halo and Lyapunov orbits, have been propagated in the high-fidelity dynamical environment and suitable solutions are selected to guarantee soft landing on the secondary asteroid. The dynamics of the lander is propagated from release up to rest on the surface of the asteroid. Results show that the extremely low gravity environment does not guarantee the lander to stay on the surface after touch down, but MASCOT-2 will most likely bounce until reaching a stable position of Didymoon. Successful landing probability is assessed for the case of study and landing dispersion is evaluated. Compared to classical Keplerian solutions, three-body dynamics are shown to be effective to lower the risk of rebounding on the surface of the secondary, and to increase the safety of the overall release maneuver to be performed by AIM.

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An RST Design Approach for the Launchers Flight Control System

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The aim of the paper is to present a new design methodology for the automatic flight control system of launch vehicles using discrete-time RST controllers. The structure of such a control system has three degrees of freedom (roll, yaw, pitch), determined such that the closed-loop dynamics of the launch vehicle tracks the output of a desired reference model. The RST control technique focuses on the pitch angle, as the roll and yaw angles are tracking their references much easier.
Although the RST controller operates with input-output discrete time system models (and not with their continuous state representations), integration into a launcher is possible, thanks to continuous-discrete-continuous conversion tools. Thus, the continuous time launcher state space representation of each freedom degree can be first converted into the transfer function (continuous as well). Next, a discretizing technique (such as bilinear) is applied, in order to determine the discrete time transfer function. The RST controller is then designed according to the resulted transfer function and a prescribed second order system, standing for the desired closed-loop behavior. Such a controller is processing the reference trajectory and the actual angle (e.g. pitch) as input signals, to return the necessary command. Finally, the 2 by 1 controller transfer function is converted to the minimal discrete time state space representation, which, on its turn, is brought back to continuous time through some interpolation technique (usually, bilinear as well). The RST design controller relies on the poles placement method and reduces to solving a dyophantine equation in this case, although some other more sophisticated methods could have been considered as well.
The paper is organized as follows: after an introductory part, the second section presents the design models (after linearization) describing the dynamics and kinematics of the launcher, together with the design objectives concerning the control system. The design methodology of the discrete time RST controller is presented in the third section. The theoretical developments are illustrated and analyzed through some numerical examples in section 4. Some concluding remarks and future developments of the proposed approach complete the article.

Debris, Safety and Awareness (III) / 183

Debris de-tumbling & de-orbiting by elastic tether & wave-based control

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We consider the problem of guidance and control of a completely passive, target, piece of debris, using an actively-controlled, "chaser" spacecraft, connected to the debris by an elastic tether. Compared with robotic capture, the use of an elastic tether reduces the need for precise chaser-target coordination during capture, reduces collision risk during capture, simplifies subsequent de-tumbling, makes for lighter and mechanically simpler systems, and limits subsequent jerk with its potentially destructive consequences. The target debris is likely to be tumbling initially, so the chaser must achieve active de-tumbling before, or during, further manoeuvres of the "stack" (debris-tether-chaser), such as imposing a delta-V in order to de-orbit. The preceding task, of attachment of the tether to the debris, could be achieved in various ways, including by net capture or harpoon. A novel net capture technique will be briefly presented, but the techniques considered here will apply regardless of the attachment method. Note that some proposed capture methods require a separate de-tumbling prior to capture, whereas methods using an elastic tether can de-tumble after capture, thereby reducing mission complexity, uncertainty and mass. The first control challenge, then, is precisely due to debris tumbling prior to capture. If there are energy loss mechanisms on board, the debris rotation will probably be mainly around the axis of maximum moment of inertia. If so, and if the dominant debris rotation axis is approximately aligned with the tether axis, its main effect after capture will be a twisting action on the tether. If the rotation axis is approximately perpendicular to the tether, the debris will tend to wind up along the tether. In

either case, the chaser needs to respond so as to reduce the tumbling towards zero over time, while imposing control and avoiding tether entanglement and debris-chaser collisions. The general case can be a dynamic combination of these two extreme cases (with spin, nutation and precession), with possible slippage during any wrapping. Aside from de-tumbling, a further control requirement is to change the velocity of the debris-tether-chaser stack by a target mount in a target direction, while again avoiding entanglement, collisions, and uncontrolled pendular and libation motions, both during and after this delta-V manoeuvre. There may also be a subsequent requirement to allow the stack to coast for a relatively long period while ensuring the tether stays fully extended, to avoid entanglement during the period, and to be ready for a subsequent stack manoeuvre. The paper will show how wave-based control easily meets all these requirements using a single control strategy, combining position control and active vibration damping. The chaser-debris interaction is modelled as two-way wave motion, longitudinal (stretching) or rotational (twisting), travelling back and forth along the tether between chaser and debris. It can be shown that the effect of the control is to make the actuator behave as a viscous damper (motion absorber) to wave motion travelling in the tether from the debris towards the chaser. In this way when the tether is under tension the debris experiences a force and torque due to the tether which appears to have a viscous damper to ground at the far end, causing it to tend exponentially towards its natural (unstretched) length, and/or zero-twist condition. Meanwhile the chaser can simultaneously impose a desired velocity (or attitude) using an outgoing wave. The details of this combined control action and its implementation will be explained. Its advantages over classical control approaches include being simple to implement; very robust to wide variations in the debris parameters (which may be unknown); requiring minimal sensing; and all sensing can be done at the chaser end. The technique is tested in a detailed computer simulation of a complete de-tumbling of an arbitrarily shaped structure undergoing arbitrary tumbling motion. Then a complete de-orbit of the Envisat satellite to a target location is simulated and tested, and parameters such as fuel requirements assessed. The control technique was also tested on hardware using two model hovercrafts, one remotely controlled. Keywords: Debris removal, elastic tether, stack control, de-tumbling, mechanical waves, GNC

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Environment Modelling / 180

Planetary Orbital Dynamics (PlanODyn) suite for long term propagation in perturbed environment

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Trajectory design and orbit maintenance are a challenging task when multi-body dynamics is involved or in the vicinity of a planet, where the effect of orbit perturbations is relevant. This is the case of many applications in Space Situation Awareness (SSA), for example in the design of disposal trajectories from Medium Earth Orbits, or Highly Elliptical Orbits or Libration Point Orbits, or in the prediction of spacecraft re-entry, or in the modelling of the evolution of high area-to-mass ratio objects. On the other hand, the natural dynamics can be leveraged to reduce the propellant requirements, thus creating new opportunities. Orbit perturbations due to solar radiation pressure, atmospheric drag, third body effects, non-spherical gravity field, etc., play an important role in SSA. The semi-analytical technique based on averaging is an elegant approach to analyse the effect of orbit perturbations. It separates the constant, short periodic and long-periodic terms of the disturbing function. The short-term effect of perturbations is eliminated by averaging the variational equations, or the corresponding potential, over one orbit revolution of the small body. Indeed, averaging corresponds to filtering the higher frequencies of the motion (periodic over one orbit revolution), which typically have small amplitudes. The resulting system allows a deeper understanding of the dynamics. Moreover, using the average dynamics reduces the computational time for numerical integration as the stiffness of the problem is reduced, while maintaining sufficient accuracy compatible with problem requirements also for long-term integrations. This paper presents the Planetary Orbital Dynamics suite for long term propagation in perturbed environment. PlanODyn implements the orbital dynamics written in orbital elements by using semi- analytical averaging techniques. The perturbed dynamics is propagated in the Earth-centred dynamics by means of the single and double averaged variation of the disturbing potential. The single averaged disturbing potential due to luni-solar perturbations is developed in series of Taylor of the ratio between the orbit semi-major axis and the distance to the third body, following the approach by Kaufmann and Dasenbrock. The effect of other orbit perturbations such as the zonal harmonics up to order 10, the effect of solar radiation pressure and aerodynamics drag are also modelled. The double averaged potential is also implemented by averaging on the variable describing the orbital motion of the perturbing body (i.e. Sun or Moon) around the Earth, but the different inclination of the perturbing bodies is retained. Different application scenarios of PlanODyn will be shown: the behaviour of quasi-frozen solutions appearing for high inclination and high eccentricity orbits (HEO) can be reproduced and the re-entry of geostationary transfer orbits can be studied. In addition, to allow meeting specific mission constraints, stable conditions for quasi-frozen orbits can be selected as graveyard orbits for the end- of-life of HEO missions, such as XMM-Newton. On the opposite side, unstable conditions can be exploited to target an Earth re-entry; this is the case of the end-of-life of INTEGRAL mission, requiring a small delta-v manoeuvre for achieving a natural re-entry assisted by perturbations. Maps of stable and unstable HEOs are built, to be used as preliminary design tool for graveyard or frozen orbit design or natural re-entry trajectories at the end-of-life. Moreover, the application of PlanODyn to design end-of-life disposal from medium Earth orbits through passive solar sailing will be demonstrated.

Students (I) / 2

Orbit Determination through Global Positioning Systems: A Literature survey of Past and Present Investigations

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Positioning a spacecraft or a satellite in a predefined orbit was and is still considered as one of the most important process of any space mission design for military or commercial purposes. A need to prepare a comprehensive review of investigations conducted in this area was realized. Therefore present research was narrowed down to most significant contributions in the area of Global Positioning Systems applications only. Research work from 1988 to 2015 only has been included in this paper. It was observed that findings from 1988 start off fruitfully, with sub-meter orbit accuracy. Data from orbit determination agrees with Very Long Baseline Inferometry (VLBI) solutions. Most investigations focused on increasing accuracy of prediction through minimization of errors. Various orbit estimation strategies, process noise models for atmospheric fluctuations, combine processing of GPS phase and pseudo-range data were investigated in this period. It was also observed that Orbit modeling is not restricted to one type of orbit rather it included distinct categories. Such as low circular orbiters with altitudes up to a few thousand kilometers; Elliptic orbiters whose perigees are as low as a few hundred kilometers, and apogee's are as high as tens of thousands of kilometers. Finally, highly circular orbits were investigated with altitudes over tens of thousands of kilometers resembling geosynchronous satellites. This paper can be used as foundation for further investigations in Global Positioning Systems.

Verification and Validation Methods / 186

Rapid Deployment of Design Environment for EUCLID AOCS

design

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Euclid is a cosmology mission dedicated to study the geometry and the nature of the Dark Universe with unprecedented accuracy. Euclid will observe a 15000 deg^2 wide area of the sky from the Lagrange point L2 of the Sun-Earth system. The scientific goals of the mission result in very demanding performances for the AOCS subsystem; as an example, the observations' Relative Pointing Error shall be kept within 75 mas over a time scale of 700 sec. The Euclid S/C is procured by ESA and supplied by TAS-I; SENER is the prime contractor of the AOCS sub-system, for which the work is executed in partnership with ADS-NL. Amongst other responsibilities, SENER is responsible for the design, implementation and verification of GNC/AOCS algorithms; in the frame of the activities for the Phase B2 of the study, a set of tools and methodologies were developed and employed for the design tasks undertaken. The core of the design effort resided in the preparation and set-up of the Euclid Design Simulator (EDS). Such simulator is based on the internal tool SENERIC, whose model library was used as core or starting point for the DKE, actuators and sensors models required for Euclid. This allowed a fast deployment of environment for completing AOCS SRR in 1 month and AOCS PDR in 7 additional months. Along with the units and dynamics simulation, a simulation management architecture was developed. The main user for the EDS is the AOCS engineer, focusing on design tasks. As a consequence, the EDS simulation management infrastructure was developed taking into account the following usage guidelines: fast data preparation and processing, ability of storing and retrieving simulated cases, ability of running different models with the same source data set, ability of running Monte Carlo simulations, all without requiring lengthy and complex case set-up. The product of such effort is a simulator which allows launching test cases from MATLAB Environment initialization (leveraging on native types for data definition), as well as flexibly generating and retrieving XML databases equivalent to the MATLAB native data types. The EDS currently makes use of Simulink libraries and model referencing, and further efforts are envisaged towards the auto-coding of the AOCS/GNC algorithms being designed through the EDS. It is being used for developing models and specifications aiming at an ideally seamless integration within the formal verification environment, in first step via FES, and later for the RTS production to be applied in the AOCS SCOE. EDS is applied during the whole process as the flexible reference design tool.

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GNC MIL for Deorbiting with Drag-augmented Devices

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The paper describes the work performed and results achieved by LuxSpace for the ESA-supported DGNC project. DGNC stands for Dragsail GNC. It is a project aiming at identifying the best GNC solution to be proposed for satellite (debris) deorbiting thanks to Dragsails. The proposed and investigated GNC options are : (1) no attitude control, (2a) active attitude control constantly maximizing the area exposed to drag, this with a flat Dragsail, and (2b) active attitude control constantly maximizing the area exposed to drag, this with a pyramidal Dragsail. All DGNC options are also compared to deorbiting with (remainings of) onboard propulsion. In support to the DGNC system design and analyses, a GNC MIL i.e. a dedicated simulation tool has been created and validated within the ESA-supported GNCDE development environment. The paper describes also this LuxSpace's GNC MIL.

Re-entry and Aero-assisted Maneouvers / 184

IRENA - International Re-Entry demoNstrator Action

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IRENA is an action performed by an international consortium aiming at defining two technology demonstrator projects to validate advanced entry/re-entry technologies for further space exploration missions. In addition, IRENA shall create the ground for the implementation of these two projects in an international framework, such as promotion to ISECG, or next step in Horizon 2020 program. Technically speaking the consortium, which consists of European major industry, CNES, DLR, JAXA and NASA, has identified the EDL technologies requiring increase of the TRL level in near future, and identified possible flight and ground demonstration missions to cover the technological gap. In order to satisfy the needs of all involved parties, the selected demonstrator concepts focus on deployable decelerators, aerocapture and TPS ground validation aspects. Series of dedicated workshops have allowed to bring the demonstrator concepts to a completion of phase 0 development stage, and also define their implementation plan which is work still in progress. The presentation/paper will provide an overview of the IRENA project, workflow and the identified technology demonstrators proposed for future implementation. This action has received funding from the European Union's Horizon 2020 research and innovation programme, under grant agreement No 640277, within the strategic objective COMPET-2014. It has started in January 2015, and is planned to finish in April 2016.

Interplanetary Flight and Non-Earth Orbits (II) / 6

Some validation checks of "TriaXOrbitaL" tool : Earth-Moon L2 orbit, Sun-Moon perturbations.

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The modelling becomes year after year a priority for each agency, each company in order to better foresee the features of space devices and spacecrafts. The paper presents a flight dynamic tool so-called "TriaXOrbitaL", freely distributed as promotional release, that has been used since 1989 with continuous improvements for constellation design, North-South station keeping manoeuvre, orbit transfers GTO or better Super-GTO to GEO, interplanetary flight and travel to Moon. The tool has been fully focused on electric propulsion long thrust arcs but can manage as well of course chemical propulsion shorter arcs. The tool is accessible and has been used by engineers or students. The main advantage of using such tool is for sure a simplification of the preliminary studies because the tool has been developed for being accessible for every engineer willing to improve his knowledge in the orbital manoeuvres field. But one of the drawbacks when using such tools is to be able to state about the validity and accuracy of the results provided by the tool. Since the beginning of its development, a great care has been taken with respect to the fundamental checks of the tool. Hence the paper exhibits some of the tests performed in order to reproduce the "every body know" specific behaviors of the flight dynamic: for example the evolution of the GEO due to real Sun, Moon and earth potential perturbations (J2) with inclination up to 15° for 54 years; the stability of the Earth-Moon Lagrangian point L2, and several other checks. The description of the tests performed may constitute a good starting point for using the tool and for the knowledge of its features.

Low Thrust (II) / 97

Many-Revolution Low-Thrust Orbit Transfer Computation using Equinoctial Q-Law Including J2 and Eclipse Effects

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Mission designers addressing the computation of low-thrust many-revolution transfers need versatile and reliable tools for solving the problem with efficient computational times. This paper proposes a Lyapunov feedback control method, Q-law by Petropoulos with algorithm modifications to accommodate for the singularities in the original equations and to include the most relevant perturbations, such as the J2 perturbation and the effect of coasting during eclipse periods. The optimization of the control-law parameters via a multi-objective evolutionary algorithm (NSGA-II) improves the results significantly and permits to easily compute the minimum time transfer and a well-spread Pareto front, trading transfer time versus propellant.

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IXV GNC verification from inspection to flight demonstration

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The Intermediate eXperimental Vehicle (IXV) is an ESA re-entry lifting body demonstrator built to verify in-flight the performance of critical re-entry technologies. The IXV was launched on February the 11th, 2015, aboard Europe's Vega launcher. The IXV´s flight and successful recovery represents a major step forward with respect to previous European re-entry experience with the Atmospheric Re-entry Demonstrator (ARD), flown in October 1998. The increased in-flight manoeuvrability achieved from the lifting body solution permitted the verification of technologies over a wider re-entry corridor. Among other objectives, which included the characterization of the re-entry environment through a variety of sensors, special attention was paid to GNC aspects, including the guidance algorithms for the unique lifting body, the use of the inertial measurement unit measurements with GPS updates for navigation, and the flight control by means of aerodynamic flaps and reaction control thrusters. From a wider perspective, the development chain for the GNC starts from the shape conception, which implements the control authority needed during orbit and atmospheric flight, up to the production of the flight software which implements the GNC design. In IXV, the design and verification of the GNC has followed an ECSS based approach in which analysis (ex: Monte Carlo simulations) and test (processor and hardware in the loop) have been the main elements of validation to deliver a Qualified product and to verify the GNC for the last set of parameters before flight. The successful flight of IXV has constituted the final verification of the GNC and hence a significant milestone for Europe: the ARD flight demonstrated the GNC for a capsule and the flight of IXV has verified the GNC for a lifting body using active flaps. The flight constitutes not only the final validation of the GNC but also a valuable source for verification and tuning of methods and tools. Several steps in the postflight analysis are foreseen which incrementally will exploit the flight performance using different tools and techniques. An initial postflight analysis has been conducted using several inspection, simulation and reconstruction techniques. This paper describes on one side the overall design, verification and missionisation process for IXV with emphasis in the tools and techniques that have been applied, which include either engineering tools to formal analysis tools like the Functional Engineering Simulator (FES) or the Real Time Test _Bench (RTTB). On the other, the techniques used to derive the initial in-flight verification of the GNC will be presented as well as the main results and conclusions.

Low Thrust (II) / 99

Optimization of low thrust multi-revolution orbital transfers using the method of dual numbers

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The possibility of using the method of dual numbers in automatic differentiation for solving optimization problems of the low-thrust multi-revolution orbital transfers is considered. Traditionally the motion equations for the spacecraft with electric propulsion for multi-revolution orbital transfers are written in osculating elements or their modifications which exclude the special features in the right-hand sides of differential equations. These right-hand sides of the equations become especially complicated when different perturbations influencing the spacecraft movement are taken into account. Within the formalism of the Pontryagin maximum principle the right-hand sides of the optimal motion equations for the adjoints equations are quite complicated which results in some difficulties in solving optimization problems. Therefore the use of dual numbers method in numerical differentiation of optimal Hamiltonian for calculating the right-hand sides of the optimal motion equations of the spacecraft is effective. Another aspect of using the dual numbers method for numerical differentiation is to calculate the sensitivity matrix when solving boundary value problem corresponding to the optimal control problem. In this case, using dual numbers method allows obtaining the accurate sensitivity matrix. When using the continuation method for solving boundary value problem it helps to improve the convergence and to significantly reduce the number of steps for the external integration of Cauchy problem. The numerical results for optimal multi-revolution orbital transfer from the arbitrary initial orbit into the geostationary orbit are presented.

Rendezvous & Docking (II) / 98

Chattering-free Sliding Mode Control for Propellantless Rendezvous using Differential Drag

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Owing to the well-established correlation between spacecraft mass and mission's cost, there is great interest in fuel-optimal relative maneuvers between two or more satellites in the literature. In this context, the exploitation of natural perturbations is an attractive means to reduce or even remove fuel consumption, and, hence, propellantless maneuvers using solar radiation pressure, geomagnetic field, Coulomb forces, and atmospheric drag were proposed. Amongst them, the idea to use differential drag as control force for relative motion is particularly attractive to enhance the maneuverability of small satellites in low-Earth orbit, so that ongoing and forthcoming missions envisage this technique to achieve propellantless rendez-vous, cluster keeping, or constellation deployment, e.g., QARMAN, SAMSON, and Flock, respectively.
Because of several assumptions and modeling limitations, however, severe uncertainties affect satellite drag estimation. In order to effectively compensate for these uncertainties, successful attempts that include linear quadratic regulators, nonlinear adaptive control, model predictive control, and sliding mode control (SMC) were applied. SMC is widely adopted to cope with uncertainties due to its high robustness, however, it has two major drawbacks when applied to real-life problems:

- It usually causes **chattering**, i.e., high-frequency oscillations in the control force. When attitude control is used to tune differential drag, chattering can degrade the control performance and even jeopardize the maneuver if attitude actuators are saturated, e.g., reaction wheels.

- Prior knowledge of the **uncertainty bounds** is required. This is in general difficult to accurately estimate especially for time-varying drag forces.

In this paper a new chattering-free sliding mode controller is developed to perform an optimal rendez-vous of two satellites in low-Earth orbit, exploiting differential drag as control force. The proposed controller is designed to successfully compensate for uncertainty effects and/or unmodeled dynamics associated with air drag and to account for practical limitations such as input saturation. **Continuity of the control functions embedded in the new controller guarantees no chattering.** It is also shown that **an accurate estimation of the uncertainty bounds is not necessary.** Specifically, selecting conservative bounds leads to smaller errors, while the required control effort is not very sensitive to these bounds. Numerical examples are used to validate the robustness and accuracy of the new chattering-free sliding mode controller.

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Ascent Designer

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A Matlab program for the evaluation of launch conditions for interplanetary trajectories has been developed. The program allows to satisfy the forecast starting conditions of the selected outbound trajectories to inner or outer planets through the definition of different input data, such as:

- The components vector of the outgoing velocity defined in the selected local reference frame
- Two reference angles which respectively define the position of the Earth around the Sun on the basis of the escape dates selected and the associated declinations of the radial components.

Based on the input values introduced, as a standard approach, the program computes the new outgoing velocity values in Earth Centered Reference Frame (ECRF) together with the associated epochs, selecting the proper values of longitude of ascending nodes, allowing the selection of the different hyperbolic escape trajectories. Hence, for each of the computed escape orbits, the perigee latitude and longitude values at the indicated departure epochs are outlined, driving the selection of the proper launch sites from which the azimuth angles and the requested direct injection conditions can be obtained. Considering that, for the designed departure dates, the spacecraft results to be located at the computed perigee conditions, the time of launch from the selected launch sites can be computed.
Finally, on the basis of the launch site selected by the user, the program allows to compute the best azimuth angles in terms of escape velocity and maximum launch mass capabilities, for the different launch dates selected.

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An Access Point to ESA's Space Debris Data: The Space Debris Office Web Based Tools

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With DISCOS (Database and Information System Characterising Objects in Space) ESA's Space Debris Office has a very powerful database in hands when it comes to space debris related analyses.

It serves as a single-source reference for launch information, object registration details, launch vehicle descriptions, spacecraft information (e.g. size, mass, shape, mission objectives, owner), as well as orbital data histories for all trackable objects, which sum up to more than 40000 object entries.
Based on DISCOS and USSTRATCOM TLEs, the Space Debris Office routinely predicts upcoming re-entries as well as performs detailed analyses on high interest re-entries and ad-hoc risk assessments to missions after severe fragmentation events. To support these processes, the Space Debris Office also does their own solar activity prediction, based on publicly available solar activity data, with the SOLMAG tool.
All this data is of high interest not only within ESA but to the whole space flight community. It can be a valuable asset for analyses and operational processes, including but not limited to Space Debris related studies and collision avoidance. A reliable and controlled access to this information with maintained data quality is thus fundamental for the community. To accomplish this, the Space Debris Office is currently developing web based tools for DISCOS data access, including a machine friendly REST API, fragmentation analyses, and re-entry predictions. These will be complemented by web pages of more static nature, like the SOLMAG solar activity predictions, and of course by the already established Space Debris User Portal (https://sdup.esoc.esa.int) serving as distribution point for ESA's risk and mitigation analysis tools MASTER (Meteoroid and Space Debris Terrestrial Environment Reference), DRAMA (Debris Risk Assessment and Mitigation Analysis), and ORIUNDO (On-ground Risk Estimation for Uncontrolled Re-entries Tool).
This paper will introduce all web based tools under development and outline their function. The design will be addressed with a focus on user friendliness, function and harmonised look-and-feel. Special emphasis is put on security to not only protect ESA's data and server infrastructure, but also to implement ESA's data access and usage policy.

Coffee break / Poster Session / Booth Exhibition / 91

STAVOR: Transition from desktop to new mobile platforms

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Software environments have changed a lot in the last few years, since the release of the first smartphones. From the typical Windows-Unix market, where languages like Java emerged as multi-platform solutions, new very different systems have appeared leaving the term multi-platform obsolete. To refer to an application that can run in different systems and platforms (Desktop, mobile, web. . .) we use the term Cross-platform.
CS Systèmes d'Information is the main developer of an open-source space dynamics library called Orekit. This library is coded in Java, to benefit of its multi-platform capabilities. Since this language is still largely used for space applications, the operational use of the library in a near future is not at risk. On the other hand, its utilization in new environments like mobile applications is restrained. Many different solutions have been tested to integrate the library in such environments. The main problems of these new platforms and devices are the heterogeneity of hardware and frameworks: very different screen sizes and resolutions, storage capacity, new GPU architectures with not fully implemented standards, battery consumption, multiple switching connections, variable signal, application and device life cycle. . .
The library is already able to run in Android, due to the fact that it embeds a Java virtual machine, similar to the one in Desktop environments. For the integration into other devices like Apple smartphones and tablets, mobile devices based on web browsers and others, two approaches were considered: convert the code or binary of the library to a language recognized by each device, or convert it to a Web language and use it everywhere.
Many source code and binary converters have been used without success due to the data interface of Orekit. Such a solution needs of many modifications in the library code to success the conversion.

A solution that obliges to maintain different versions of the code and binaries was not desired, and the conversion to JavaScript requires the modification of all the data interface of Orekit, so this was saved as a possible solution in the future.
The last possible solution is to connect remotely Orekit, which demands a server and a fast internet connection, not very common in mobile devices.
To help with this study, the application STAVOR has been produced as an exemple. It uses Orekit as a space mission simulator, wrapping it in a touch-ready UI with some 3D and 2D visualization modules to represent the simulation results. This application has been implemented in an Android-native UI + embedded browser for the 3D and 2D models (based on WebGL and OpenLayers respectively), and in a pure embedded web solution, including all the UI.
The study concludes that Native solutions prevail over Cross-platform alternatives for the moment due to performances issues. In a near future, all mobile web technologies will be mature and stable enough to run heavy 3D and simulation software like any Desktop platform. At this point, cross-platform solutions will take the place to ease the development with adaptive interfaces and single code implementations.

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Solar System geometry tools with SPICE for ESA's planetary missions

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ESA has a number of science missions under development and in operation that are dedicated to the study of our Solar System (i.e. MEX, Rosetta, ExoMars, BepiColombo, Solar Orbiter and JUICE). The Science Operations Centres for these missions, located at the European Space Astronomy Centre (ESAC) in Spain, are responsible for all science operations planning, data processing and archiving tasks, being the essential interface between the science instruments and the spacecraft, and with the scientific community. From the concept study phase to the day-to-day science operations, these missions produce and use auxiliary data (spacecraft orbital state information, attitude, event information and relevant spacecraft housekeeping data) to assist science planning, data processing, analysis and archiving.
SPICE is an information system that uses auxiliary data to provide Solar System geometry information to scientists and engineers for planetary missions in order to plan and analize scientific observations from space-born instruments. SPICE was originally developed and maintained by the Navigation and Ancillary Information Facility (NAIF) team of the Jet Propulsion Laboratory (NASA).
This article outlines the different set of tools that are dedicated to geometry handling, visualisation and analysis software using SPICE from mission concept development though the entire analysis of mission data for ESA planetary missions.

Ascent (II) / 93

Launch vehicle multibody dynamics modeling framework for preliminary design studies

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Launch vehicle dynamics modeling is quite challenging mainly because of the highly interconnected disciplines involved: propulsion, aerodynamics, structures, mechanisms, and GNC among others.

Discipline experts perform their respective design often independently and with separate dedicated tools. Consequently, during launcher preliminary design studies, numerous iterations are required in order to keep mission objectives synchronized.
These preliminary design efforts can potentially be reduced by using a multidisciplinary launch vehicle model integrated in one single tool. Because this allows to reduce the number of iterations and the associated costs, a launch vehicle multibody dynamics modeling framework is a key technology to aim for.
Dedicated developments of multidisciplinary modeling tools for launch vehicle multibody dynamics have been presented in the relevant literature. However, none fully profits from an object-oriented, equation-based, and acausal modeling language like Modelica. As yet, such an approach is still missing. It is therefore the objective of this paper to introduce such an alternative approach employing this modeling framework.
This framework enables object-oriented and physics-based modeling of subsystems and components related to most key analyses of a launcher system. These include among others: launcher configuration, staging and separation dynamics, end-to-end trajectories, performance, controllability and stability. Moreover, all this can be done within a single simulation environment.
The paper gives an overview on the first building blocks leading to an integrated and multidisciplinary tool for launcher preliminary design studies. Particularly, its easiness of implementation is demonstrated along with the benefits of this approach.

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GNC simulation tool for active debris removal with a robot arm

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Studies have shown that the population of large objects has become a problem in Low Earth Orbit (LEO). The danger of collisions is higher than ever before and important objects are at high risk of major damage. One proposed solution for this problem is Active Debris Removal (ADR) using a robotic arm mounted on a spacecraft. A gripper or another adequate tool installed on the robot arm could capture almost every debris part (target) that endangers other satellites in the orbit. Once a connection (docking) is established, the chaser spacecraft can be used to safely deorbit the target by transferring it to a disposal orbit. To analyze this approach, a GNC simulation tool for Active Debris Removal with a robot arm was developed. A realistic benchmark scenario based on the capturing of the inactive Envisat satellite was chosen for a simulation study. The scenario considers uncertainties in the mechanical parameters and measurement noise. It focuses on the most critical phases of an ADR mission. In particular, the rendezvous phase, the capture phase as well as the GNC controlled de-orbiting phase.
The tool is used to design and simulate the GNC algorithms, satellite dynamics, kinematics and environment as well as the robot arm control. The developed control system consists of multiple parts at different hierarchical levels. A trajectory planning module coordinates different controllers of chaser satellite and robotic arm and provides reference trajectories for a successful docking of the two satellites. An optimization based approach trajectory was designed using an inverse satellite model of the chaser and the robotic arm. The results are used as the feed-forward control part of the two degree of freedom control approach. The trajectory design plays an important role to avoid a collision between the rotating passive target and the chaser satellite. Due to the rapid rotation of the target satellite, a simultaneous combined control of the chaser satellite together with the mounted robot arm is necessary, and considering actuator limits is crucial. The feed-back controller synthesis for the thrusters, reaction wheels and robot joint control was implemented using multi-case and multi-objective optimization with parameterizable models generated by the newly developed tool. This results in a robust control setup that is able to handle mechanical uncertainties and sensor noise.
The simulation tool can also be used to visualize the ADR scenario, based on CAD models of the satellite and the robot arm. The object-oriented design allows the change of modular components

and parameters of the simulation. Simulation results show that the suggested ADR benchmark scenario can be successfully completed using the developed control algorithms. The object-oriented GNC simulation tool for active debris removal with a robot arm allows the comprehensive design and analysis of an ADR scenario. It can be used to develop and verify the required GNC of the satellite and the control of the robot arm.

Low Thrust (II) / 160

Advanced Electric Orbit-Raising Optimization and Analysis with LOTOS 2

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Telecommunication satellites located in the Geostationary Equatorial Orbit (GEO) are typically not directly placed there by the launch vehicle. The satellites are often injected in a Geostationary Transfer Orbit (GTO) and then transferred to the GEO using their own onboard propulsion system. State of the art for the GTO to GEO transfer is still the chemical propulsion. Just recently few satellites transferred or are transferring to GEO using Electric Propulsion (EP), since it is very attractive to exploit their high specific impulse reducing the propellant mass of the orbit transfer. Since the total spacecraft mass is reduced this yields launch vehicle cost reductions. Further, Electric Orbit-Raising (EOR) is now available for most telecommunication satellite platforms or at least under development. But electric orbit-raising requires much more complex maneuver sequences than what is needed for pure chemical transfers. Since EP provides only small thrust magnitudes in comparison to chemical propulsion, the transfer lasts many months. A careful planning of the spacecraft attitude maneuvers is required in advance to fulfill this mission. In recent years, many software tools have been developed for the preliminary assessment of low-thrust orbit transfers. Unfortunately, most tools lack both maturity and accuracy necessary to fully exploit the capabilities of electric orbit-raising. For example, during the transfer any crossing of the GEO ring poses a certain collision risk with high value assets. Thus, the precomputed transfer trajectory has to avoid crossings of the GEO belt. Further, ground station visibility might be considered for transfer planning as well as limitations and constraints related to different spacecraft subsystems, such as eclipse handling, power generation, storage and consumption, or EP firing limitations in general. Other possible limitations are related to the attitude of the spacecraft or consider environmental aspects like the radiation dose. Using Non-Linear Programming (NLP) to optimize the attitude profile in combination with detailed modelling of complex mission constraints and limitations of the spacecraft model is essential, especially under consideration of tight accuracy and fidelity requirements for achieving optimality in sense of propellant consumption and transfer duration. Besides optimization, many aspects have to be analyzed in more details. It encompasses subsystem issues for example of the Attitude and Orbit Control System (AOCS) as well as station visibilities. In the newest version of the low-thrust optimization and analysis tool LOTOS (Low-thrust Orbit transfer Trajectory Optimization Software) all aforementioned features are available. But the tool is not only limited to electric orbit-raisings; it also supports hybrid transfers where chemical maneuvers are followed by the low-thrust transfer. Another feature of the software is the support of spacecraft operations. This mode identifies the spacecraft location on a pre-computed reference trajectory and uses its attitude profile for re-optimization. Such a processing is required due to deviations of the real-flown spacecraft trajectory from the nominal one. A full overview of the software capabilities and features will be given in this paper, such as hybrid transfers, 6 degrees of freedom attitude control and verification of trajectories. Highlights of the Graphical User Interface (GUI) will be presented as well. It includes automatic user defined analyses and customizable reports to support mission analysis engineers and to relieve them from repeating tasks. For example, the reports provide full access to all data of the user defined scenario and may include tables and plots.

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Simulation of autonomous landing near a plume source in a

tiger stripe canyon on the south pole of Enceladus

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The Institute for Space Technology and Space Applications (ISTA) of Bundeswehr University Munich is investigating mission concepts for the in-situ astrobiological exploration of the icy moons of the outer solar system. A concept studied in the context of the DLR funded Enceladus Explorer project (EnEx) aimed to place a lander near one of the plume sources on the bottom of a “tiger stripe” canyon on the south pole of Saturn’s moon Enceladus. Once there, the lander would deploy a melting probe to sample relatively shallow liquid water in the ice under the plume source. The lander would have to achieve a landing accuracy of 50 m and manage to land safely on an extremely challenging terrain interspersed with landing hazards like ice blocks, but also uncertain terrain like soft and unconsolidated snow or hard ice. To achieve this, a landing Guidance, Navigation, and Control (GN&C) system would be necessary to allow for autonomous landing operations. To achieve the required accuracy, terrain relative navigation can use sensors such as optical and thermal cameras, LIDAR, etc. to navigate relative to detected terrain features. To ensure a safe landing the system must be able to assess if the originally planned landing site is safe and if not, to then autonomously command a retargeting to another, safer spot. The guidance and control function must then calculate a viable trajectory and thrust arc to the newly chosen landing site.
To validate that the landing satisfies the accuracy and reliability requirements we are developing a tool in Matlab/Simulink to simulate the operation of the autonomous landing GN&C system. In this paper we present a first version of this tool, used to simulate the final phase of landing operations as described above. It comprises the following parts:
• Terrain simulation block: Generates a simple terrain model based on a given Digital Elevation Model (DEM) file. The topology can be modified using fractal algorithms and arbitrary simple shapes can be added to represent hazards. Terrain texture is also simulated.
• Simultaneous Localization and Mapping (SLAM) block: Simulates Terrain Relative Navigation (TRN)/feature matching, whereby features are extracted from the simulated terrain and a SLAM approach is followed for accurate navigation.
• Hazard Detection and Avoidance (HDA) block: Creates fused hazard maps from the output of camera and LIDAR sensors based on the simulated interaction of these sensors with the terrain. A fuzzy logic approach is used to evaluate landing safety and command a retargeting if necessary.
• Guidance and control block: Implements E-guidance and D’Souza guidance algorithms to generate a thrust arc and trajectory to a new landing spot if commanded by HDA.
Using this tool we will attempt to show the feasibility of an adequately accurate and safe landing near a plume source on the bottom of a tiger stripe canyon on the south pole of Enceladus.

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Trajectory Plan for the ascent and Re-entry of a Suborbital Passenger Spaceplane

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Compared to the orbital flight, the suborbital flight has much lower cost, therefore, the suborbital flight vehicle is more attractive for space tourist and scientific experiments. In our project, an aircraft-like suborbital flight vehicle is proposed. The flight vehicle has an aerodynamic layout similar to a normal aircraft. It can take off from a civil airport with two turbofans. At the

altitude above 12 km, the aircraft has a pull up flying into a parabolic trajectory and switches to the rocket propulsion mode at the same time. The rocket motor can propel the aircraft to an apogee about 120 km. Thereafter, the aircraft re-entries into the atmosphere. During the parabolic flight, the aircraft can obtain a micro-gravity flight of several minutes.
In this work, the ascent and re-entry trajectory of the suborbital aircraft is planned to make the micro-gravity flight as longer as possible. The aircraft intends to apply a low weight thermal protection systems, therefore the planned re-entry trajectory should result a low heating load to the aircraft. The aerodynamic load should also be as lower as possible to reduce the weight of the structure. The finally designed ascent and re-entry trajectory aims to make the cost the aircraft possibly lower based on a tradeoff of a variety of factors mentioned above.

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Sensor fusion analysis for HEO space debris using BAS3E

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One of the main missions of a Space Surveillance system is the detection and cataloguing of space objects having a size compatible with the detection constraints of its sensors. While radars are used to observe objects placed at Low Earth Orbits (LEO), and Telescopes to observe objects orbiting in Medium (MEO) and Geostationary Orbits (GEO), for objects orbiting in Highly Elliptical Orbits (HEO), both types of sensors are suited for observations. In particular, the passage through the perigee can be observed from radar stations, while in the high-altitude orbit portion telescopes are intended to be the source of observations. In this way, combining data derived from different type of sensor seems advantageous for tracking HEO objects.
BAS3E (Banc d'Analyse et de Simulation d'un Systeme de Surveillance de l'Espace) is a space surveillance system simulation bench that includes the "real world" simulation (objects and sensors) and the operational surveillance system (e.g. Cataloguing and catalogue maintenance, collision risk assessment, re-entry prediction and fragmentation detection). This tool enables the performance analysis of the surveillance network depending on its features, as well as the algorithms involved in the catalogue maintenance, analysis and planning systems. BAS3E is composed of different subsystems including:

- Observations simulation of surveillance network sensors: reference orbits generation, considered as "real" orbits, and the associated sensors observations generation.
- Catalogue maintenance system: sensors observations treatment and correlation, objects orbit determination, and objects database maintenance (the catalogue).
- Catalogue analysis system: orbits and covariances propagation, from catalogue objects; collision risk and reentry risk assessment using the propagated orbits; and fragmentation analysis and detection.
- Planning of sensor observations depending on previous analysis systems requests.

BAS3E makes use of parallel computing techniques, allowing, in a reasonable time effort, the simulation of thousands of space debris objects and the operation of a dedicated space surveillance network.
This paper describes an application using BAS3E, focused on the performance analysis of different space surveillance networks in the detection and cataloguing of a HEO population of objects. The use of BAS3E is described, including the concatenation of computation stages, the databases management and the persistence layers. Sensor networks considered within this study contain different combinations, in quantity and quality, of ground-based telescopes and radars. These sensor networks have been selected attending to criteria such as viable number of stations for a mid-term deployment and quality (measurement noise, observability constraints ...) in line

with already-operating sensors. This study tends to evaluate the advantages of fusing data from different sources (only optical measurements, or optical measurements along with radial distance and/or radial velocity data) concerning HEO orbit determination and, besides, conclusions are presented on the gain of incorporating additional sensors in a hypothetical space surveillance system ground network.

Coffee break / Poster Session / Booth Exhibition / 11

Dromobile: A multi-platform tool for orbit propagation on mobile devices

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The development of mobile devices such as smartphones and tablets has caused big social changes since the beginning of the 21st century. Their computing power and memory has also been growing continuously and can be competitive against traditional machines for not computationally expensive problems. The combination of state of the art Graphical Processing Units and Central Processing Units allows most of these devices to develop remarkable capabilities, both numerical and graphical. While they are being used in several professional fields, they still haven't caused a big impact in the astrodynamics community.
In this paper a multi-platform web application for client's side accurate orbit propagation is presented. This tool can be used not only on a computer, but also in modern mobile devices such as tablets and smartphones, either with or without Internet access. It exploits both the processor and graphical capabilities of the hardware to calculate the orbit and other useful information for a mission analyst and to present the results to the user in a friendly and intuitive manner. The astrodynamics tools available at the moment typically require a workstation to be used, which may delay the introduction of the mobile devices in the astrodynamics field. Since an easy-to-access and intuitive use of software should not be against accuracy and results consistency, Dromobile is intended to be a growing tool able to combine both aspects.
The special perturbation method Dromo is used to propagate the orbit, since it has been proved to have a good behavior in terms of computational speed and accuracy (see Urrutxua et. al. "Dromo propagator revisited", Celestial Mechanics and Dynamical Astronomy, 2015. doi: 10.1007/s10569-015-9647-y). The main advantages of Dromo are: the regularization of the equations of motion using a second order generalized Sundman transformation, the use of quaternions to orientate the orbital plane, and the fact that the Dromo elements are constant for Keplerian orbits, among others. The simulation environment is limited at the moment to geocentric orbits and the most important perturbations are included in the simulation.
The tool graphical interface is based on WebGL, which provides a light framework that does not takes away significantly computational power from the relatively heavier orbit calculations. It also allows complete portability between operative systems and devices since its standard is supported by all the major web browsers. Additionally, open source libraries based in the WebGL specification have been used in order to offer a free and flexible framework easy to modify and to add functionalities into.

Open Source (II) / 10

WebGeocalc: Web Interface to SPICE

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The WebGeocalc tool (WGC) provides a web-based graphical user interface to many of the observation geometry computations available from the "SPICE" system (SPICE=Spacecraft, Planet, Instrument, Camera- matrix, Events; see http://naif.jpl.nasa.gov). It is based on client-server architecture. A user selects a computation and computation parameters using GUI widgets in a standard Web browser. The request is sent to a WGC geometry server that performs the computation and sends results back to the client.
WGC can compute various geometric parameters such as positions, orientations, and surface intercept coordinates for a given time or a series of times; perform geometric event searches for time intervals when a geometric parameter had a particular value or when a geometric condition is met; and perform time conversions. By a simple drag-n-drop action output value or time intervals from one WGC computation can be saved within a WGC session for use in subsequent computations, allowing users to perform cascading geometric event finder searches, with output from one search being used as input for the next one. Saved intervals can also be used to compute geometric parameters over a time window retuned by a search.
WGC offers many useful features such as plotting results and downloading results. It can plot geometric parameters versus time and versus each other, with both kinds of plots providing zoom capability, with ability to return to the previous or initial zoom levels. It presents numeric results in a scrollable table that can be downloaded in MS Excel, CSV, and simple text format.
Currently the NAIF group at JPL operates a WGC server (see http://naif.jpl.nasa.gov/naif/webgeocalc.html) that has access to all data on the NAIF web site. This server is configured to provide easy access to generic SPICE kernels and to SPICE data sets formally ingested in NASA's Planetary Data System. Users can select these data sets by simply picking them from the WGCs kernel set menu. Users can also load into WGC any kernels available on the NAIF server, including operational kernels for many missions, but it has to be done manually, which is more time consuming and requires some knowledge of kernel naming schemas and contents.
WGC can be deployed on any computer that has Java, Apache Tomcat web server, and MySQL Community Server. While NAIF does not plan to make the WGC software available to the general public, it might make the WGC binary war file together with installation and kernel database configuration instructions available to organizations involved in planetary exploration, with significant experience with SPICE and a clear need to manage their own kernel sets used by WGC.

Multidisciplinary Design Optimization / 13

Optimization and tools for deployment and reconfiguration of formation flying Missions

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Formation flying plays an important role in future space missions. Formation deployment and reconfiguration are key technologies of formation flying missions. Optimal multiple impulsive formation deployment and reconfiguration are studied in this paper, software design tool is developed.Firstly, Efficiency of tangential, radial and normal direction maneuvers is analyzed. Secondly, optimal tangential maneuvers for formation deployment and reconfiguration are studied, mathematical models of three and four impulsive deployment and reconfiguration are derived. Initial guesses are studied for solutions of three and four impulsive maneuver model. Three impulsive maneuver problem is resolved by differential corrections method. Optimal three impulsive maneuver solution are obtained using the continuation method. Four impulsive maneuver problem is resolved by analytical method. Global optimal four impulsive maneuver solution is obtained by genetic algorithm. Symmetric deployment trajectories are proposed to

avoid the attitude disturbance of mother satellite.Thirdly, multiple impulsive maneuver model without maneuver direction constraints is derived and optimized using genetic algorithm.Fourthly, optimal semi-analytical solutions for formation deployment and reconfiguration are studied.Finally, the algorithms are integrated in a design software tools. we won the first prize in the 7th Chinese Space Trajectory Design Competition with the aid of the design tool.

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OCCAM: Optimal Computation of Collision Avoidance Maneuvers

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The continuous growth of the population of objects in Low Earth Orbit (LEO) has caused an increase of the conjunctions between active satellites and other objects, either space debris or other satellites. It is mandatory to evaluate the risk these conjunctions pose, and to design the corresponding collision avoidance maneuvers if necessary. Since several maneuvers are to be performed in the satellite lifetime, the maneuvers should consume as low fuel as possible.
OCCAM (Optimal Computation of Collision Avoidance Maneuvers) is a novel software tool aimed at computing minimum-fuel collision avoidance maneuvers in the short-term encounter scenario, which is generally applicable in LEO. Developed by the Space Dynamics Group of the Technical University of Madrid, it employs advanced modeling and optimization techniques, which make it an extremely fast and robust design tool. OCCAM features an extensive set of input parameters, different optimization strategies and output options to provide a high design flexibility for the user. Several methods of collision probability computation are also supported. Its user-friendly graphical interface and intuitive design logic make it really straightforward to master even for non-experts, and it can be employed either as a standalone tool or in conjunction with other satellite operation planning frameworks.
In an increasingly complex operational scenario, OCCAM does what other collision avoidance planning tools do but in a fraction of their computation time, making it a fast and reliable design and planning tool for the space operators seeking to minimize the cost of their collision avoidance maneuvers.
A trial version of this tool called OCCAM lite is available on-line for the interested potential user at the web page of the Space Dynamics Group: http://sdg.aero.upm.es/index.php/online-apps/occam-lite. The fact that this tool is capable of running in a web-browser (either on workstations or mobile devices such as tablets or smartphones) is a proof of the outstanding velocity of this software.

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Adaptive Fuzzy Controller Design for Flexible Air-breathing Hypersonic Vehicle

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An investigation of a tracking control system with uncertainties for a flexible air-breathing hypersonic vehicle is proposed. First, a control-oriented model was developed by modifying the flexible air-breathing hypersonic vehicle model with the uncertainty of flexibility taken into consideration. Then, a tracking control model for a flexible air-breathing hypersonic vehicle was established based on input/output feedback linearization. A self-adaptive control law based on fuzzy approximation (approximation of the uncertainty function related to the flexibility of the

fuzzy system) was designed, and the stability of the system was verified using Lyapunov's theorem. Additionally, numerical simulation was carried out to investigate the tracking control system of a flexible air-breathing hypersonic vehicle, thus verifying the effectiveness and robustness of the fuzzy self-adaptive control law.

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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.
For the DESEO development, reliable and proven algorithms and routines have been used. Moreover, the need of flexibility and modularity drove the design toward an Object-Oriented (OO) architecture design. The OO architecture also reduces maintenance and upgrade costs. The DESEO toolkit, besides the DEIMOS native algorithms, also integrates ESA EO CFIs libraries, providing additional means for performing embedded analyses.
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 currently composed of 38 modules (Analysis Processes) that can be used as stand-alone tools (command line) or operated via a Graphical User Interface (GUI). The Analysis Processes 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). The GUI provides functionalities to manage the input insertion process (e.g. importing data from a database or from other input files), the analysis executions and monitoring (by means of log messages and progress bars) and the output visualisation. The GUI visualisation module is capable of producing 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 and Mac OS X Operating Systems.

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A fast and efficient algorithm for onboard LEO intermediary propagation

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Usual onboard orbit propagators are provided as navigation maintenance aids for earth orbiting satellites. These programs should be able to forecast satellite ephemeris within a reasonable accuracy for short time periods, which may range from minutes, as in the case of momentary lack of GPS signal, to several satellite orbits. In these brief intervals the accumulation of second order effects of the Geopotential is barely apparent, and, therefore, the propagation model can be very simple. Hence, common onboard orbit propagators are based in the fixed-step numerical integration of the J2 model, a truncation of the Geopotential limited to the zonal harmonic coefficient of the second degree.
On the other hand, the use of analytical, intermediary solutions of the J2 problem has been recently proposed as an efficient alternative to the numerical integration. The accuracy of common intermediary orbits of the J2 problem is limited to first order effects, thus providing less precise solutions than the numerical integration. However, because of the inherent uncertainty of the initial conditions to be propagated onboard, it can be shown that both alternatives, the numerical integration and the intermediary approach, enjoy the same statistics. Other benefits of using analytical solutions is that they may improve both memory allocation and computation time, a fact that may be crucial to Cubesats ore other small satellite missions, in which the computational abilities may be restricted.
Note, however, that neglecting the long-period effects associated to the odd zonal harmonics introduces small errors in the propagation, which are clearly observable even in the short-term. These errors can exceed 1 km in the along-track direction at the end of one day. Hence, taking into account the disturbing effects of some higher order harmonics may notably improve the propagation model.
Here, we propose a new intermediary solution that takes into account the first three zonal harmonics of the Geopotential (J2, J3 and J4). Since the solution is analytical, the evaluation is very fast and is not constrained to a step-by-step evaluation. In spite of the forces model of the new intermediary is much heavier than the simple J2 model, its evaluation can be fastened using some simplifications that alleviate the computational burden, in this way making the new intermediary definitely competitive when compared to the numerical integration of the J2 problem.

Open Source (II) / 19

Uniform Trajectory Locators (UTLs) – an open API for trajectory discovery and utilisation

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With the dramatic increase in rideshare opportunities for CubeSats and other small spacecraft, landers and rovers, platforms offering to fly large numbers of virtual payloads, and projects to manufacture spacecraft-on-demand on orbit by the thousand, discovery and planning of trajectories for actual and potential space objects is becoming a bottleneck. Multiple organisations have, or are planning, projects that require the launch and calculation of trajectories for hundreds to hundreds of thousands of space systems to be sourced and managed over the next decade.
To address this issue, we propose Uniform Trajectory Locators (UTLs), an open specification for a web based representational state transfer (RESTful) application programming interface (API) permitting the manual and automated discovery of existing, upcoming or potential trajectories for use by new and existing, virtual and physical, space systems and payloads.
UTLs permit the distributed decentralized publishing of trajectories of space objects, whether launch opportunities, existing spacecraft in orbit, or potential opportunities for deployment in low earth orbit, in interplanetary space or on planetary bodies. As well as publishing existing

opportunities, UTLs can be used to request trajectory solutions from domain specialists and automated tools, permitting production and operations schedulers responsible for sourcing and managing the launch and operation of constellations of satellites and exploration tools to autonomously search for optimal solutions for missions.
UTLs build on existing web standards such as Uniform Resource Locators (URLs) and eXtensible Markup Language (XML), and return results in an object orientated Trajectory Markup Language (TML). UTLs are designed to permit different levels of truth and detail to be returned by their publishers to consumers by tools ranging in complexity from static text files to sophisticated algorithmic back ends. The level of detail and information provided about the same trajectory by different stakeholders (e.g. launch providers, brokers, potential consumers, space situational awareness providers, etc.) can be substantially different for commercial, political, technical or temporal reasons. The same trajectory can be published by multiple independent distributed providers with a security and transaction model allowing consumer by consumer level information disclosure and automated electronic trading of access to trajectories. UTLs can be used to provide lightweight wrappers around existing tools to provide platform independent interfaces to existing databases of launch opportunities, orbital parameters and flight dynamics tools.
A variety of demonstration proof of concept UTL tools and data sources under development with academic and commercial space systems providers will be presented, with a focus on sub-kilogram scale platforms supporting the mass exploration of space for science, education and commerce.

Ascent (I) / 18

An Intelligent Multidisciplinary Design and Optimization Environment for Conceptual Design of Launch Vehicle

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Conceptual design process of aerospace launch vehicle is extreme challenge. A great number of alternative innovative or traditional configurations need to be evaluated, and the corresponding design parameters need to be balanced and defined. To quantitatively account for the interactions between disciplines and obtain a set of optimal design parameters, the methodology of Multidisciplinary Design and Optimization (MDO) is prevalently applied in the community of engineering. However, MDO only support parameter optimization for a given configuration. While too many potential configurations need to be evaluated during conceptual design, most of designer's work hours was spent on setting up MDO problems and models for alternative configurations. How to efficiently utilize MDO technologies and speedup conceptual design process is still a challenge topic in launch vehicle industry. Recently, a more general and intelligent multidisciplinary design and optimization environment, MCDesign-LV, is under development in Northwestern Polytechnical University (NPU) of China. In this paper, we will present the preliminary progress on MCDesign-LV. Firstly, the function requirements and architecture of MCDesign-LV is defined. Then, the main modules involved are introduced, including Configuration Reasoning Module, Disciplinary Design and Analysis Modules, Parametric MDO Model Generation Module, MDO Solving Strategy Generation Module. After that, three critical technologies to support MCDesign-LV are discussed: (1) How to define system configuration arbitrarily? The configuration of system is composed by that of disciplines. To explore design space as large as possible, the disciplinary design modules should have the capabilities of defining arbitrary configuration. We will discuss how to implement in the disciplines of aerodynamics, trajectory, propulsion, guidance, control, structure, and thermal. (2) How to generate the parametric MDO models automatically? To alleviate human task in MDO modeling for each configuration, parametric model generation tools were developed, including a CAD tool for geometry, multi-fidelity simulation models for disciplinary analysis. The input/output of disciplinary models will be tagged and integrated automatically. (3) How to create MDO solving strategy intelligently? By defining objective functions, shared and local design variables, system and local constraints, the coupling strength and scale of MDO problem will be evaluated. Then, with certain algorithm, a solving strategy will be generated, i.e., which kind of architecture (MDF/IDF/CO/CSSO) applied, which optimization

algorithm and surrogate modeling process used. At the same time, the computational procedure will be distributed to network nodes automatically. Finally, two launch vehicle conceptual design examples with different configurations are introduced to demonstrate the capability of MCDesign-LV. One example is rocket based launch vehicle, and another is combined cycle based air breathing launch vehicle.

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Comparison of deterministic, safety margin and reliability-based MDO formulations for the design of a launch vehicle

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In the preliminary level of a launch vehicle design process, an adapted Multidisciplinary Design Optimization framework can support the designers to reduce time and cost of development while exploring and analyzing a high number of possible architectural concepts necessary at these phases. Designers may then identify the most promising solutions that will become the baselines for concept refinements at the detailed phase. The solutions obtained at the early design phase are impacted by uncertainty due to the lack of knowledge about the future system and its environment and due to the use of low fidelity models in order to tackle the computational cost for the design process. The assessment of the launch vehicle performance and reliability requires handling uncertainty in the MDO process to identify robust and reliable potential solutions. Two main approaches exist in the literature to deal with MDO under uncertainty, either safety margin-based MDO methods or reliability-based MDO methods. Safety margin-based methods usually perform the system optimization deterministically using conservative safety factors in order to take the uncertainty effect into account. These approaches are interesting because they enable to limit the computational cost but they require a process to find the appropriate safety margin in order to avoid conservative design and to increase the system performance while guaranteeing its reliability. These margins have to be tuned by performing a complete reliability analysis to control the conservativeness of the solution. Reliability-based MDO methods directly handle uncertainty in the MDO process by a two level nested process with uncertainty propagation and optimization. These methods do not use any safety margin and find a solution satisfying the reliability specifications but can be very computationally expensive. This paper proposes to compare the classical deterministic design method to the safety-margin-based and RBDO methods on a representative launch vehicle design test case in order to evaluate the conservativeness of the different methods and the quality of the found solutions.

Orbit Determination and Prediction Techniques (II) / 116

A TLE-based Representation of Precise Orbit Prediction Results

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A TLE-based Representation of Precise Orbit Prediction Results

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Openly accessible TLEs are widely used to predict orbit positions of space objects. The computing efficiency of the analytic SGP4 algorithm is particularly attractive in applications where orbit positions of thousands of objects are needed. On the other hand, the accuracy of the TLE-computed positions is far less satisfactory than that of orbit predictions using more rigorous orbit determination and prediction methods. The delivery of the accurate orbit determination and prediction results is usually in the form of evenly spaced position and velocity, that would need a data file with size of hundreds of KB. This would require large computer storage for hundreds of

thousands of space objects, and the use of these orbit data files in the space conjunction analysis would also require large computing storage.
This paper presents an algorithm using TLE/SGP4 to represent the accurate orbit prediction results. First, the accurately predicted positions are used as observations to determine a set of TLE employing the SGP4 algorithm. Then, the differences between the accurate positions and TLE-computed positions are fitted with a series of sinusoid terms. In this way, the accurately predicted position is computed as the sum of the position computed using TLE and the correction computed using the series of sinusoid.
The algorithm is experimented for a prediction time of 30 days for 4 satellites at altitudes of 690km, 820km, 1500km and 5800km. The accurate orbit predictions are obtained after the orbit determination using laser tracking data. 100 simulation runs are performed for each satellite. The maximum errors representing the accurately predicted positions over the 30-day prediction period are about 220m, 85m, 76m and 34m for the four test satellites, respectively. The representation accuracy clearly has dependency on the orbit altitude.
Experiment results show that the predicted orbits over a period as long as 30 days can be represented in the proposed algorithm in accuracy at 200m for satellite at altitude of 700km, and 30m for satellite at altitude of 5840km. The algorithm is purely analytic, so it is computationally very efficient. Also, it uses TLE and a short series of sinusoid, only a data file of 1 KB is needed to represent the predicted orbit. Therefore, the algorithm could find application in the future space conjunction analysis involving hundreds of thousands of debris objects.

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A semi-analytical orbit propagator program for Highly Elliptical Orbits

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A semi-analytical orbit propagator to study the long-term evolution of spacecraft in Highly Elliptical Orbits is presented. The perturbation model taken into account includes the gravitational effects produced by the first nine zonal harmonics and the main tesseral harmonics affecting to the 2:1 resonance, which has an impact on Molniya orbit-types, of Earth's gravitational potential, the mass-point approximation for third body perturbations, which on ly include the Legendre polynomial of second order for the sun and the polynomials from second order to sixth order for the moon, solar radiation pressure and atmospheric drag. Hamiltonian formalism is used to model the forces of gravitational nature so as to avoid time-dependence issues the problem is formulated in the extended phase space. The solar radiation pressure is modeled as a potential and included in the Hamiltonian, whereas the atmospheric drag is added as a generalized force. The semi-analytical theory is developed using perturbation techniques based on Lie transforms. Deprit's perturbation algorithm is applied up to the second order of the second zonal harmonics, J2, including Kozay-type terms in the mean elements Hamiltonian to get "centered" elements. The transformation is developed in closed-form of the eccentricity except for tesseral resonances and the coupling between J2 and the moon's disturbing effects are neglected. This paper describes the semi-analytical theory, the semi-analytical orbit propagator program and some of the numerical validations.

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Low Thrust Trajectory Design and Optimization: Case Study of a Lunar CubeSat Mission

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The NASA CubeQuest Challenge offers a launch on the 2018 Exploration Mission 1 (EM1) to competing teams who can demonstrate a high probability of success for a 6U CubeSat design. The selected competitors will have their CubeSat disposed into a high-energy lunar flyby trajectory soon after EM1 launch. If the CubeSat remains ballistic, the lunar flyby will eject it from the Earth-Moon system, reaching a distance of 4 million km in 2 months. The trajectory design goal of the Challenge is a thrust profile that will allow the CubeSat to enter a stable lunar orbit (with prescribed orbit size bounds), and maintain that orbit for 28 days, all within 1 year of disposal. Due to EM1 safety considerations, high-thrust chemical propulsion is not allowed, which eliminates the possibility of direct lunar orbit injection during the lunar flyby. This limits competitor teams to using low-thrust propulsion and designing trajectories that take advantage of the natural Earth-Moon-Sun system dynamics.
In this paper, we examine the incremental and iterative process of designing and optimizing a low-thrust trajectory that achieves and maintains lunar orbit within the prescribed 1 year. We start by exploring the post-disposal dynamics of the Earth-Moon-Sun system, and determine the general transfer trajectory types that remain in the vicinity of the Earth-Moon system. Next, we design a transfer trajectory using multiple unconstrained impulsive maneuvers, and look for minimum Delta-V solutions where the individual Delta-V magnitudes would be attainable in realistic time with thrust levels on the order of 1 mN. Finally, we convert the impulsive maneuvers to finite-burn maneuvers and re-optimize to maintain a continuous trajectory while minimizing total thrust duration. In doing so, we incorporate the need to perform periodic tracking and orbit determination as a constraint on the maximum continuous thrust time, and discuss the impact of this constraint on the optimal trajectory.
This is a particularly difficult trajectory design problem due to a highly nonlinear dynamical system (Earth-Moon-Sun) and a milli-newton thrust propulsion system. We discuss the use of Earth-Sun L1 dynamics to transition from an Earth-departing trajectory to a lunar arrival trajectory. We use a multiple-shooting method to match these trajectories, and show how this greatly aids in optimization convergence. We also discuss the benefits of using a stable lunar distant retrograde orbit (DROs) as an intermediate target before spiraling the CubeSat down into its final low lunar orbit. Finally, we address the design of the lunar spiral-down phase and final lunar orbit itself, including orbit stability and visibility from ground stations.
We use the NASA GSFC General Mission Analysis Tool (GMAT), combined with the VF13ad optimizer, for trajectory design and optimization. GMAT allows us to set up the trajectory design problem using impulsive propulsion, then easily transfer that solution to finite-burn propulsion. Additionally, the ease with which the VF13ad optimizer can be used from GMAT allows us to transition from a feasible to an optimal solution with minimal changes to the problem setup.

Optimization and Dynamics (II) / 153

Massively parallel optimization of low-thrust trajectories on GPUs

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The optimization of low-thrust trajectories is a difficult task. While techniques such as Sims-Flanagan transcription gives good results for short transfer arcs with at most a few revolutions, solving the low-thrust problem for orbits with large numbers of revolutions is much more difficult. We developed a massively parallel genetic optimization algorithm to obtain low-thrust solutions for targeting of a celestial body, such as the Moon or a planet. The solution is not limited to simple arcs or few rotations, but is capable of solving the problem also in the case of many revolutions where other classical methods fail. While we perform the propagation in a two-body model, in principle the propagation can also be performed in more complete models such as the circular restricted three body problem.

The optimization algorithm chosen is a genetic algorithm with large population size. Due to its massively parallel nature, this type of problem is a natural fit for implementation on a GPU. Modern GPUs are capable of running thousands of computation threads in parallel, allowing for very efficient evaluation of the fitness function over a large population. In particular, we optimize the shape of the control function as well as the departure time. Propagation of the spacecraft is then performed in massively parallel fashion on the GPU before the results of the fitness function are read back into CPU memory for preparation of the next iteration of the algorithm.
We demonstrate with various examples how this algorithm can provide good initial guesses for a following local optimization to compute very accurate low-thrust orbits.

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Hybrid SGP4: tools and methods

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The propagation of an orbit which is subject to perturbation forces is a non-integrable 6-degree-of-freedom problem that has been classically tackled in three different ways. General Perturbation Theories attempt to deduce an analytical expression for the future position and velocity of the orbiter as a function of its initial conditions and time. Nevertheless, the non-integrability of the problem makes it necessary to consider simplified models for the perturbing phenomena, as well as certain approximations that have a negative effect on long-term accuracy. Another classical approach, Special Perturbation Theories, consist in applying numerical integration methods to the problem, which allows for the consideration of very complex mathematical models of perturbing phenomena. Nevertheless, in order to obtain highly accurate results, small integration steps must be taken, which implies long computational time. The third way to handle the problem, Semi-analytical techniques, apply analytical transformations so as to remove the short-period dynamics from the equations of motion, which can be numerically integrated then through longer integration steps, and hence more reduced computational time. By doing so, long-term propagation can be performed very efficiently. Short-period dynamics can be recovered at the final epoch in order to complement the propagated mean elements and thus provide the osculating elements. More recently, we have proposed a new approach, Hybrid Perturbation Theories, which consist of an integration method followed by a forecasting technique. The former, which can be any of the aforementioned techniques, is intended to generate an initial approximation, whereas the latter, which might be a statistical time series model or computational intelligence method, complements that approximation by forecasting its error at the final epoch. In order to achieve it, the second stage must model the dynamics corresponding to the difference between the output of the first stage and the real behaviour of the orbiter. An initial control period containing such dynamics is, therefore, necessary. We will consider a hybrid propagator composed of SGP4 plus an additive Holt-Winters method in order to describe this methodology and the software it involves.

Interplanetary Flight and Non-Earth Orbits (II) / 155

Trajectory Design Tools for Libration and Cis-Lunar Environments

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The Sun-Earth libration and Cis-Lunar environments are challenging regimes for trajectory designers with complex multi-body dynamics, perturbation modeling, and integration of propulsion influences. Beginning with libration orbits and research on dynamical systems (aka manifolds), several tools with application to libration orbits and Cis-lunar regions have been developed in cooperation between NASA's Goddard Space Flight Center and Purdue University. One of these innovative tools, the Adaptive Trajectory Design (ATD) tool is being used in conjunction with commercial software to design both a multi-body trajectory for the upcoming Lunar IceCube (L-IC) Cubesat mission and the Wide-Field Infrared Survey Telescope (WFIRST) Sun-Earth L2 mission transfer. As a payload deployed by the Exploration Mission-1 (EM-1) on the maiden flight of NASA's Space Launch System (SLS), L-IC will use a lunar gravity assisted, multi-body transfer trajectory with an innovative RF Ion engine to achieve lunar capture and delivery to the science orbit. WFIRST trajectory deign is based on an optimal direct transfer trajectory to an L2 orbit. In the paper, ATD utilities that permit the designer to categorize orbits by energy and amplitudes among other numerous design variables, Circular Restricted Three Body methods, and manifold generation are described along with the transfer trajectory design process for both missions. Based on the constrained L-IC EM-1 architecture and deployment, an assessment using ATD and dynamical system research tools has uncovered Euclidian regions of Cis-lunar space which permit a transition onto stable/unstable manifolds that encounter the Moon at the prerequisite arrival conditions, resulting in an innovative process. Using ATD's powerful Poincare mapping tools and libration orbit generation via energy or orbit amplitudes, feasible WFIRST science orbits are generated that feed into the selection of optimal transfer manifolds from the low Earth orbit injection condition. These ATD utilities for both missions permit the interweave mapping of manifolds and conics to complete any Cis-lunar or Libration orbit design. ATD's innovative applications will be fully defined and the basic operations and its interface to GSFC's General Mission Analysis Tool (GMAT) for high fidelity modeling are presented.

Coffee break / Poster Session / Booth Exhibition / 154

Analysis of spacecraft trajectories in proximity to small bodies: Phobos & NEO

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Associated to sample return missions to Phobos (Phobos-Grunt) or a mission targeting NEOs (MarcoPolo-R, AIDA), one generally aims at following trajectories that optimize Delta-V consumption, hovering stations, solar panel illuminations, … We perform analysis of trajectories around such low gravity object considering in both cases the perturbed gravitational or non gravitational environment. Missions constraints consists in phases of hovering, different altitudes approaches and duration, and touch down. Starting from given reference missions, we will present orbitability studies around either quasi-satellite orbits (QSO) or equilibrium points (EP).

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Proposed algorithm for on-board maneouvres calculation

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The orbit control for LEO missions is becoming more and more demanding in terms of manoeuvring. This paper proposes a simplified algorithm in order to calculated on board the drag make-up manoeuvres.

It is not pretended to give a full solution of the problem, but a starting point for future implementations. There are two main rationale of calculating this manoeuvres on-board, the first one is to mitigate the operational load to prepare and execute them, the second one is to remove the uncertainty on the orbit evolution between the manoeuvre calculation to its execution time, mainly due to the atmospheric drag. The resulting operational concept allows to keep very tight orbit control with high accuracy and low operational load.
The proposed algorithm compares the actual time of ascending node crossing with respect to the reference one. The delta-time provides the shift with respect to the reference ground-track while its increment allows to determine the altitude of the satellite. When a threshold in the ground-track shift is reached a manoeuvre is triggered based on the current altitude.
Alternative more complex calculations can be implemented allowing applications for formation flying. For example improving the on-board knowledge of the position of the other satellites in the constellation.
The algorithm is based on the one used to control the ground-track of GOCE, therefore it is not only applicable to chemical propulsion or cold gas, but on the contrary it is highly appropriate to low thrust missions. It introduces also the concept of micro-manoeuvres, that allows to emulate a low thrust control with medium thrust propulsion.

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Asteroid Rendezvous Uncertainty Propagation

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Most methods of propagating orbit uncertainty assume posteriori Gaussian distributions, require an intrusive implementation or suffer from the curse of dimensionality associated with high-dimensonal random inputs. Although Monte Carlo techniques avoid these draw backs, the approach has a slow convergence rate. This paper considers the application of separated representations for orbit uncertainty propagation and discusses the theory behind their generation. The computation cost of a separated representation is largely linear with respect to dimension, thereby improving tractability when compared to methods that suffer from the curse of dimensionality. Generation of a separated representation requires the propagation of a small number of samples and yields an approximate solution, or surrogate, to a given stochastic differential equation describing the propagated orbit. This surrogate provides information on the moments and spatial density of possible solutions, as well as the sensitivity of the quantities of interest with respect to random inputs. This paper presents the case of spacecraft targeting an asteroid for a rendezvous, for which the initial conditions of each and the components of the interceptor maneuver are uncertain. Separated representations is used to estimate the probability of a successful rendezvous and analyze the resulting probability distribution functions.

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Evaluation of Iterative Analytical Techniques for Interplanetary Orbiter Missions

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The preliminary design of interplanetary direct transfer trajectories is generally done using the patched conic technique. This design consists of hyperbolic excess velocity vectors at both the ends, say Earth and Mars. For an orbiter mission, a particular inclination and periapsis altitude of

the approach hyperbola must be achieved. For a given departure date and a fixed flight duration, there are two hyperbolic transfer trajectory options at each end that matches the excess velocity vector for a specified inclination. Further, each of the two Earth side options can be mapped to each of the Mars side options, thus resulting in four distinct hyperbolic transfer trajectories. Two iterative analytical techniques that generate four distinct design options are introduced in this paper. The iterative analytical techniques are based on the concepts of patched conic and pseudostate methods. The distinct design options are achieved using an analytical tuning strategy. This strategy arrives at suitable hyperbolic orbit characteristics that achieves the excess velocity vector after certain duration, which is a fraction of the total flight duration, in the departure phase. The iterative method based on patched conic technique considers the gravity field of the planet alone within its sphere of influence and that of the Sun alone outside. So, the design obtained by this technique results in large deviations in the target parameters such as the closest approach altitude (CAA) and the related time on numerical propagation under a realistic force model. To improve the achievable accuracies on the target parameters, an iterative method based on pseudostate technique is used and an improved design is generated. A comparison on the deviations in the target parameters that are obtained upon numerical propagation of these designs under similar force model is made. The benefits derived while attempting numerical refinement of these analytical designs are quantified.

Ascent (II) / 158

Model Validation Framework for Launchers: Post-Flight Performance Analysis

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The launchers GNC performance is strictly depending upon its knowledge of the system properties and its evolution throughout the flight. The GNC design and verification integrates mathematical models derived at sub-system level to predict system level behaviour. Their improvement is a key task through the development and validation of a new launch vehicle. Hence the exploitation of flight data shall be maximized in order to reduce at minimum the lack of accuracy in the mathematical models. It is clear that the exploitation of in-flight data during qualification flights in order to improve the launch vehicle models plays a key role in the tuning of the GNC system algorithms and architecture for future flights.
This paper presents the Model Validation Framework for Launchers which is a set of tools and algorithms for the determination and validation of accurate mathematical models and parameters based on pre-, in- and post-flight measurement data.
The Model Validation Framework for Launchers is made up of the following components:

- Measurement Pre-processing component in charge of preparing the flight measurements in order to be used by the Trajectory Reconstruction and Parameter Estimation components.
- Trajectory Reconstruction component in charge of computing the launch vehicle trajectory using the pre-processed flight measurements.
- Parameter Estimation component in charge of characterizing the selected launch vehicle models based on flight measurements, after pre-processing, and vehicle states based on a-priori postulated models.
- Model Validation component in charge of confirming the correctness, accuracy, adequacy and applicability of the identified models with their corresponding estimated parameters.

The target application of the Model Validation Framework for Launchers in this activity has been the VEGA Flight Programme Software Alternative (FPSA).

Multidisciplinary Design Optimization / 112

Joint Optimization of Main Design Parameters of Electric Propulsion System and Spacecraft Trajectory

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The problems of joint optimization of the main design parameters of electric propulsion, power supply systems and trajectory of the spacecraft is considered in two formulations: minimum-thrust problem and maximum payload mass problem under the optimal parameters of the electric propulsion system (thrust, exhaust velocity, power). The first problem allows solving one of the fundamental problems in optimization of trajectories for the spacecraft with finite thrust related with the existence of the solution which is one of the reasons for the complexity of the constructing robust and efficient numerical optimization technique. Indeed, if the numerical scheme does not converge to a solution then the real reason is unknown since it can be either the absence of a solution or the numerical scheme failure. Therefore, identification of the boundary of the solution region is an actual problem. In this way, solving the minimum thrust problem gives necessary assessment. The minimum thrust problem is formulated similar to any other low thrust trajectory optimization problem but the functional to minimize is the thrust. It is supposed that the thrust can be either maximal or zero, the maximal thrust magnitude is constant along the trajectory and its direction is unconstrained. The transfer time is fixed. The second problem is directly related to the optimization of the parameters of the propulsion system in the region of the existence of solutions. It is well known that for every space transportation operation there is an optimal value of specific impulse corresponding to the minimum total mass of electric propulsion system, power supply system providing electric propulsion operation and propellant. It is easy to show that there is also an optimal value of electric power of the electric propulsion system that being associated with the growth of the required characteristic velocity while reducing the thrust. It is obvious that the optimal values of specific impulse and electric power of the main electric propulsion system can be found by joint optimization of the trajectory and design parameters. An approximated design model of the spacecraft equipped with main electric propulsion system and the Pontryagin's maximum principle are used for optimization. The necessary conditions for the optimality of specific impulse and electric power of the electric propulsion system are derived. Numerical examples of the joint optimization for the interplanetary spacecraft trajectory are presented.

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Low Thrust transfers applications for Earth Orbiting Satellites and Constellations

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The SOFTT (Space Optimal Finite Thrust Transfer) is a tool designed to compute the spacecraft optimal orbital transfer trajectories using low thrust propulsion system; SOFTT is able to optimize very long orbital transfers - optimal solutions lasting up to 2 years have been studied - between Earth orbits considering eclipses and perturbations. SOFTT was developed in Thales Alenia Space Italy to study various low thrust missions like GALILEO (LEO to MEO) and NEO50 (LEO to GEO and GTO to GEO). Among the specific features of the mission studied that will be presented are: o LEO to GEO in presence of eclipse and perturbations o Optimal deployment of constellations on multiple planes. o Perigee altitude constraint o Vmin solutions"

Re-entry and Aero-assisted Maneouvers / 49

Optimal Lunar Landing

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“This paper discusses the problem of lunar landing guidance. It is focused on the powered descent portion that leads to a lunar touchdown. The paper starts by discussing the dynamical models and environment models used in the analysis. This includes the spherical harmonic gravity model of the moon and the moon-fixed dynamical models that include the accelerations induced by the moon’s rotation. It then discusses two successful lunar landers, the U.S. Surveyor and Apollo Lunar Module, and their guidance systems. This is followed by a literature survey on optimal lunar landing algorithms.
The next sections discuss lunar transfer and give results from a linear tangent landing algorithm. This is a two-dimensional problem with a flat lunar surface. This is applied to a two-dimensional spherical moon problem by rotating the thrust vector as the descent progresses. The terminal algorithm is based on a bang-bang controller and is designed to insure a vertical descent. The next section replicates these results with MATLAB’s fmincon and then applies fmincon to the full three-dimensional problem. For both problems in fmincon the descent is discretized into N segments but the time of each step is also a decision variable. In the three-dimensional problem both thrust angle and throttle are decision variables. This allows an easy computation of fuel-time optimal landings. The two-dimensional result matches the analytical solution exactly.
The last section applies the YALMIP package to the 3D problem. Convex solvers and branch and bound solvers are employed and the results compared. Different constraints are applied and their effect on the trajectories are demonstrated.”

Satellite Constellations and Formations / 46

Analysis of Electric Propulsion Capabilities in Establishment and Keeping of Formation Flying Nanosatellites

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Nanosatellite clusters constitute one of the current and future trends in space technology. [1] These clusters can be used in a variety of applications including search and rescue, communication, earth monitoring, etc. In order to maintain a satellite cluster over a long period of time it is required that the inter-satellite distances between the cluster agents will be controlled. [2] The nanosatellites need to mitigate the along-track drift created by the initial orbit injection, using the very limited resources available onboard. In the mass range of 1 – 10 kg, Cubesats have strict constraints on allowed mass, volume, electrical power, and carry only limited sensor and actuator capability. [3] This paper explores the capability of state of the art miniaturized electric propulsion (EP) systems to establish and maintain a nanosatellite cluster mission. Different EP technologies will be considered. Due to the low thrust provided by an EP system, long orbit control maneuvers are required. Therefore, mission design is highly effected by long-term attitude and power constraints. In contrast to previous work that either ignored the attitude limitations or assumed that the satellite orientation is dedicated to the orbit control maneuver, [4] the proposed paper will focus on developing an autonomous controller that can work under realistic attitude and mission constraints. The paper compares 6 types of satellite body forms, with deployable and body fixed solar panels, each with its own constraints on mass, volume, and power. The paper will present the results of the analysis and provide a conclusion regarding the possibility of current EP technology to be applied for maintaining a nanosatellite cluster.
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Coffee break / Poster Session / Booth Exhibition / 86

Modular Fuzzy Interacting Multiple Model for Maneuvering Target Tracking

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As a branch of the field of target tracking, maneuvering target tracking plays an increasing role in military and civilian fields. A novel maneuvering target tracking algorithm is investigated. Drawing on the experience of combination idea of the modular structure and the fuzzy interacting multiple model algorithm (FIMM), a modular fuzzy interacting multiple model algorithm (MFIMM) is presented. The MFIMM algorithm consists of three independent modules working in parallel, called non-maneuver, weak maneuver and strong maneuver respectively. And the change of a target motion is also divided into three levels: no, small, and big. The motion of a target is detected by a fuzzy control motion detector, which imitates the thoughts of human beings to detect a target's motion. Once the maneuver is detected, the MFIMM algorithm selects one of the three modules matching the actual movement of the target every moment according to maneuver condition, and the state vector and covariance matrix is compensated, so that the modified state can suit the actual motion well. Afterwards, the MFIMM algorithm estimates the state of the target through interactive multiple model algorithm (IMM) based on square root unscented Kalman Filter (SRUKF) of the selected module. Therefore, under the architecture of the proposed algorithm, the fuzzy motion detector deals with the level of motion and the modules switching, whereas the IMM-SRUKF accounts for the estimation of the dynamic system. At the end of the paper, simulation is performed on the problem of maneuvering target tracking in two-dimensional space. In order to evaluate the effectiveness of the MFIMM method, Root Mean Square Error (RMSE) of the estimated state is used. The simulation test tracks a same target with determined trajectory by MATLAB emulation comparing the two algorithms, MFIMM and IMM. Results demonstrate that the proposed MFIMM algorithm improves the tracking precision and reduces the computational burden compared with traditional IMM.

Low Thrust (I) / 44

Enhancement of DLR/GSOC FDS for Low Thrust Orbit Transfer and Control

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On behalf of future missions with electric propulsion controlled by DLR's German Space Operations Center (DLR/GSOC) the present operational multi-mission Flight Dynamics System (FDS) is enhanced to support the preparation and operation of corresponding satellite missions.
For designing an easily extendable framework, low-thrust scenarios including orbit raising to GEO, GEO station keeping as well as LEO missions were considered. Each low-thrust phase is modelled by a thrust profile comprising non-equidistant thrust vector and thrust level. In a first step, we consider the thrust level to be constant over a thrust phase.

Based on this design several multi-mission FD software modules are enhanced to handle low thrust phases, e.g. orbit prediction, orbit determination including thrust level estimation, and generation of orbit related products.
The low-thrust transfer trajectories are optimized by means of the software package ASTOS/GESOP [1]. This package was configured within the present FDS with astrodynamics models used at DLR/GSOC to ensure maximum compliance between Mission Planning/Analysis results and operational realization. Operational and technical constraints, e.g. thrust interruption during eclipses, are implemented, too.
Throughout the paper, we will demonstrate the extended FDS capabilities by means of a GEO positioning reference mission. We found a remarkable consistency between resulting ephemerides by the optimizer in comparison to the FDS, which shows the correct processing of thrust profiles within the FDS. For the future the need for extension to variable thrust levels will be analyzed.
[1] GESOP© by Astos Solutions (2015), Version 7

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Lunar Mission One: Crowdfunded Exploration of the Lunar South Pole

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In 2024, Lunar Mission One plans to launch an unmanned robotic landing module to an area of the South Pole of the Moon unexplored by previous missions to provide understanding of the formation of the solar system. Lunar Mission One has been in development for seven years and in December 2014 was successfully crowd-funded through the crowd-funding platform 'Kickstarter'. The mission is going to use and develop pioneering technology to drill down into the lunar surface a depth of at least 20m and potentially as deep as 100m. By drilling this deep we will access lunar rock dating back 4.5 billion years to discover the geological composition of the Moon which has been unaffected by radiation and discover the ancient relationship it shares with our planet and the effects of asteroid bombardment. Ultimately, the project will improve scientific understanding of the formation of the solar system, and the conditions that initiated life on Earth.
We will engage with the public's imagination by placing a 21st century time capsule inside the borehole that we drill on the Moon to be preserved for about a billion years by the exceptional conditions within the Moon. There will be two sections to this time capsule: A private archive which will allow the public over the next 10 years, anyone around the world will be able to buy a 'place in space' to have their own individual Digital Memory Box where they can upload pictures and video files of their choice. They can also have a DNA sample (a strand of hair) placed inside the physical capsule. This will be buried into the surface of the Moon.
The second section is public archive will contain a publically assembled, authoritative record of life on Earth, with a history of humankind and a species database that chronicles the Earth's known biodiversity and how it all fits together – from geology to atmosphere. This archive will be available online both during development and after the Mission has been accomplished.
Lunar Mission One has recently partnered with Astrobotic Technology Inc. and has launched a campaign for the public to send pictures of their footsteps to the Moon, free of any charge by 2017."

Ascent (I) / 42

Innovative Strategy for Z9 Reentry

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ICATT 2016, 6th International Conference on Astrodynamics Tools and Techniques, 14 - 17 March 2016, ESOC, Germany Innovative Strategy for Z9 Reentry Gregor Martens*, Elena Vellutini**, Irene CrucianiELV Corso Garibaldi 22, 00034 Colleferro (Italy)**Aizoon Viale Città d'Europa 681, 00144, Roma (Italy)
Abstract Large Footprint of Zefiro 9, the 3rd stage of VEGA launcher spanning even more than 2000 km in equatorial missions, is one of the major system drawbacks constraining both performances and missionization process. This paper proposes a new strategy for the reentry of Zefiro 9 third stage of the VEGA launch vehicle, consisting of employment of retro-rockets coupled with closed loop guidance, which permits to improve the performance of the launcher and to reduce Z9 footprint. Z9 is one of the specific characteristics of VEGA: it works at high velocities and high altitude. Being a solid rocket motor it cannot be simply cut off and the impulse delivered depends on the propulsion scattering, not known a priori. This uncertainty produces a big variation in the Z9 impact point, function of the propulsive performance of the SRM. Current solution for the Z9 reentry foresees the employment of Neutral Axis Maneuver by orienting the thrust along the (neutral axis) predefined direction in order to minimize the impact point variation when an impulse is delivered along. NAM is performed in open loop guidance several seconds before Z9 cut off which is not predictable with precision, hence a certain percentage of its propulsion capability is lost in the maneuver. The new reentry logic permits to exploit the whole Z9 energetic capacity by not performing the neutral axis maneuver. Footprint extension is moreover drastically reduced by employment of retro rockets: small solid rocket booster with a fixed impulse of velocity. After Z9 exhaustion a slew manoeuver points the launch vehicle to the target attitude computed on-board and retro rockets are activated immediately after separation. The reentry logic is deeply analyzed and the adopted optimal reentry strategy is formulated. The improvements are evaluated in terms of Z9 footprint extension. Obtained results are compared with respect to the current reentry strategy. Possible error sources (i.e. navigation, guidance and control errors) are critically evaluated and their impact on the results is highlighted.

Rendezvous & Docking (I) / 43

GNC Techniques for Proximity Manoeuvring with Uncooperative Space Objects

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Active debris removal and satellite servicing are some of the current hot spots in space research: plenty of engineering challenges, they deal with fully or partially uncooperative orbiting objects to be approached and captured autonomously by another space vehicle. The Active Debris Removal (ADR) topic focuses on trading-off, designing and making operational mechanisms placed on board an active chaser that can rendezvous with and grapple an inert and tumbling target, to eventually change its dynamics transferring it to a disposal orbit. On the other hand, satellite servicing (SS) deals with refuelling and/or maintenance of active spacecraft, and therefore supposed to be partially cooperating, to be approached and docked by the active chaser to carry out the needed operations when connected. To perform these tasks, different techniques are currently being proposed in literature, starting from the robotic arm to grasp the target to tethered nets/tentacles to wrap it. From dynamics point view, these technologies differ for the flexibility involved in different elements and connections. A general-purpose system design should effectively intervene on objects different in configuration, materials and possibly in dimensions.
ADR and SS tasks define new challenges for Guidance, Navigation and Control (GNC): these missions cannot be tele-operated and ground-controlled due to communications delays, intermittence, and limited bandwidth between the ground and the chaser. Therefore, there is substantial interest in performing these operations autonomously: the research work, here presented, moves in that direction and have the main objectives of • developing reliable and validated dynamics models, to drive ADR and SS systems design and support GNC implementation, including the

flexibility modelling and contact dynamics of capture mechanisms and coupled stacks configurations; • implementing GNC laws adapted to perform the involved operations, from approaching to removal\servicing, to demonstrate mission feasibility and increase the level of autonomy; • validating dynamics models and control laws through experimental activities, including microgravity campaigns and hardware-in-the-loop testing.
A multi-body dynamics simulation tool was developed in house at Politecnico di Milano – Department of Aerospace Science and Technologies, fully integrated in Matlab/Simulink and suited to design guidance and control laws: it provides a fast and accurate simulation environment to describe multiple bodies' six degrees of freedom dynamics, possibly linked by different flexible/rigid connections and including flexible appendages, propellant sloshing and a detailed environmental model to account for all the relevant perturbations, especially at low altitudes. An upgrade of the abovementioned simulator was also implemented to describe the deployment and wrapping dynamics of flexible nets around targets, with the inclusion of collision detection and contact dynamics algorithms. A parabolic flight campaign was successfully performed to validate both flexible dynamics and contact dynamics models: the net 3D trajectory was reconstructed using stereovision (an ad hoc Matlab software was implemented to this end).
In the paper, the abovementioned multibody dynamics tools, their validation process and preliminary simulation output are presented, for both rigid and flexible techniques."

Verification and Validation Methods / 40

Tool for Real-time Pradiction of IXV Trajectory in the Mission Control Center

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IXV (Intermediate eXperimental Vehicle) re-entry vehicle has successfully flown a fully autonomous mission from the launch until it's splashdown in the eastern Pacific Ocean on 11th February 2015, and has been regarded as a major milestone in European re-entry technology roadmap. The vehicle has been designed to fly autonomously without the commanding from ground and without a need for a continuous downlink. It's trajectory and attitude however had to be closely monitored in MCC for operational and safety purposes including when out of the visibility window. For this purpose a dedicated too has been developed - TPVT (Trajectory Propagation and Visualization Tool).
The tool updated the current state and trajectory predictions based on the IXV vital-layer telemetry data whenever it was in the direct visibility of any of the ground stations, and also taking into account latest atmospheric measurements by the sounding balloons. In order to represent the vehicle's behavior as close as possible, the same GNC algorithms have been implemented in the propagation algorithm. For sake of propagation velocity the utilized models had to be kept simple in order to allow regular update of the trajectory every couple of seconds whenever fresh telemetry data was available. The propagated trajectory was displayed for the MCC operators for monitoring purposes, and the current state vector transmitted to the naval ground station in specific antenna pointing data format in order to facilitate the acquisition of the signal on the naval station. At the end of the mission the TPVT had an essential role in the vehicle's localization by providing an updated expected splashdown position, where it was found by the recovery ship's crew just 20 minutes later.
This paper gives and overview of various features of the tool, its interfaces with the MCC, visualization features, and also an assessment of its propagation performance during the real mission.

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Launcher mission analysis platform for fast and accurate mission domain performance assessment

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Facing the civil launcher market evolution toward higher versatility of missions and shorter development cycles, Airbus Defence & Space has improved its mission analysis platform for Ariane 6 needs. This platform enables fast and accurate assessments of launcher architectures throughout the development phases. It contributes to launcher staging optimization, tuning of engine characteristics and sizing trajectories delivery to design teams. The mission domain covers a large range of injection orbits often requiring multiple firings of the upper stage. The trajectories have to satisfy numerous design and operational constraints, especially related to safety rules. The upper stage disposal scenario is also part of the mission analysis since it must comply with the space law while inducing the lowest possible performance penalty.
The mission analysis platform is based on Airbus Defence & Space optimization trajectory tool. An automatic and robust convergence toward the optimal payload injection trajectory must be guaranteed on the whole Ariane 6 mission domain. The core of the optimization process is a solver for the upper stage flight. This solver based on an indirect method yields quasi instantaneously the upper stage optimal leg reaching the targeted orbit whatever the initial state. An automatic initialization procedure using the in-house nonlinear programming libraries guarantees the fast convergence over the Ariane 6 mission domain.
The Flight Performance Reserves (FPR) ensure a given mission success probability regarding flight dispersions and design uncertainties. FPR must be carefully estimated at the early stages of the design process since they directly influence the launcher performance. The FPR are assessed on-line along the optimal trajectory by a semi-analytical method yielding the optimal sharing between several stages. The results are checked by a Monte-Carlo simulation modelling a simplified on board guidance. The FPR assessment process has thus been significantly fastened.
Close-range safety is analysed by a preliminary modelling taking into account the features of the debris generated in case of an in-flight launcher destruction. The safety constraints are thus accounted at the early stages of the launcher design. Refined assessments are later performed by statistical methods to check the results.
The upper stage disposal scenario is also systematically assessed within the optimization process whenever the injection perigee lies within the protected LEO region. The required propellant budget is minimized taking into account the fall down safety and the boil-off penalty involved by ballistic legs. The scenario is a posteriori validated during the development by usual Ariane5 methods.
The entire process is integrated in an automated workflow enabling thousands of computations while checking data transfers between engineering domains. Realistic propellant budget models enable to adapt, when necessary, the upper stage propellant loading. Various architectures can thus be assessed very quickly yielding performance derivatives and dispersed trajectories.
This paper will present the whole process, the main technical new developments and illustrative cases.

Coffee break / Poster Session / Booth Exhibition / 1

Application of Kalman Filters in Orbit Determination: A Literature Survey

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Orbit determination has been a critical area of research ever since the start of space age. Research has been done in this area from defining accuracy goals for predicting orbits, to a host of software and hardware tools, techniques and methods. This paper was narrowed down to the most significant contributions in the area of Kalman filtering applications. Research work from 1967 to 2013 only is included in this paper. It was observed that examples from 1967 demonstrate the importance of pre-flight parametric studies when orbit determination estimations are carried out in short periods. Research also shows that extended Kalman filters can be made to work in real-data situations, contrary to what some professionals in the aerospace industry believe. Development of the extended Kalman filter requires some proper selection of parameters that define the probability density of the initial state vector, and any other parameters required by modifications to the extended Kalman filter algorithm. More recent research uses numerical methods such as the 4th order Runge-Kutta method to conclude that propogation errors are not a problem for 15-day solutions for orbits. This paper can be used as foundation for further investigations in the area of orbit prediction.

Open Source (II) / 5

poliastro: : An Astrodynamics library written in Python with Fortran performance

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Python is a fast-growing language both for astronomic applications[[1]] and for educational purposes[[2]], but it is often critizised for its suboptimal performance and lack of type enforcement. In this paper we present **poliastro**, a pure Python library for Astrodynamics that overcomes these obstacles and serves as a proof of concept of Python strengths and its suitability to model complex systems and implement fast algorithms.
poliastro features core Astrodynamics algorithms (such as resolution of the Kepler and Lambert problems) written in pure Python and compiled using numba, a modern just-in-time Python-to-LLVM compiler. As a result, preliminary benchmarks suggest a performance increase close to the reference Fortran implementation, with negligible impact in the legibility of the Python source. We analyse the effects of these tools, along with the introduction of new ahead-of-time compilers for numerical Python and optional type declarations, in the interpreted and dynamic nature of the language.
poliastro relies on well-tested, community-backed libraries for low level astronomical tasks, such as astropy[[3]] and jplephem. We comment the positive outcomes of the new open development strategies[[4]] and the permissive, commercial-friendly licences omnipresent in the scientific Python ecosystem.
While recent approaches involve writing Python programs which are translated on the fly to lower level code, traditional Python libraries for scientific computing have succeeded because they leverage computing power to compiled languages. We briefly present tools to build wrappers to Fortran, C/C++, MATLAB and Java, which can be also useful for validation and verification, reusability of legacy code and other purposes.

Optimization and Dynamics (I) / 9

MOSQP: an SQP Type Method for Constrained Multiobjective Optimization

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We propose an SQP type method for constrained nonlinear multiobjective optimization. The proposed algorithm maintains a list of nondominated points that is improved both for spread along the Pareto front and optimality by solving single-objective constrained optimization problems. We provide numerical results for some trajectory optimisation problems as well as for a test set of unconstrained and constrained multiobjective optimization problems. The numerical results confirm the superiority of the proposed algorithm against a state-of-the-art multiobjective solver, either in the quality of the approximated Pareto front or in the computational effort necessary to compute the approximation. Moreover, we discuss convergence to local optimal Pareto points under appropriate differentiability assumptions

Satellite Constellations and Formations / 146

Design Formation Flying in the small satellite class

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The exploitation of cooperative satellites flying in close formation is an appealing technology with potential to overcome some limitations intrinsic to the single platform satellite systems. Scientific applications go from the effective implementation of single pass Synthetic Aperture Radar (SAR) interferometers for high accurate planetary digital elevation models generation, to the implementation of dual(multi)-spacecraft telescopes. Moreover, the satellite formation concept brought the key benefit of distributing the scientific payload among the satellites of the formation, with mass and power demand reduction on each single space segment; this offers the chance to exploit small and even micro/nano-satellite satellite class . Those classes offer an economical benefit from one side, being the currently available technology performance the main limitation as far as the micro/nano-sats are considered.
In this framework, the authors aim to set up a software tool to support the satellite formation architecture definition, depending on the specific scientific/commercial mission requirements. Flexibility is one of the key drivers of the tool here presented, to concurrently handle the constraints related to the specific satellite class of interest together with the imposed mission objectives requirements.
The assessment among possible formation architectures takes into account a wide range of design variables, as for example: the platform technology, the reference orbit, the number of satellites to set up the formation, the geometry for the spacecraft relative configuration. Moreover, the tool leans on a spacecraft 6 DOF dynamics modeling, fast and accurate, to exploit at the best the environmental perturbation to drive the formation flying architecture design to minimize the active control. This assessment procedure is integrated in a multi-objective optimization that takes as input the high-level mission requirements and constraints, and gives as output the formation sizing parameters that maximize the required performances.
Considering the SAR interferometry application. The vector of decision variables to be optimized (x*) is composed, for example: by the most relevant sizing parameters of the Radar system (e.g. antenna dimensions, frequency of transmission and associated bandwidth, maximum angle of incidence of the Radar signal at the ground, the pulse repetition frequency of the system), by the formation geometry sizing parameters (e.g. number of satellites, initial conditions of the relative motion) and by some satellite characteristics (e.g. ballistic coefficients of the different elements of the formation). Other parameters (p) that characterize either the Radar system or the satellites are considered fixed (e.g. the Radar system temperature, ground back scattering coefficient, reflectivity property of the satellite surfaces). The cost function vector (J) is composed by the most relevant resolution performance indexes of both cross-track and along-track SAR interferometers (e.g. phase height sensitivity factor, ambiguity height, variety of baselines provided by the formation over a specific planetary area of interest). Finally, the fundamental constraints

(g) are imposed (e.g. minimum antenna area, maximum admissible cross-track and along-track baselines to not incur in decorrelation, minimum admissible safe separation of the satellites, maximum power available).
As a first case study, useful to identify the key variables for the tool, a formation flying concept around the Saturn's moon Titan for complete bi-static SAR mapping is here presented and critically discussed. The solution exploits the natural perturbations around Titan to provide a big variety of baselines at all planetary latitudes and therefore, high resolution and unambiguous digital elevation model of the whole Titan surface, while minimizing the fuel consumption necessary for the formation control (45 m/s of total ΔV during 600 days of planetary mapping for both formation and station keeping manoeuvers).

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Optimization and Dynamics (II) / 147

Optimal real-time landing using deep networks

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Optimal trajectories for spacecraft guidance, be it during orbital transfers or landing sequences are often pre-computed on ground and used as nominal desired solutions later tracked by a secondary control system. Linearization of the dynamics around such nominal profiles allows to cancel the error during the actual navigation phase when the trajectory is executed.
In this study, we assess the possibility of having the optimal guidance profile be represented, instead, by a deep artificial neural network trained, using supervised learning, to represent the optimal control structure. We show how the deep network is able to learn the structure of the optimal state-feedback outside of the training data and with great precision. We apply our method to different interesting optimal control problems, including the inverted pendulum swing-up and stabilization problem, a quadrocopter time and power optimal pin-point landing control problem and a time and mass optimal spacecraft landing problem. In all cases the deep network is able to safely learn the optimal state-feedback also outside of the training data making it a viable candidate for the implementation of a reactive real-time optimal control architecture.
We perform our study making use of the open source projects Space-AMPL, to compute the supervision signal, and Theano to train the deep network.

Debris, Safety and Awareness (III) / 144

Space Situational Awareness Capabilities of the Draper Semi-analytical Satellite Theory

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The fundamental requirement of space situational awareness (SSA) is to provide actionable knowledge about events and activities in Earth orbit. A key component of SSA is space surveillance – determining the present position of space objects and the ability to predict their future orbital paths. Related requirements are the detection of new space objects, the detection of spacecraft maneuvers, and the prediction of when one space object may interfere with another space object. Such interference may be physical or electromagnetic in nature. The Draper Semi-analytical Satellite Theory (DSST) is based on based on a synergistic combination of techniques from analytical satellite theory and from traditional numerical integration. The conventional equations of motion and the associated numerical processes are replaced with: 1. Equations of motion for the mean elements, 2. Expressions for the short periodic motion, 3. Semi-analytical theory for the partial derivatives, 4. Semi-analytical theory truncation algorithms, and 5. Interpolation for the mean elements, mean element state transition matrix, mean, mean element partial derivatives, and the Fourier coefficients in the short-periodic expansions. There are three comprehensive implementations of the DSST: 1. GTDS Orbit Determination system (Fortran 77) 2. DSST Standalone Orbit Propagator (re-factored Fortran 77) 3. Orekit open source DSST (java) GTDS was the original development platform for the DSST. The DSST Standalone Orbit Propagator (F77) makes the DSST algorithms available without the overhead of GTDS. The Orekit DSST demonstrates portability of the DSST algorithms to an object-oriented software platform. The DSST is very desirable for Space Situational Awareness due to: Comprehensive force models The large grid size employed to integrate the mean element equations of motion and to compute the short-period Fourier coefficients The capability to tailor the semi-analytical theory force models at execution time. The compactness of finite time interval ephemeris representations for multiple space objects orbit estimation algorithms (both batch and recursive) built to estimate the mean elements directly from the tracking data The near linearity of the mean element equations of

motion is very desirable from the perspective of propagating realistic uncertainties The structure of the DSST is very amenable to parallelization via modern day computing hardware and software techniques The present study analyzes the DSST for the purpose of Space Situational Awareness. The study is divided into two parts. The first part deals with the error analysis of a semi-analytical batch least square orbit determination process. The second part of the study deals with uncertainty realism and the propagation of uncertainties over time using the DSST. Uncertainty realism for SSA requires a proper characterization of a space object's full state probability density function (pdf) in order to faithfully represent the statistical errors. DSST covariance propagation using the unscented transformation is discussed.

Optimization and Dynamics (II) / 145

Development, validation and test of optical based algorithms for autonomous planetary landing

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In recent years, a renewed interest in space exploration has induced investing a growing amount of human and financial resources to provide next generation spacecraft with enhanced autonomous navigation and landing capabilities. Complex missions in which close approach to and landing on uncooperative objects play a major role are being developed by numerous space agencies: in particular, ESA is working together with ROSCOSMOS on a cooperative programme for Mars, Phobos and Moon exploration: as our satellite is concerned Luna-Resurs Lander (Luna-27) mission planned for 2020 and the Luna 29, the Lunar Sample Return mission to follow are involved, strongly focused on the South Pole landing to collect icy volatiles located in a very precise region of the huge Aitken crater. Part of the European contribution for the Luna-27 mission is the PILOT (Precise and Intelligent Landing using Onboard Technologies) subsystem, for enhancing autonomous landing capabilities in terms of high precision landing and hazard detection and avoidance functionalities. An analogous collaboration has been established for the ExoMars programme, which include a lander delivery first, followed by a rover release on Mars, to be launched in 2016 and 2018 respectively and for the exploration of the small Mars moon with the Phootprint Mission, a Phobos Sample Return mission planned for the Twenties.
Since 2006, technologies for autonomous landing are studied by NASA in the frame of the Autonomous Landing Hazard Avoidance Technology (ALHAT) program. ALHAT technologies, tested at the end of 2014 in free flight on the Morpheus lander demonstrator, are going to be integrated in future exploration missions. Recently CNSA has performed its first lunar landing with a lander/rover system with the Chang'e 3 mission (with missions 4 and 5 already scheduled), while ISRO is planning to put a lander carrying a rover on the lunar surface by the early 2018 in the Chandrayaan 2 mission. Among the various technologies under study, vision-based systems represent one of the most promising tools to provide the required level of accuracy, unattainable by classical technologies. In this paper the research carried out at the Department of Aerospace Science and Technology (DAER) of Politecnico di Milano about algorithms and tools dedicated to autonomous navigation and hazard detection is presented.
A hazard detection algorithm, based on artificial neural networks, has been developed and extensively tested. A single grayscale image of the landing area is filtered to extract essential information regarding shadows, surface roughness and slopes, and then it is analyzed by a cascade neural network, which provides a hazard map. Hence, hazard information is exploited by a subroutine that computes the most suitable landing site, taking into account safety requirements and trajectory constraints. The achieved computational efficiency allows the system to operate real time.
Also a vision-based relative navigation tool, relying on features tracking between subsequent frames, is currently under development. Camera information is fused together with measures coming from classical sensors, like Inertial Measurement Units and radar/laser altimeters. Collected data are filtered, taking into account the lander dynamics to obtain a convergent estimate of the system

status. Collected information are also exploited to build a dynamical semi-dense map of the landing area, exploitable by hazard detection to estimate slopes and to locate the selected target. To further increase the TRL of the aforementioned algorithms, an experimental facility is under setup at DAER premises. Since the scarce availability of complete real landing imagery datasets, vision-based algorithms development relies widely on synthetic images. To validate such approach, experiments are necessary. Moreover, the whole navigation system performance can be assessed only connecting the composing parts together, to verify mutual influences. The experimental facility under development represents a Hardware-in-the-Loop environment, and it is composed by a 3D mock-up simulating the planetary surface; a 7 DoF robotic arm to carry the sensors suite and simulate the spacecraft dynamics; an illumination system consisting of a LED array and a dimming subsystem to provide a realistic and controllable light. Hence the aim is to simulate the landing maneuver with the robotic arm in a scaled environment with realistic illumination conditions, over a reproduced planetary surface. The system is designed to verify either hardware and software breadboards up to TRL 4, with possible further enhancements to qualify flight models to TRL 5. The development status of the vision-based tools is presented. The design and the first activities for functional verification of facility components are shown, such as first tests of the hazard detector in a lunar environment.

Low Thrust (II) / 142

Low-Thrust Transfers from Distant Retrograde Orbits to L2 Halo Orbits in the Earth-Moon System

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This paper presents a study of transfers between distant retrograde orbits (DRO's) and L2 halo orbits in the Earth-Moon system that could be flown by a spacecraft with solar electric propulsion (SEP). Pseudo-spectral optimal control is used to optimize these highly non-linear transfers. Similar types of transfers that have been studied in the literature include: from Earth orbit to Moon orbit using low-thrust1, from Earth orbit to libration point orbits using low-thrust1, from Earth to DRO using impulsive maneuvers1, from Earth to DRO using low-thrust1, and from L1 halo orbit to L2 halo orbit. Transfers between DRO's and halo orbits using low-thrust propulsion have not been studied previously.
This paper takes advantage of modern advancements in computer hardware and optimization software to perform a study of the trajectories that could be flown by a low-thrust, SEP spacecraft from a DRO to a halo orbit about L2. This includes fleshing out several families of transfers that exist, identifying the minimum thrust-to-weight ratio required to fly on each family, and providing an understanding of the time-of-flight vs. propellant-mass. Topputo[5] showed that many distinct families of ballistic transfers exist between the Earth and Moon in a four-body model, and others have demonstrated that such variations exist for other types of transfers in Earth-Moon space[6], [7]. By exploring the families of transfers that exist between DRO's and L2 halo orbits, this paper provides deeper insights into the trade space available. The circular restricted 3-body problem (CRTBP) is used throughout the paper so that the results are autonomous and simpler to understand.
DRO's are a type of orbit that have received increased attention in the past few years because of the unique characteristics they exhibit. DRO's are a type of repeating orbit that exists only in the 3-body problem[8]. When viewed in a reference frame that rotates with the orbits of the primary and secondary bodies, a DRO is retrograde about the secondary body, at a relatively high altitude such that the orbit is significantly perturbed by both the primary and secondary bodies. DRO's are unique in that they sit between two-body orbits and libration point orbits in terms of stability. These orbits are often dynamically stable, though it has been shown that perturbations in a high-fidelity model of the solar system may cause a spacecraft to depart an otherwise stable DRO[9]. Parker, Bezrouk, and Davis demonstrated several trajectories that transfer from Earth to a DRO, requiring no maneuvers and remaining on the DRO for thousands of years3.

Mission concepts that have examined DRO's include the proposed NASA/JPL ARM (Asteroid Redirect Mission)[10] and the Orion/MoonRise concept[11], [12]. Both of these mission concepts would benefit from the capability to transfer between a DRO and a halo orbit about L2. Ongoing research by Davis and Parker is finding that impulsive transfers between those orbits do exist, but they are costly on the order of 150 m/s and require transfer times on the order of weeks to months. Spacecraft with SEP have the potential to greatly reduce the propellant mass required to make such transfers, without much increase in time of flight.
The open source, pseudo-spectral optimal control package PSOPT[13] is used to optimize transfers in the CRTBP. A variety of families of solutions are discovered by seeding different initial guesses, and by then using the continuation method to discover similar transfers. An example transfer optimized using PSOPT is presented in Fig. 1, below. The new software package Maverick[14], developed at CU Boulder, is also used to compare solutions.
[*See attachment for Figure*]. An example transfer viewed in the Earth-Moon synodic reference frame. Dynamics are the CRTBP. The Moon is plotted to scale on the x-axis at (1-μ), and the Earth would appear at (-μ). This transfer has a thrust-to-mass ratio of 1.3E-4 N/kg and a transfer time of 47 days. Thrust is nearly always on for this example. 1.75% of the initial mass is used as propellant. Thrust vectors are plotted only in the bottom-right plot.

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Open Source (I) / 143

Scilab open-source modeling & simulation platform

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"Share your knowledge. It is a way to achieve immortality." - the Dalai Lama.

Scientists and Engineers worldwide leverage platforms to model & simulate, to expand knowledge and to collaborate. Open platforms have a key advantage in that they allow a wider and more open collaboration.
Scilab is a powerful open-source modeling and simulation software platform installed every month by more than 100,000 engineers and scientists from around the world.
The goal of this presentation is :

- to introduce Scilab, its history and its governance
- to showcase some applications of Scilab in the Aerospace industry (CNES, Airbus DS, ESA)
- to demonstrate the functional capabilities of the Software

In addition to the Celestlab flight dynamics application developed by CNES, other Scilab-based applications will be covered :

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- Sizelab: An Airbus DS application for the mechanical pre-sizing of Ariane 6 launcher
- The Aerospace Blockset developed during the ESA Summer Of Code In Space

Creating knowledge, converting knowledge into applied research and bringing applied research to the industry through sustainable commercial projects is accelerated by connecting these different pools of talent:

- academic experts and students in Universities and Engineering schools
- scientific experts in research centers (public and private)
- research and development teams in industries and companies

The vision we want to share and convey through Scilab and the open-source software is the power of connecting different and complementary talents.
Innovators, inventors and entrepreneurs ultimately prefer to build on open platforms because Open platforms are able to create the most value in the long run.

Satellite Constellations and Formations / 140

Spacecraft formation control using analytical integration of Gauss variational equations

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This paper derives a control concept for far range Formation Flight (FF) applications assuming circular reference orbits. The paper focuses on a general impulsive control concept for FF which is then extended to the more realistic case of non-impulsive thrust maneuvers. The control concept uses a description of the FF in relative orbital elements (ROE) instead of the classical Cartesian description since the ROE provide a direct access to key aspects of the relative motion and is particularly suitable for relative orbit control purposes and collision avoidance analyses.
Although Gauss' planetary equations have been first derived to offer a mathematical tool for processing orbit perturbations, they are suitable for several different applications. If the perturbation acceleration is due to a control thrust, Gauss' variational equations show what effect such a control thrust would have on the keplerian orbital elements. Integrating the Gauss' variational equations offers a direct relation between velocity increments in the local vertical local horizontal (LVLH) frame and the subsequent change of keplerian orbital elements.

For proximity operations, these equations can be generalized from describing the motion of single spacecraft to the description of the relative motion of two spacecraft. This will be shown for impulsive and finite-duration maneuvers. Based on that, an analytical tool to estimate the error induced through impulsive planning is presented. The resulting control schemes are simple and effective and thus also suitable for on-board implementation. Simulation results show that the proposed concept improves the timing of the thrust maneuver executions and thus reduces the residual error of the formation control.

Orbit Determination and Prediction Techniques (I) / 177

MONTE: The Next Generation of Mission Design and Navigation Software

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MONTE (Mission Design and Operations Navigation Toolkit Environment) is an astrodynamic toolkit produced by the Mission Design and Navigation Software Group at the Jet Propulsion Laboratory. It supports operational orbit determination and flight path control for deep space and Earth orbiting flight missions, as well as providing an array of tools that can be used in mission design and analysis. Starting with initial development in 1998, it had a primary goal to encompass into a single software system the capabilities of a large suiteof software including the highly successful DPTRAJ/ODP. Since 2009, the Masar project has funded Monte to include development of mission designcapabilities including trajectory optimization and analysis as well as 3d visualization. It was first used in flight operations starting with the launch of Phoenix to Mars (2007) and currently is the prime Orbit Determination software for all JPL missions. Monte has also been used to support missions from ESA, JAXA, and ISRO. The mission design capabilities are being used for the design of future missions including Europa/Clipper, Mars 2020, and InSight. Monte has also been used for non-operations tasks including gravity analysis and satellite ephemeris estimation.
Monte is presented to the user as an importable Python-language module. This allows a simple but powerful user interface via CLUI or script. In addition, the Python interface allows Monte to be used seamlessly with other canonical scientific programming tools such as SciPy, NumPy, and matplotlib. This paper gives an overview of the Monte system, a history of its successes to date, and a preview of what's to come for the software.

Rendezvous & Docking (II) / 148

Dynamical analysis of rendezvous and docking with very large space infrastructures in non-Keplerian orbits

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The idea of building a space station in the vicinity of the Moon as a gateway for future human exploration of the Solar System is being investigated since many years, and recently the attention of scientific community has become even more intense. The natural location for a space system of this kind is about one of the Earth-Moon libration points, in particular EML1 or EML2. Therefore, the entire analysis has to take into account the non-Keplerian dynamics that regulates the orbital motion in these environments.
At the current level of study, the final configuration of the entire system is still to be defined. However, it is already clear that in order to assemble the structure several rendezvous and docking activities will be carried out, many of which to be completely automated. In addition, numerous proximity manoeuvres will be held along all the nominal lifetime of the space infrastructure. In

non-Keplerian orbits the motion is completely different from the one in LEO, since the effects of three-body problem are not negligible, and various problems still deserve particular attention despite the knowledge about autonomous approaching operations in space, acquired with the experience of the International Space Station.
In this paper, the dynamics of very large space structures in non-Keplerian orbits is analysed, taking into account the flexibility of the system and the coupling effects between the modes of the structure and those related with the orbital motion. The results are then exploited to have a sensitivity analysis about the different families of non-Keplerian orbits as a function of the possible configurations of the space station with respect to the numerous rendezvous and docking manoeuvres to be facilitated: the dynamical stability of the system can be evaluated and assessed with respect to safe rendezvous and docking manoeuvres design between robotic vehicles and orbiting infrastructure.
A Multi-Body approach is here preferred, but the flexibility of the system is included both with a lumped parameters and with a distributed parameters technique. The results obtained with the two different approaches are compared, analysing the precision of the results and the computational time that is required to perform the computations. The configuration and the parameters of the large space structures are fully parametrized and the model is maintained as generic as possible, in a way to delineate a global scenario of the mission. However, the developed model can be tuned and updated according to the information that will be available in the future, when the system will be defined with a higher level of precision.
The results are critically presented with respect to the proximity manoeuvring complexity and required resource budgets for some reference scenarios.

Students (II) / 149

Efficient numerical propagation of planetary close encounters with regularized element methods

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In Solar System dynamics, a close encounter with a major body is the only natural phenomenon capable of modifying the orbital elements of a body on a very short timescale. If not properly taken into account during orbit propagation close encounters may heavily degrade the quality of a solution, or even completely compromise it. When numerically integrating the equations of motion of the body in a heliocentric reference frame, a close encounter will introduce an impulsive perturbation which has to be dealt with either by decreasing the step size close to the perturbing body or by some other device. Also, a close encounter introduces *gravitational scattering*: trajectories which are close before the close encounter may diverge afterwards due to different post-encounter major axes and therefore different orbital periods 1.
The accumulation of numerical error can be reduced by integrating regularized equations of motion, whose characteristics are more advantageous for numerical integrations. When perturbations are weak it is especially convenient to integrate regularized equations which describe the variation of orbital elements, since they will follow an almost-linear behaviour 2. At first, this fact seems to rule out orbital elements for the propagation of close encounters, which by definition introduce a strong perturbation in heliocentric orbits. Here, we circumvent this problem by switching the primary body from the Sun to the planet during the close encounter phase, along the lines of the patched conics method of preliminary orbit design 2. Thus, the propagation is split in three weakly-perturbed legs: heliocentric pre-encounter, planetocentric, and heliocentric post-encounter.
A particularly delicate aspect is the definition of the point at which to switch primary bodies during the propagation. Ideally, this has to be chosen according to a criterion which minimizes the final propagation error and the computational cost.

We tested our approach by performing large-scale numerical simulations of close encounters in the planar Sun-Earth CR3BP. Each simulation is parametrized in the conditions at the point of minimum approach distance. The propagation performance is evaluated for encounters taking place in a wide range of asymptotic velocities, and for different kinds of heliocentric orbits. As an additional parameter we choose the geocentric distance at which the switch between the dynamics is executed, as to study the influence of the switch point on the propagation efficiency. We employ different formulations of the Dromo family of element methods [4, 5], and we compare them against the integration of the equations of motion in Cartesian coordinates (Cowell's method) and the Kustaanheimo-Stiefel method 2. The integrator used is an implicit multistep with variable step size and order, which automatically alternates between Adams-Bashforth-Moulton and BDF numerical schemes [6]. For each propagation, the accuracy is estimated with respect to a reference solution computed in quadruple precision using Cowell's method, while the computational effort is measured by the number of calls to the right-hand side of the equations.
Adopting regularized element methods and switching the primary bodies increases propagation efficiency, especially for relatively low minimum approach distances. Figure 1 depicts results for close encounters with a minimum approach distance of 5.03 Earth radii. Even with a sophisticated integration scheme with variable step size and order, regularized element methods guarantee a gain of up to three orders of magnitude in accuracy for about the same computational cost as Cowell's method. Varying the geocentric distance at which the dynamics are switched does not have an effect on the final propagation error, but it does influence the number of function calls. An optimal range of switch distances exists in which the function calls reach a minimum. Preliminary tests with different integrators for a limited set of initial conditions have shown that the existence and extension of an optimal switch distance range depend on the characteristics of the numerical scheme.
Work which is currently being carried out includes a comprehensive study aimed at the definition of a criterion for switching between heliocentric and planetocentric dynamics, with the objective of maximizing the propagation efficiency with regularized element methods. The simulations will be extended and re-parametrized for the 3D case, and the propagation efficiency will be estimated for case studies modelled on objects of particular significance for Space Situational Awareness activities, such as 99942 Apophis. Open-source software tools for the propagation of close encounters with regularized element methods will be made available through an online repository.

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Environment Modelling / 77

Using the attitude response of aerostable spacecraft to determine thermospheric wind

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The residual atmospheric density present at orbital altitudes produces an aerodynamic disturbance, both to satellite orbits and attitude, that at Very Low Earth Orbits (<450 km) can be significant, usually being the strongest disturbance. Certain spacecraft shapes are considered to be aerostable as a restoring aerodynamic torque appears when the spacecraft aerodynamic equilibrium attitude - where no aerodynamic torque is produced - is lost. The equilibrium attitude is defined with respect to the relative flow which is composed of the spacecraft's inertial velocity, the atmospheric co-rotation and the atmospheric wind. A cone is a simple example of an aerostable shape, with its aerodynamic equilibrium attitude achieved when the cone's axis of symmetry is aligned with the relative flow.

The proposed method is able to measure the atmospheric wind by observing the spacecraft's attitude motion when it is allowed to freely react to the aerodynamic torques caused by the relative flow. Estimates of the other attitude disturbance sources are required to isolate the response cause by the aerodynamic torque and knowledge of the spacecraft's aerodynamic properties and atmospheric density is needed to determine the contribution from the wind magnitude and direction in the observed aerodynamic torque.

Aerostable spacecraft behave as an undamped oscillators with their natural frequency depending on the dynamic pressure and their aerostable properties (i.e. the aerodynamic stiffness). If the spacecraft aerostability is strong enough the attitude motion will remain a bounded oscillation around the aerodynamic equilibrium point and thus no additional control input is required. The natural frequency of the system along with the velocity of the spacecraft, determines the achievable spatial resolution of the wind measurements. As the atmospheric density increases exponentially with decreasing altitudes, improved spatial resolutions at lower altitudes are achieved by using the same aerostable properties. A high spacecraft inertia to aerodynamic stiffness ratio also increases the natural frequency and implies that small spacecraft (with high area per unit of mass) are better suited to be used in the proposed method.

Aerostable spacecraft only provide an aerodynamic torque that is normal to the relative flow direction (i.e. pitch and yaw). So the cross-track wind components have a stronger effect on the spacecraft's attitude than its in-track wind counterpart. The dynamic pressure is a function of the in-track wind and thus the in-track wind component is also observable although it has a much weaker effect and thus it imposes more stringent requirements.

The method described in this paper can provide global cross-track and in-track wind measurements. The measurements accuracy and spatial resolution with respect to the system parameters (i.e. altitude, aerostable properties and its uncertainty, inertia and uncertainty on the knowledge of atmospheric density) are also analyzed.

Rendezvous & Docking (I) / 76

An open-source simulator for spacecraft robotic arm dynamic modeling and control.

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A wide range of space missions require a robotic arm (e.g. satellite servicing, active debris removal and berthing). The kinematics and dynamics of space manipulators are highly non-linear and differ considerably from their terrestrial counterparts. The base-spacecraft is not anchored to the ground and thus it is free to react to the manipulator's motion, making the modeling and control of space-based manipulators a complex task. The base-manipulator interaction is stronger when the mass and inertia of the manipulator and of the base-spacecraft are comparable. Therefore, the difference between space and terrestrial manipulators tends to become more acute on manipulators mounted on small spacecraft.

There is a wealth of literature tackling the subject of spacecraft manipulators modeling and control but each research group has typically developed its own code in order to simulate and validate control approaches. In an attempt to help speed the process and make space manipulators

a more accessible research topic an open-source kinematics, dynamics and control simulator for space based robotic arms has been developed.
The simulator is written in MATLAB/ Simulink and it can be used for standalone MATLAB scripts and Simulink models. The Simulink models can subsequently be used for automatic code-generation and compiled to run in real-time on the selected target hardware. This open-source code is thus suitable from prototyping work all the way to hardware implementation.
The six degrees-of-freedom simulator is capable of computing the homogeneous coordinate transformation matrices, the velocity Jacobians, the combined inertia matrices and the velocity terms (using the Lagrangian approach) as well as the joint and base reaction using the Newton-Euler approach. Several control manipulator and base-spacecraft control approaches have also already been implemented.
The architecture and usage of the simulator will be presented as well as some examples that demonstrate how the simulator performs under some basic manipulator control applications. Finally an implementation example, on a real-time embedded hardware on an experimental test-bed, will be provided.

Verification and Validation Methods / 75

Design and parameter identification by laboratory experiments of a prototype modular robotic arm for orbiting spacecraft applications

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This paper describes the design and the parameter identification procedure of a modular spacecraft robotic arm that combined with appropriate ancillary equipment (base-spacecraft, target and end-effector) provides an experimental set-up where control approaches and whole mission scenarios (e.g. servicing and debris removal) can be validated and demonstrated.
The originality of the here studied prototype robotic arm consists in its modularity. Each manipulator link contains its own power system, communications, harmonic drive motor with controller, torque sensor and a computing platform. The structure of the links has been manufactured using additive manufacturing allowing to quickly and inexpensively generate links of different lengths (with different mass and inertia properties). The manipulator links are modular and easy to re-arrange to meet the requirements of a particular experiments (number of links and length of these links).
A spacecraft equipped with a robotic arm is required to fulfill a wide range of missions (e.g servicing and debris removal) and few spacecraft with robotic arms have already been successfully flown. The dynamics of a space manipulator are substantially different from its terrestrial counterparts as the base-spacecraft is free to react to the manipulator motion. When the base-spacecraft mass and inertia are comparable to the manipulator's ones (i.e. as it is the case for small base-spacecraft) the base-spacecraft reaction can be significant and can not be safely omitted when modeling or treated as a small disturbance during control.
A substantial amount of theoretical and simulation work has been previously conducted by many researchers to tackle the operation and control of a space manipulator and its base-spacecraft. The difficulty to recreate the conditions of a space manipulator mounted on a small spacecraft on the ground limits the availability of validation experiments on control approaches and dynamical modeling.
A prototype modular robotic manipulator consisting of three links with three rotational degrees-of-freedom has been designed and integrated. Each link has a length of ˜40 cm and has a mass of ˜2 kg. This robotic manipulator is mounted onboard a ˜10 kg Spacecraft Simulator that floats via air-pads over of a 4-by-4 meter granite monolith recreating in two dimensions the reduced gravity and quasi-friction-less environment of space. As it operates on a granite table the movement of both the manipulator and the base-spacecraft are restricted to two translational and one

rotational degrees-of-freedom (planar movement). The base-spacecraft is equipped with eight cold-gas thruster and a reaction wheel and thus is able to control its position and attitude.
As the manipulator can be re-arranged in a different number of configurations, a method to quickly identify the manipulator parameters (mainly mass, inertia and the Denavit-Hartenberg parameters) will be presented. Finally, a few basic manipulator and base-spacecraft control capabilities of the experimental set-up are demonstrated.

Open Source (II) / 74

Rugged: an open-source sensor-to-terrain mapping tool

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The Sentinel-2 mission is a component of the Copernicus program. It consists of two spacecrafts each carrying a high resolution multispectral imager (13 bands) devoted to environmental, security, and agricultural applications. One of the key feature of Sentinel-2 is the huge amount of imagery data that will be produced, as each spacecraft produces 1.7TB of raw data daily. All data are transformed on ground at the Image Processing Facility to create the various products levels.
One element of this processing chain is the Rugged library: an open-source library built on top of the open-source Orekit space flight dynamics library. Rugged is used to compute very quickly and accurately the mapping between ground points and on-board pixels, taking into account a Digital Elevation Model. Direct location computation is used to identify which ground point is seen by a specified sensor pixel. Inverse location computation is used to identify which sensor pixel will see a specified ground point. These methods are the basic elements for complete processing algorithms. They are computationally intensive due to the very large number of pixels to manage (12 detectors, 13 bands, global coverage of land surfaces).
The geo-location methods are at the boundary between image processing and flight dynamics as they handle accurate geometrical models. Rugged has therefore been designed as an intermediate level library, and it relies on Orekit to compute the global geometry (spacecraft orbit and attitude, Earth precession nutation and proper rotation including all IERS corrections, Earth mean ellipsoidal shape). The Rugged library adds on top of this the Digital Elevation Model intersection computation.
The presentation will describe the overall Rugged technical architecture, the issues that were faced and how they were solved in order to achieve a high performance level while not compromising accuracy. It will also present the open-source strategy and the governance model of the library. The library is already operationally used, some perspectives for it will be discussed.

Optimization and Dynamics (I) / 73

from low level toolbox to orbit determination: handling users requests in Orekit

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Orekit is a core space flight dynamics library published as free software under the terms of the permissive Apache Software License V2. Since its inception in 2002, Orekit was designed as a low level layer providing the foundation objects for operational applications. From the very beginning, the foundation objects included time, frames, orbits, but also attitude, maneuvers and a rich framework for state propagation with continuous output and on the fly detection of discrete events. As new versions were published, this set of features has been extended, with new

propagators (the semi-analytical DSST being a major example), new predefined events (16 as of end 2015 and counting), new frames and various improvements.
As an open-source project, Orekit is essentially driven by its users requests and contributions. Looking back at the project evolution, one notices that after the initial stabilization phase during which the core features were completed, users requests led to introduce more and more intermediate features. These features where more mission or operation-oriented, showing us the library was used in various contexts for real problems solving. Many features added in the last two or three versions were really not envisaged at project start. These features clearly show the benefits we get from an open community. Some users have a problem to solve that first appears to be really mission-specific, but often as they notify the project about it, it appears more general than expected and can benefit other users after some reformulation. One typical example is the ellipsoid tessellation. This strange feature was needed for one project in early 2015, but just one month after its design, a second project raised a similar need and a few weeks later an independent user opened a feature request on the same topic.
Some other features have been on our plans for a few years without being implemented, both because of lacking resources to do the job and because they were considered at the boundary of Orekit scope. This was the status of orbit determination. Here again, as more and more users were demanding for it, we finally embarked on it and added it.
This presentation focus on how an open-source project can interact with its users while still maintaining a general orientation, using some examples from the last few releases of the library, up to the latest addition of orbit determination.

Coffee break / Poster Session / Booth Exhibition / 72

open-source publication: a strategic choice for private companies

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Open-source has become a mainstream asset in most parts of today's economy, including space. It's use is widespread and in every projects it is customary to look what existing open-source tools could be used as building blocks. Both private companies and public organizations are heavy users of open-source. The benefits are well-known by now (reliability, vendor-neutrality, standardization, maintenability, ...). Creating an open-source project on the other hand is quite a different topic. For the entity that does the initial work, what could be the incentive to publish freely some cutting edge know-how? If it is a private company in a highly competitive environment and it has to pay for the development, what could be the rationale behind this move?
This presentation relates such a story. It describes the history of the Orekit project, from its inception as an internal tool in a private company, CS-SI, up to its wide adoption in the space domain by now. It explains the reasons for the move to open-source and the rationale behind the license selection. We dicuss the expectations at project publications, what worked and what did not work. We describe the various phases from closed source to full open governance with a meritocratic model. At the end, we assess the current status of the project (spoiler: it is a success).

Debris, Safety and Awareness (II) / 71

A Series for the Collision Probability in the Short-Encounter Model

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The increase of conjunctions between active satellites in Low Earth Orbit (LEO) and other objects, either space debris or other satellites, has made necessary to evaluate the risk posed by these conjunctions in order to decide if evasive maneuvering is needed. The calculation of the collision probability between a pair of objects must done by a precise and fast algorithm because of the enormous and growing LEO population and because numerical methods for collision avoidance maneuver optimization may need to evaluate this probability several times during its execution. In this article, a series to compute the collision probability of two spheres under the assumptions of short-term encounter, which generally hold in LEO, has been derived. It is valid for both Gaussian and non-Gaussian distributions of the position of the spheres, and in the particular case of a Gaussian distribution the use of Hermite polynomials yields a simple form for the series. The parameters that appear in our formula, or in others which address the same issue, are the axes of the projection of some incertitude ellipsoid on the collision plane and the projection on the same plane of the relative coordinates of the objects which might collide. A region of practical interest in this parameter space has been carefully defined based on satellites' real data, and a representative sampling set was chosen. On this sampling set a comparison between the new series and previous algorithms has been performed for the Gaussian case to measure the performance of the proposed method. The presented series is found to be faster than any other algorithm in every explored case. Numerical evidence suggests that if the series for the Gaussian case is truncated when the last term is smaller than the computed probability times a tolerance of 0.1, then the last term is an upper bound for the error. This article also presents very strong evidence for the case that the first two terms of the series are sufficient for the computation of the probability of collision, and that the absolute value of its second term is an upper bound for the error made when using it.

Students (I) / 70

Orbit prediction of high eccentricity satellites using KS elements

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Highly elliptical Earth satellite orbits are used for communication services, geostationary transfer and space and planetary exploration missions. A major requirement for the efficient mission planning is the precise computation of orbit ephemeris. The orbit evolution mainly pertains to the complex interaction of the Earth's oblateness, Atmospheric drag and luni-solar gravity, which cause variation in the perigee altitude leading to increase or decrease in the satellite lifetime. Classical Newtonian equations yield singularity at the collision of two bodies, which render them unstable for long-term orbit propagation. KS total energy element equations are less sensitive to round off and truncation errors in the numerical integration algorithm and offer a very powerful method to obtain numerical solution with respect to a complex force model. Better accuracy is obtained for in-orbit computations since an orbital frequency is based on total energy and the equations are regular everywhere in the unperturbed case. The equations are smoothed for eccentric orbits because eccentric anomaly is an independent variable.
This paper concerns with the development of an orbit prediction package for high eccentricity satellites using KS elements. Plataforma Solar de Almería (PSA) algorithm and a Fourier series algorithm are used to obtain the accurate position vectors of the Sun and the Moon, respectively. An oblate diurnally varying atmospheric model for drag and zonal harmonic terms up to J6 for oblateness, are considered. Orbit computations for few test cases are carried out using a fourth-order Runge-Kutta-Gill method and the resulting orbital parameters are found to be satisfactory when compared with the observed values. Due to low memory requirement, this package can be used in multiple applications including on-board navigation and guidance software.

Coffee break / Poster Session / Booth Exhibition / 79

Mercury rotational state estimation applied to the Bepi-Colombo Mission: an optimized approach on the selection of optical observations

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The BepiColombo mission is finally approaching the last phase, preceding its launch, which will bring the spacecraft to its final destination, planet Mercury, around which the probe will start its one year nominal mission in 2024. In-orbit investigations are a privileged condition for exploring a planet, and Mercury has demonstrated to be a fascinating destination since Mariner 10 first unveiled some of its mysteries. MESSENGER disclosed several questions but also arose other interrogatives and much is still to be discovered. One of the objectives of the BepiColombo Radio science Experiment is the study of Mercury's core. Since a strict connection exists between core and rotational state, measurements of Mercury's obliquity and librations at unprecedented accuracies became one the main purposes of MORE (Mercury Orbiter Radio science Experiment) rotation experiment. The rotation experiment will be accomplished thanks to the cooperation among different payloads: precise orbit determination data will be derived from MORE and high resolution images will be provided by HRIC, part of the SYMBIO-SYS payload. An end-to-end simulator has been built up employing the camera images as the primary observables with the final aim of defining their optimal acquisition scheduling. In this simulator the images are employed to correlate surface landmarks extrapolated by pictures of the same area taken at different epochs: their displacement in time represents the observable to be fed into an estimation process for deriving Mercury's rotation parameters. An extensive simulation campaign has been performed leading to the identification of the most favorable observational strategy and location of the landmarks on the surface so as to fulfill accuracies lower than 1 arcsecond for both obliquity and libration estimation. Concurrently an Orbit Determination simulation scenario for MPO has been set, in which high fidelity, but still evolving, models are implemented to consider all possible degradation effects. The setup consider an on-board accelerometer error model, simulating ISA instrument, errors in the radiometric and optical observables and a full gravity field implementation to degree and order 30 and tidal love number k_2. A pure multiarc orbit determination filter provides the simulated trajectory, the computation of simulated observables and then estimates the whole global and local parameters in the play providing direct insight on mission performances. This architecture has been exploited for the evaluation of the difference in performance by using a random generated set of observation locations or an optimized set for optical observables focusing in the achievable uncertainty in the orientation parameters, in particular for what concern the Mercury obliquity ε and amplitude in longitudinal libration Φ_0 validating the optimization approach.

Optimization and Dynamics (II) / 78

FALCON.m – The free and fast optimal control tool for MATLAB

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No matter if ascent trajectories are to be determined, satellites orbits need to be changed, docking maneuvers must be performed or space debris needs to be collected – in all of those applications optimal trajectories are sought. These examples may be stated as optimal control problems, containing a dynamical model which describes the system behavior and a set of equality and inequality constraints. The optimization is subject to a cost function, which may be minimal energy, minimal time, or any other desired cost function. Since the structure of the optimal

control problem remains the same, the constraints, the cost function, and the dynamic model can be changed easily.
In the past, these problems have often been solved using simplifying assumptions and indirect optimal control methods. Today, with the availability of more powerful computers and advancements in numerical optimization theory, direct optimal control has become increasingly important.
The Institute of Flight System Dynamics at the Technische Universität München has developed a new optimal control tool for MATLAB called FALCON.m. The tool is implemented in MATLAB which allows the user to define all required equations within the popular environment. Additionally, exceptional performance is achieved by using automatic analytic derivatives and automatic MEX compilation with multi-threading for the heavy duty parts of the calculations.
FALCON.m implements a direct collocation method for discretizing the optimal control problem and allows the use of any collocation scheme – such as Trapezoidal, Hermite Simpson, or classical explicit Runge Kutta. The resulting numeric parameter optimization problems are solved using state of the art solvers such as IPOPT, SNOPT or WORHP.
The aforementioned solvers use gradient based algorithms, requiring the gradient of the full discretized problem. Its calculation is crucial for the overall performance, in terms of calculation time and convergence robustness. FALCON.m calculates the sparsity of the gradient and utilizes the Symbolic Math Toolbox and MATLAB Coder Toolbox to efficiently and automatically calculate the analytic gradients for dynamic models, constraints, and cost functions. The subsystem derivative approach used by FALCON.m enables a highly performant source code transformation even for large and complex models.
User friendliness is a major focus of FALCON.m. As the tool is delivered as a library of MATLAB classes, a clear information flow allows for a distinct problem formulation. Additionally, checks for common mistakes in the problem formulation are performed, mitigating errors that may be hard to find by hand.
FALCON.m is a state of the art optimal control tool that is available for free. It has been designed for real world applications and an easy access to the world of optimal control. The power of FALCON.m as well as the variety of its features will be demonstrated in this paper using the example of a space application problem.

Ascent (II) / 41

Optimal Launch Strategies in the Presence of Competition

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Launch decisions are crucial in a space mission. In case of problems (technical ones, bad weather, ...) the planned launch has to be postponed to subsequent dates. Dynamic strategies, based on Bellman's dynamic programming, may help to make good launch decisions. The situation is a stopping problem, related to quantitative versions of the classic secretary problem.
A special situation is a race between two or more competitors (like "US Air Force (Vanguard) vs. US Army (Explorer)" in 1957/58 for the first US satellite; like "US (Apollo 8) vs USSR (Zond 6, ...)" in the last quarter of 1968 for the first manned flight around the moon; or currently "Google's Lunar X Prize" with several competitors) where only the first successful mission is highly rewarded.
We compute optimal launch strategies for situations with and without competition (1, 2 to 5 teams) and discuss them in comparison. It turns out that optimal strategies involve taking more and more risk with increasing numbers of competitors.
We are well aware that the approach in this talk is not a standard one in the design of space missions: instead of technical details we discuss game-theoretic scenarii with stochastic aspects. In doing so, readiness of a mission for launch is measured as a "simple" probability, i.e. as a number between 0 and 1.
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Low Thrust (I) / 47

About Combining Tisserand Graph Gravity-Assist Sequencing with Low-Thrust Trajectory Optimization

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Gravity-assist maneuvers have the potential to be mission enablers, due to “"free energy"” they provide. The efficiency of low-thrust propulsion is further one means of improving mission payload mass. Combining both for a given mission possibly improves overall mission performance, which makes it desirable to investigate low-thrust gravity-assist missions.
For means of investigating a broad range of mission options, the System Analysis Space Segment group of DLR is working on methods of combining the optimization of low-thrust trajectories and gravity-assist sequences with the help of the Tisserand Criterion and shape-based trajectory models. The hurdles faced by violations of Tisserand Criterion premises are shortly discussed and the repercussions these have on planning a gravity-assist sequence for a low-thrust mission. A metholodology, based on benchmarking the results with non-gravity-assist trajectories is presented in this paper, grounded on a loop combining the optimization of the trajectory and the selection of the next gravity-assist partner. Furthermore it is shown how the solution space can be reduced with the help of constraints originating in the maximum possible Delta-V gain and the gravity-assist partner pool.”

Loitering / Orbiting (I) / 39

An efficient code to solve the Kepler equation for elliptic and hyperbolic orbits

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The Kepler equation for the elliptical motion, $y - e \sin y - x = 0$, involves a nonlinear function depending on three parameters: the eccentric anomaly $y =$ E, the eccentricity e and the mean anomaly $x =$ M. For given e and x values the numerical solution of the Kepler equation becomes one of the goals of orbit propagation to provide the position of the object orbiting around a body

for some specific time (see references [1–6]). In this paper, a new approach for solving Kepler equation for elliptical and hyperbolic orbits is developed. This new approach takes advantage of the very good behavior of the Laguerre method [7] when the initial seed is close to the looked for solution and also of the existence of symbolic manipulators which facilitates the obtention of polinomial approximations. The central idea is to provide an initial seed as good as we can to the modified Newton-Raphson method, because when the initial guess is close to the solution, the algorithm is fast, reliable and very stable. To determine a good initial seed the domain of the equation is discretized in several intervals and for each one of these intervals a fifth degree interpolating polynomial is introduced. The six coefficients of the polynomial are obtained by requiring six conditions at both ends of the corresponding interval. Thus the real function and the polynomial have equal values at both ends of the interval. Similarly relations are imposed for the two first derivatives. Consequently, given e and $x =$ M, selecting the interval $[x_i, x_{i+1}]$ in such a way that M ϵ $[x_i, x_{i+1}]$ and using the corresponding polynomial $\mathrm{p}_i(x)$, we determine the starter value $\mathrm{y}_o = \mathrm{E}_o$. However, the Kepler equation has a singular behavior when M is small and e close to unity (singular corner). In this case, the exact solution of the equation has to be described in a different way to guarantee the enough accuracy to be part of the seed used to start the numerical method. In order to do that, an asymptotic expansion in power of the small parameter $\varepsilon = 1-e$ is developed. In most of the cases, the seed generated by the Space Dynamics Group at UPM(SDG-code) leads to reach machine error accuracy with the modified Newton-Raphson methods with no iterations or just one iteration. The final algorithm is very stable and reliable. This approach improves the computational time compared with other methods currently in use. The advantage of our approach is its applicability to other problems as for example the Lambert problem for low thrust trajectories.

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A fast and efficient algorithm for onboard LEO intermediary propagation

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Usual onboard orbit propagators are provided as navigation maintenance aids for earth orbiting satellites. These programs should be able to forecast satellite ephemeris within a reasonable accuracy for short time periods, which may range from minutes, as in the case of momentary lack of GPS signal, to several satellite orbits. In these brief intervals the accumulation of second order effects of the Geopotential is barely apparent, and, therefore, the propagation model can be very simple. Hence, common onboard orbit propagators are based in the fixed-step numerical integration of the J2 model, a truncation of the Geopotential limited to the zonal harmonic coefficient of the second degree.

On the other hand, the use of analytical, intermediary solutions of the J2 problem has been recently proposed as an efficient alternative to the numerical integration. The accuracy of common intermediary orbits of the J2 problem is limited to first order effects, thus providing less precise solutions than the numerical integration. However, because of the inherent uncertainty of the initial conditions to be propagated onboard, it can be shown that both alternatives, the numerical integration and the intermediary approach, enjoy the same statistics. Other benefits of using analytical solutions is that they may improve both memory allocation and computation time, a fact that may be crucial to Cubesats ore other small satellite missions, in which the computational abilities may be restricted.
Note, however, that neglecting the long-period effects associated to the odd zonal harmonics introduces small errors in the propagation, which are clearly observable even in the short-term. These errors can exceed 1 km in the along-track direction at the end of one day. Hence, taking into account the disturbing effects of some higher order harmonics may notably improve the propagation model.
Here, we propose a new intermediary solution that takes into account the first three zonal harmonics of the Geopotential (J2, J3 and J4). Since the solution is analytical, the evaluation is very fast and is not constrained to a step-by-step evaluation. In spite of the forces model of the new intermediary is much heavier than the simple J2 model, its evaluation can be fastened using some simplifications that alleviate the computational burden, in this way making the new intermediary definitely competitive when compared to the numerical integration of the J2 problem.

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SIRIUS-DV: The new Flight Dynamics algorithms for the future CNES missions

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The SIRIUS project aims to develop a set of Flight Dynamics products that will be used operationally in the control centers of the upcoming CNES missions. It mainly covers three different layers: the mathematical low level libraries (PATRIUS), intended to be used either in an operational environment or in expert studies; the flight dynamics algorithms, implementing the operational functionalities (SIRIUS-DV); and the FD applications, that include the assembly of the algorithms to build stand-alone applications – with dedicated GUI - and the infrastructure services (such as time, messages, logging, . . .) needed in an operational FDS.
Due to its unique architectural conception, SIRIUS provides a higher level of flexibility (so as to be easily adapted to any future mission, almost in a "plug and play" manner) and scalability (the effort to add new functionalities is reduced) with respect to other state-of-the-art systems. The choice of technologies used in the line of products also guarantees its non-obsolescence up to - at least - twenty years from now.
This paper focuses in the second layer, the software applications implementing the flight dynamics algorithms that are divided in several technical domains:

- Conversion: Dates conversion, orbital/attitude parameters conversion.
- Ephemeris Generation: orbit propagation and ephemeris generation.
- Events: events/phenomena calculation.
- Scenario: Functionalities dealing with the data scenario, which represents the whole mission of a given satellite
- Orbitography: measurement treatment, orbit determination, collision risk assessment.
- Orbital Maneuvers: orbital maneuvers computation, station keeping.

- Guidance & Programming: AOCS programming, guidance, constraints checking.
- Mission: reference orbit calculation.
- Monitoring: monitoring of TM parameters, thresholds checking.
- Interfaces: External data retrieval/production.
- Scenario Processings: treatment of the different parts of the data scenario, such as trajectory, attitude, maneuvers, MCI, thrusters, tanks and solar arrays

These algorithms rely on a data model managed by the CNES domain experts and which is updated gradually as the development advances. It contains the definition of all the data that are used in the algorithms, the definition of the interfaces (inputs/outputs) of each algorithm and the software requirements that the implementations must meet. Using this model as input, the implementations of both the data and algorithms interfaces are automatically generated (using a code generator that is also part of the SIRIUS line of products), which serve as starting point for the development carried out by the team.
The SIRIUS-DV applications are developed in Java using an Agile/SCRUM methodology with sprints (realization iterations) lasting four weeks. The functionalities to be developed in a given sprint are presented (at the beginning of each sprint) to the team by the CNES domain experts. During the sprint a constant communication flow is established between both parties in order to ensure the understanding – and hence the quality – of the tasks to be done (the development team being physically located at CNES premises). At the end of each sprint, those functionalities that are finished are presented to the users by the development team, so a fully usable product is available once a month, with increased functionalities over time.
This paper gives a brief description of the development process of SIRIUS-DV and describes the key concepts of this new line of flight dynamics algorithms, the data model and its impact in the developed software and several of the most representative applications, paying special attention to the architectural design of the propagator and its link with the data scenario, since it constitutes the core of the system.

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Comparison of the Orekit DSST Short-Periodic Motion Model with the GTDS DSST and the F77 DSST Standalone Models

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Development of the DSST started in the mid 1970's at the Computer Sciences Corporation and continued at the Charles Stark Draper Laboratory and the MIT Lincoln Laboratory. These developments employed the non-singular equinoctial elements. The Draper Semi-analytical Satellite Theory used the GTDS orbit determination system as the development platform. However, users external to the Draper Laboratory wanted access to the Semi-analytical Satellite Theory without the 'overhead' of GTDS. The DSST Standalone program was developed in 1983-84. The Standalone included complete models for the mean element motion (based on the conventions then employed in GTDS) and a portion of the short-periodic model. The intent was to provide accuracy for LEO orbits of approximately 200 meters. By 1996, extensive improvements to the GTDS DSST had been made. These included 50 x 50 geopotential fields, solid Earth tides, and J2000 coordinate systems. In 1997, an effort to extensively upgrade the DSST Standalone was undertaken. This effort included improvements to the force modeling and to the maintainability of the source code. While the 1997 upgrade touched large portions of the DSST Standalone source code, testing primarily focused on the mean element equations of motion. In 2010, the first author presented the paper "Open Source Software Suite For Space Situational Awareness And Space Object Catalog Work" at the ICATT meeting in Madrid. This paper proposed the migration of the DSST Standalone Orbit Propagator from Fortran 77 to an Object-Oriented software platform.

In 2011, the implementation of the mean element motion portion of the DSST in the Orekit open source library was initiated. Implementation in the Orekit library involved migration of the DSST to the object-oriented java language and to a different functional decomposition strategy. Resolution of the F77 Standalone DSST code and documentation anomalies was an important product. Orekit DSST mean element predictions were compared with those produced with the F77 DSST Standalone. For several test cases involving several thousand day arcs, the Orekit and F77 mean element histories could not be distinguished. The DSST employs Fourier series for the short-periodic motion in the equinoctial elements. These expressions are closed form for the zonal, lunar-solar, and the tesseral m-daily terms. The Fourier coefficients in these expressions are functions of the slowly-varying mean elements (a, h, k, p, and q) and are slowly varying when plotted over time.1 The Fourier coefficients are computed 'off-grid' via an interpolation process. This Fourier coefficient interpolation must be compatible with both the high order Dormand-Prince and the classical RK integrators employed in Orekit DSST. This paper provides a description of the java classes adopted for the Orekit DSST with emphasis on the short-periodic models and the associated interpolation processes. The Orekit class DSSTForceModel includes the functions getMeanElementRate and getShortPeriodicVariations. This paper provides detailed comparisons of the Fourier coefficients computed by the Orekit DSST, F77 DSST Standalone, and GTDS DSST programs on a perturbation-by-perturbation basis. We also consider the overall accuracy that is possible with the present Orekit DSST implementation.

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A semi-analytical orbit propagator program for Highly Elliptical Orbits

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A semi-analytical orbit propagator to study the long-term evolution of spacecraft in Highly Elliptical Orbits is presented. The perturbation model taken into account includes the gravitational effects produced by the first nine zonal harmonics and the main tesseral harmonics affecting to the 2:1 resonance, which has an impact on Molniya orbit-types, of Earth's gravitational potential, the mass-point approximation for third body perturbations, which on ly include the Legendre polynomial of second order for the sun and the polynomials from second order to sixth order for the moon, solar radiation pressure and atmospheric drag. Hamiltonian formalism is used to model the forces of gravitational nature so as to avoid time-dependence issues the problem is formulated in the extended phase space. The solar radiation pressure is modeled as a potential and included in the Hamiltonian, whereas the atmospheric drag is added as a generalized force. The semi-analytical theory is developed using perturbation techniques based on Lie transforms. Deprit's perturbation algorithm is applied up to the second order of the second zonal harmonics, J2, including Kozay-type terms in the mean elements Hamiltonian to get "centered" elements. The transformation is developed in closed-form of the eccentricity except for tesseral resonances and the coupling between J2 and the moon's disturbing effects are neglected. This paper describes the semi-analytical theory, the semi-analytical orbit propagator program and some of the numerical validations.

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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.
For the DESEO development, reliable and proven algorithms and routines have been used. Moreover, the need of flexibility and modularity drove the design toward an Object-Oriented (OO) architecture design. The OO architecture also reduces maintenance and upgrade costs. The DESEO toolkit, besides the DEIMOS native algorithms, also integrates ESA EO CFIs libraries, providing additional means for performing embedded analyses.
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 currently composed of 38 modules (Analysis Processes) that can be used as stand-alone tools (command line) or operated via a Graphical User Interface (GUI). The Analysis Processes 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). The GUI provides functionalities to manage the input insertion process (e.g. importing data from a database or from other input files), the analysis executions and monitoring (by means of log messages and progress bars) and the output visualisation. The GUI visualisation module is capable of producing 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 and Mac OS X Operating Systems.

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Modelling and Simulation of Autonomous Cubesats for Orbital Debris Mitigation

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It is well known that orbital debris about Earth impose increasingly stringent restrictions on the operation and commissioning of both current and future space applications. These orbital debris, which are becoming ever so prevalent, can literally destroy a satellite. Even particles of diminutive stature can result in disastrous ramifications. Many of these endangered satellites, of which humans are reliant upon for supplying the infrastructure necessary to support modern life in the twenty-first century, routinely have to perform avoidance maneuvers in response to ground data indicating that an object is on a trajectory that could pose a threat, negating away precious finite amounts fuel. These orbital debris are vexatious, with respect to not only a spacecraft's integrity, but also its lifespan. It is imperative that a solution be realized. This study aims to demonstrate the feasibility of utilizing modular cubesats to deorbit space debris. A large high fidelity multidisciplinary simulation is constructed with the goal to simulate the cubesat's orbital and attitude dynamics, as well as its autonomous functions. Such autonomous functions will manifest in the development of autonomous control algorithms to execute mitigation procedures,

such as path planning and rendezvous, as efficiently as possible with respect to multiple subsystems' criteria.

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OCCAM: Optimal Computation of Collision Avoidance Maneuvers

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The continuous growth of the population of objects in Low Earth Orbit (LEO) has caused an increase of the conjunctions between active satellites and other objects, either space debris or other satellites. It is mandatory to evaluate the risk these conjunctions pose, and to design the corresponding collision avoidance maneuvers if necessary. Since several maneuvers are to be performed in the satellite lifetime, the maneuvers should consume as low fuel as possible.
OCCAM (Optimal Computation of Collision Avoidance Maneuvers) is a novel software tool aimed at computing minimum-fuel collision avoidance maneuvers in the short-term encounter scenario, which is generally applicable in LEO. Developed by the Space Dynamics Group of the Technical University of Madrid, it employs advanced modeling and optimization techniques, which make it an extremely fast and robust design tool. OCCAM features an extensive set of input parameters, different optimization strategies and output options to provide a high design flexibility for the user. Several methods of collision probability computation are also supported. Its user-friendly graphical interface and intuitive design logic make it really straightforward to master even for non-experts, and it can be employed either as a standalone tool or in conjunction with other satellite operation planning frameworks.
In an increasingly complex operational scenario, OCCAM does what other collision avoidance planning tools do but in a fraction of their computation time, making it a fast and reliable design and planning tool for the space operators seeking to minimize the cost of their collision avoidance maneuvers.
A trial version of this tool called OCCAM lite is available on-line for the interested potential user at the web page of the Space Dynamics Group: http://sdg.aero.upm.es/index.php/online-apps/occam-lite. The fact that this tool is capable of running in a web-browser (either on workstations or mobile devices such as tablets or smartphones) is a proof of the outstanding velocity of this software.

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Innovative Method for the Computation of Safety Re-entry Area Based on the Probability of Uncertainities in the Input Parameters

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The risk reduction measures required for the reentry of a spacecraft at its end of life are regulated in Europe by requirements documented in Space Agencies' instructions and guidelines. In particular, according to the French law, "the operator responsible of a spacecraft controlled reentry shall identify and compute the impact zones of the spacecraft and its fragments for all controlled

reentry on the Earth with a probability respectively of 99% and 99,999% taking into account the uncertainties associated to the parameters of the reentry trajectories". According to European Space Agency guidelines the Safety Re-entry Area (SRA) delimits the area where the debris should be enclosed with a probability of 99,999%. The computation of SRA is required for a significant number of space missions like spacecraft in low Earth orbits at its end of life and last stages of launchers that shall be controlled to a destructive reentry. This information is crucial to get safety clearance. A similar box, relative to smaller probability level (99%), is required to implement the procedures of warning and alerting the maritime and aeronautic traffic authorities of the concerned countries.

The dynamics of a space object like a spacecraft or a rocket stage entering the atmosphere is quite complicated and quite sensitive to a number of parameters linked to the fragmentation process, to the trajectory models and to the initial conditions of the arc hitting the atmosphere. The computation of the SRA must take into account the uncertainties of those parameters with a satisfactory accuracy or sufficient level of conservatism to ensure that debris will not fall outside the SRA with a probability larger than 0.001%. There are two main modeling aspects for the computation of the SRA: 1) the characterization of the fragmentation and explosion process and of the properties of surviving fragments 2) the computation of the impact points from the trajectory propagation taking into account dynamics model and initial conditions uncertainties. The problem is complicated due to the extremely low probability of interest, which makes quite difficult and inaccurate to use classic statistical techniques and requires to rely on specific extrapolation of the results by fitting distributions tails. A Montecarlo analysis may be performed to estimate the footprint of fragments impacts. The limitation of this method is the number of dispersed fragments impact points to be simulated by the Montecarlo analysis in order to measure the size of the footprint box associated to a low probability (e.g. 0.001%) of occurrence. The number of samples generated as output of the Montecarlo is constrained by computational time and particular statistical tools are used to estimate the quantile of interests when the number of outputs samples are smaller than required.

This paper describes an innovative method to compute the SRA, considering that the input models and its uncertainties are well defined. The method focuses on the statistical distribution of the uncertainties of necessary input parameters contrary to classical methods that generates a large number of impact points with Montecarlo simulation and processes the outputs of this computation.

As a different approach, the innovative method described in this paper processes only sets of the input dispersions associated to a given probability and, consequently, does not require generating Montecarlo simulations and processing the statistics of the output. The probability of interest is computed integrating the multivariate density function of the input parameters and, then, an optimization process is used to find the output worst case among a reduced set of inputs corresponding to a given probability. Three advantages of extreme importance can be recognized: the probability is computed constraining the input dispersions, that are directly associated to the causes driving the phenomena; a large amount of computational time is saved since a Montecarlo simulation is not required; the level of probability can be arbitrarily small because the computational time is not quite sensitive to the probability level to be achieved. The drawback of the method is related to the simplification that is introduced in the computation of the overall input probability. This simplification leads, in some cases, to an overestimation of the size of the box providing a conservative solution to the problem.

The method is applied to an example of the SRA computation for the destructive shallow re-entry of a large space vehicle in the South Pacific Ocean. When a vehicle performs a shallow re-entry, it travels on a final orbit whose perigee radius is larger than the Earth radius and it impacts the atmosphere with a flight path angle shallower than usual. Consequently, the spacecraft is exposed for a long period to particular aero-thermo-dynamic conditions, which enlarge significantly the dispersion of the ground impact area of the surviving fragments and makes particularly challenging the estimation of the SRA.

The paper presents the results of the computation of the shallow re-entry SRA using the classical Montecarlo approach and this innovative approach. The results are compared highlighting advantages and drawbacks in terms of accuracy, level of conservatism and computational time.

The method described in this paper is suitable for many future applications taking advantage of its computational speed and reliability: the destructive controlled re-entry of large structures, including in particular the International Space Station (ISS) and the ISS visiting vehicles at its End of Life (EoL); the destructive re-entry of large uncooperative satellites orbiting LEO and MEO as conclusive event of the Active Debris Removal (ADR) technology; the destructive controlled re-entry of last stages of launchers.

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Conjunction Risk Assessment and Avoidance Maneuver Planning Tools

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The ever increasing number of objects in the near Earth region has been causing growing concerns about the space environment and accordingly about the safety of future space missions. Since most of orbital debris stay in the orbit for years, even a single collision between space objects could seriously increase the debris population, making further collisions more likely.
The German Space Operations Center (GSOC) has been performing collision avoidance operation since 2009. Additional to the operational satellites currently 5 in LEO and 2 in GEO, conjunction monitoring and mitigation for several satellites are supported, which are operated by other organizations or ended the operation phase. The Conjunction Data Message (CDM) provided by the Joint Space Operations Center (JSpOC) is currently the main source for the orbital information of the space objects due to the quality and timeliness of the available information. Conjunction prediction, risk assessment, and avoidance maneuver planning are performed automatically in the operational process.
In the collision avoidance operation, numerous conjunctions are reported daily, which were detected within certain thresholds. Early detection of possible critical conjunctions among all predicted events is therefore important to handle critical situations promptly and efficiently. Additionally, earlier estimation of a possible avoidance maneuver strategy is also important, because the decision of the avoidance maneuver execution is mostly done within one day before the closest approach based on the latest prediction. Especially, the operational satellites TerraSAR-X and TanDEM-X are flying in a very close formation with a minimum distance of ˜300 m, therefore the avoidance maneuver for both satellites shall be taken into account to handle close approaches of each encountering object. The avoidance strategies to meet the control requirements and to optimize the maneuver shall be investigated in the limited time.
The important factors for the criticality assessment are the conjunction geometry and the collision probability. In addition to the geometry and probability analysis at the latest and in the historical prediction, estimation of the future prediction is also necessary to detect the high risk conjunction. Even after a detection of the high probability event, the conjunction shall be carefully analyzed due to the large variety of the orbit prediction accuracy depending on the encountering objects. In case of very large orbit uncertainties, the probability density is spread widely, therefore the cumulative probability that the object comes near the target could become lower as a result. For the avoidance maneuver planning, it is useful for operators to be able to estimate the effect of the different maneuver strategies to the conjunction geometry as well as the collision probability. The maneuver size and epoch shall be then adjusted depending on the desired safety criteria, the time constraints, and the orbit control requirements.
In the paper, algorithms for the conjunction risk assessment and the avoidance maneuver planning are described, followed by a presentation of their application in the automated collision avoidance process in the user-friendly way to facilitate the handling of the critical conjunctions. Finally, the lessons learned through the operation are presented.

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GNC MIL for Deorbiting with Drag-augmented Devices

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The paper describes the work performed and results achieved by LuxSpace for the ESA-supported DGNC project. DGNC stands for Dragsail GNC. It is a project aiming at identifying the best GNC solution to be proposed for satellite (debris) deorbiting thanks to Dragsails. The proposed and investigated GNC options are : (1) no attitude control, (2a) active attitude control constantly maximizing the area exposed to drag, this with a flat Dragsail, and (2b) active attitude control constantly maximizing the area exposed to drag, this with a pyramidal Dragsail. All DGNC options are also compared to deorbiting with (remainings of) onboard propulsion. In support to the DGNC system design and analyses, a GNC MIL i.e. a dedicated simulation tool has been created and validated within the ESA-supported GNCDE development environment. The paper describes also this LuxSpace's GNC MIL.

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Applicability of COBRA concept to de-tumbling space debris objects

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COBRA is a contactless concept for de-tumbling and controlling the attitude of a target space debris object that exploits the torques generated on the target by the plume impingement of a thruster facing the target to impart torques on the target. The control strategy for de-tumbling is based on a switching strategy for the de-tumbling thruster and a pointing strategy for aiming the de-tumbling thruster at a specific region of the target. This control strategy has been developed in a previous study of the concept. This article discusses several refinementsfurther developments of the original strategy and examines the general applicability of the COBRA concept, mainly in Active Debris removal missions in line with Cleanspace. The applicability of COBRA concept is investigated by examining several scenarios, namely, de-tumbling and attitude control of debris objects of varying configurations in terms of object geometry and mass parameters and in various initial rotation states. The main targets examined in this study are Envisat and Cryosat PROBA 2.
Simulation results described in a previous article have shown that the simple pointing and switching strategy for COBRA can successfully de-tumble a large space object such as Envisat in a relatively short time and using only a modest amount of ΔV, namely from an initial rotation rate of 5°/s to 0.5°/s in under one orbit. These simulations assumed a relatively favourable rotation state of Envisat, which allowed pointing the thruster at the Solar panel. The normal of the Solar panel was perpendicular to the rotation axis, such that a large torque could be generated. In the current article, new simulations will be presented for less favourable rotation states of Envisat. In addition simulations will be presented for a different target, PROBA 2. PROBA 2 is a roughly cube-shaped object, which means that there is no particular geometry (location plus orientation) of the thruster with respect to target that generates high torques. Previous results have indicated that the ΔV required ΔV required could be improved, in part by updating the thruster layout of the chaser and in part by updating the control strategy. The thruster layout is updated to include thrusters in the direction exactly opposite to the de-tumbling thruster to limit cosine losses. The control strategy is improved in a number of ways.
Previous results indicated that the ΔV overhead (that is, the ΔV not directly contributing to de-tumbling the debris) was higher than expected based on the ΔV required to compensate for the activation of the de-tumbling thruster. A part of the overhead can be explained by the cosine losses, but a substantial fraction was due to attitude control. The pointing strategy and the attitude control of the chaser are improved to reduce this overhead. In addition some effort is spent in improving the pointing strategy of the chaser. The current strategy uses a very simple model to predict the torques imparted on the target. The prediction model is improved such that the torque imparted on the target is closer to the desired torque.

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An Intelligent Multidisciplinary Design and Optimization Environment for Conceptual Design of Launch Vehicle

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Conceptual design process of aerospace launch vehicle is extreme challenge. A great number of alternative innovative or traditional configurations need to be evaluated, and the corresponding design parameters need to be balanced and defined. To quantitatively account for the interactions between disciplines and obtain a set of optimal design parameters, the methodology of Multidisciplinary Design and Optimization (MDO) is prevalently applied in the community of engineering. However, MDO only support parameter optimization for a given configuration. While too many potential configurations need to be evaluated during conceptual design, most of designer's work hours was spent on setting up MDO problems and models for alternative configurations. How to efficiently utilize MDO technologies and speedup conceptual design process is still a challenge topic in launch vehicle industry. Recently, a more general and intelligent multidisciplinary design and optimization environment, MCDesign-LV, is under development in Northwestern Polytechnical University (NPU) of China. In this paper, we will present the preliminary progress on MCDesign-LV. Firstly, the function requirements and architecture of MCDesign-LV is defined. Then, the main modules involved are introduced, including Configuration Reasoning Module, Disciplinary Design and Analysis Modules, Parametric MDO Model Generation Module, MDO Solving Strategy Generation Module. After that, three critical technologies to support MCDesign-LV are discussed: (1) How to define system configuration arbitrarily? The configuration of system is composed by that of disciplines. To explore design space as large as possible, the disciplinary design modules should have the capabilities of defining arbitrary configuration. We will discuss how to implement in the disciplines of aerodynamics, trajectory, propulsion, guidance, control, structure, and thermal. (2) How to generate the parametric MDO models automatically? To alleviate human task in MDO modeling for each configuration, parametric model generation tools were developed, including a CAD tool for geometry, multi-fidelity simulation models for disciplinary analysis. The input/output of disciplinary models will be tagged and integrated automatically. (3) How to create MDO solving strategy intelligently? By defining objective functions, shared and local design variables, system and local constraints, the coupling strength and scale of MDO problem will be evaluated. Then, with certain algorithm, a solving strategy will be generated, i.e., which kind of architecture (MDF/IDF/CO/CSSO) applied, which optimization algorithm and surrogate modeling process used. At the same time, the computational procedure will be distributed to network nodes automatically. Finally, two launch vehicle conceptual design examples with different configurations are introduced to demonstrate the capability of MCDesign-LV. One example is rocket based launch vehicle, and another is combined cycle based air breathing launch vehicle.

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Design of Optimal Observation Strategy for Re-entry Prediction Improvement of GTOs Upper Stages

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From 2004 up to the date more than 200 launch vehicles operated by five independent nations and two international organizations placed satellites in Geostationary Earth Orbit (GEO). In almost all cases, each successful launch left one or more pieces of debris in Geostationary Transfer Orbits (GTO). Particularly, many of this space debris consist of large spent upper stages of launch vehicles whose atmosphere re-entry might violate the constraint on casualty risk of 1/10000: as of 16 October 2014, it is expected that about 79 spent upper stages operating on GTOs with an inclination lower than 20 degree will enter the Earth atmosphere in the next 200 years. Moreover, the GTOs are highly eccentric orbits with perigee normally at low altitudes (170–650 km) and the apogee near geo-stationary altitude (35,780 km). Thus, space debris in GTOs generally passes through densely populated regions such as Low Earth Orbit (LEO) and GEO regions, being a hazard for the safety of other operating spacecraft. In light of the above, the improvement of re-entry prediction of GTO spent upper stages is a key issue to manage both on-orbit collision risk and on-ground casualty risk. Currently the only public data source available for re-entry prediction of a space object are represented by Two Line Elements (TLEs), provided by the United States Strategic Command (USSTRATCOM). However, this set of data are inaccurate and do not come with uncertainty information, making their use in re-entry prediction and conjunction analyses challenging, especially for the GTO space object. This leads to the need of using the observational data to improve the re-entry prediction. The design of observation strategy for GTO upper stage is not trivial. The detection and tracking of space objects on GTOs might require more than a single sensor in fact, since the distance from the observer has large variation along the orbit; this multiple sensors configuration might involve problems such scheduling or data fusion, making space object observation complex and costly. In addition, design of an optimal observation strategy for improvement of re-entry prediction involves the definition of a high-accuracy orbit determination (OD) algorithm, and the implementation of proper methods for uncertainty mapping. This might require the definition of accurate dynamical models in order to describe the effects of third-body perturbations and the Earth's oblateness and to capture the intricacies of re-entry phase, as well as the use of nonlinear technique for orbit determination. In this paper, a systematic approach to design the observation strategy of spent upper stage moving on GTOs is presented. More specifically, the design is formulated as a multi-objective optimization problem solved by means of a multi-objective genetic algorithm (MGA). This approach allows minimizing both the number of total measurements required to detect the space object and the error on re-entry prediction. Within the optimization process a nonlinear OD algorithm is run to determine the estimates of both initial state and model parameters. The Nonlinear Least Square Filter (NLSF) technique is implemented, exploiting the differential algebra framework to reduce the computational effort related to OD problem solution. Finally, the software tool IRIS is developed to accurately simulate the observation campaigns based on geometry and constraints of existing sensors currently available to European Space Agency (ESA).

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Modeling and Performance Evaluation of Multistage Launch Vehicles through Firework Algorithm

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Multistage launch vehicles of reduced size, such as "Super Strypi" or "Sword", are currently investigated for the purpose of providing launch opportunities for microsatellites. Currently, microsatellites can be launched according to the time and orbital requirements of a main payload. The limited costs of microsatellites and their capability to be produced and ready for use in short time make them particularly suitable to face an emergency (responsive space), therefore small launch vehicles dedicated to microsatellites would be very useful. On the other hand, in order to reduce the launcher size without increasing too much the launch cost per kg of payload it is necessary to simplify the launch system as much as possible, including the guidance algorithms. A simple open-loop guidance strategy is proposed in this research and applied to the Scout rocket, a micro-launcher used in the past. Aerodynamics and propulsion are modeled with high fidelity through interpolation of available data. In order to simplify the open loop guidance law employed for the first three stages, the aerodynamic angle of attack is assumed constant for each stage.

Unlike the original Scout, the terminal optimal ascent path is determined for the upper stage, using a firework algorithm in conjunction with the Euler-Lagrange equations and the Pontryagin minimum principle. Firework algorithms represent a recently-introduced heuristic technique inspired by the firework explosions in the night sky. The concept that underlies this method is relatively simple: a firework explodes in the search space of the unknown parameters, with amplitude and number of sparks determined dynamically. The succeeding iteration preserves the best sparks. With regard to the problem at hand, the unknown parameters are (i) the aerodynamic angles of attack of the first three stages, (ii) the coast time interval, (iii) the initial values of the adjoint variables conjugate to the upper stage dynamics, and (iv) the thrust duration of the upper stage. The numerical results unequivocally prove that the methodology at hand is rather robust, effective, and accurate, and definitely allows evaluating the performance attainable from multistage launch vehicles with accurate aerodynamic and propulsive modeling.

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An RST Design Approach for the Launchers Flight Control System

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The aim of the paper is to present a new design methodology for the automatic flight control system of launch vehicles using discrete-time RST controllers. The structure of such a control system has three degrees of freedom (roll, yaw, pitch), determined such that the closed-loop dynamics of the launch vehicle tracks the output of a desired reference model. The RST control technique focuses on the pitch angle, as the roll and yaw angles are tracking their references much easier.

Although the RST controller operates with input-output discrete time system models (and not with their continuous state representations), integration into a launcher is possible, thanks to continuous-discrete-continuous conversion tools. Thus, the continuous time launcher state space representation of each freedom degree can be first converted into the transfer function (continuous as well). Next, a discretizing technique (such as bilinear) is applied, in order to determine the discrete time transfer function. The RST controller is then designed according to the resulted transfer function and a prescribed second order system, standing for the desired closed-loop behavior. Such a controller is processing the reference trajectory and the actual angle (e.g. pitch) as input signals, to return the necessary command. Finally, the 2 by 1 controller transfer function is converted to the minimal discrete time state space representation, which, on its turn, is brought back to continuous time through some interpolation technique (usually, bilinear as well). The RST design controller relies on the poles placement method and reduces to solving a dyophantine equation in this case, although some other more sophisticated methods could have been considered as well.

The paper is organized as follows: after an introductory part, the second section presents the design models (after linearization) describing the dynamics and kinematics of the launcher, together with the design objectives concerning the control system. The design methodology of the discrete time RST controller is presented in the third section. The theoretical developments are illustrated and analyzed through some numerical examples in section 4. Some concluding remarks and future developments of the proposed approach complete the article.

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Innovative Strategy for Z9 Reentry

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ICATT 2016, 6th International Conference on Astrodynamics Tools and Techniques, 14 - 17 March 2016, ESOC, Germany Innovative Strategy for Z9 Reentry Gregor Martens*, Elena Vellutini**, Irene CrucianiELV Corso Garibaldi 22, 00034 Colleferro (Italy)**Aizoon Viale Città d'Europa 681, 00144, Roma (Italy)
Abstract Large Footprint of Zefiro 9, the 3rd stage of VEGA launcher spanning even more than 2000 km in equatorial missions, is one of the major system drawbacks constraining both performances and missionization process. This paper proposes a new strategy for the reentry of Zefiro 9 third stage of the VEGA launch vehicle, consisting of employment of retro-rockets coupled with closed loop guidance, which permits to improve the performance of the launcher and to reduce Z9 footprint. Z9 is one of the specific characteristics of VEGA: it works at high velocities and high altitude. Being a solid rocket motor it cannot be simply cut off and the impulse delivered depends on the propulsion scattering, not known a priori. This uncertainty produces a big variation in the Z9 impact point, function of the propulsive performance of the SRM. Current solution for the Z9 reentry foresees the employment of Neutral Axis Maneuver by orienting the thrust along the (neutral axis) predefined direction in order to minimize the impact point variation when an impulse is delivered along. NAM is performed in open loop guidance several seconds before Z9 cut off which is not predictable with precision, hence a certain percentage of its propulsion capability is lost in the maneuver. The new reentry logic permits to exploit the whole Z9 energetic capacity by not performing the neutral axis maneuver. Footprint extension is moreover drastically reduced by employment of retro rockets: small solid rocket booster with a fixed impulse of velocity. After Z9 exhaustion a slew manoeuver points the launch vehicle to the target attitude computed on-board and retro rockets are activated immediately after separation. The reentry logic is deeply analyzed and the adopted optimal reentry strategy is formulated. The improvements are evaluated in terms of Z9 footprint extension. Obtained results are compared with respect to the current reentry strategy. Possible error sources (i.e. navigation, guidance and control errors) are critically evaluated and their impact on the results is highlighted.

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Dromobile: A multi-platform tool for orbit propagation on mobile devices

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The development of mobile devices such as smartphones and tablets has caused big social changes since the beginning of the 21st century. Their computing power and memory has also been growing continuously and can be competitive against traditional machines for not computationally expensive problems. The combination of state of the art Graphical Processing Units and Central Processing Units allows most of these devices to develop remarkable capabilities, both numerical and graphical. While they are being used in several professional fields, they still haven't caused a big impact in the astrodynamics community.
In this paper a multi-platform web application for client's side accurate orbit propagation is presented. This tool can be used not only on a computer, but also in modern mobile devices such as tablets and smartphones, either with or without Internet access. It exploits both the processor and graphical capabilities of the hardware to calculate the orbit and other useful information for a mission analyst and to present the results to the user in a friendly and intuitive manner. The astrodynamics tools available at the moment typically require a workstation to be used,

which may delay the introduction of the mobile devices in the astrodynamics field. Since an easy-to-access and intuitive use of software should not be against accuracy and results consistency, Dromobile is intended to be a growing tool able to combine both aspects.
The special perturbation method Dromo is used to propagate the orbit, since it has been proved to have a good behavior in terms of computational speed and accuracy (see Urrutxua et. al. "Dromo propagator revisited", Celestial Mechanics and Dynamical Astronomy, 2015. doi: 10.1007/s10569-015-9647-y). The main advantages of Dromo are: the regularization of the equations of motion using a second order generalized Sundman transformation, the use of quaternions to orientate the orbital plane, and the fact that the Dromo elements are constant for Keplerian orbits, among others. The simulation environment is limited at the moment to geocentric orbits and the most important perturbations are included in the simulation.
The tool graphical interface is based on WebGL, which provides a light framework that does not takes away significantly computational power from the relatively heavier orbit calculations. It also allows complete portability between operative systems and devices since its standard is supported by all the major web browsers. Additionally, open source libraries based in the WebGL specification have been used in order to offer a free and flexible framework easy to modify and to add functionalities into.

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Analysis of spacecraft trajectories in proximity to small bodies: Phobos & NEO

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Associated to sample return missions to Phobos (Phobos-Grunt) or a mission targeting NEOs (MarcoPolo-R, AIDA), one generally aims at following trajectories that optimize Delta-V consumption, hovering stations, solar panel illuminations, ... We perform analysis of trajectories around such low gravity object considering in both cases the perturbed gravitational or non gravitational environment. Missions constraints consists in phases of hovering, different altitudes approaches and duration, and touch down. Starting from given reference missions, we will present orbitability studies around either quasi-satellite orbits (QSO) or equilibrium points (EP).

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Application of Kalman Filters in Orbit Determination: A Literature Survey

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Orbit determination has been a critical area of research ever since the start of space age. Research has been done in this area from defining accuracy goals for predicting orbits, to a host of software and hardware tools, techniques and methods. This paper was narrowed down to the most significant contributions in the area of Kalman filtering applications. Research work from 1967 to 2013 only is included in this paper. It was observed that examples from 1967 demonstrate the importance of pre-flight parametric studies when orbit determination estimations are carried out in short periods. Research also shows that extended Kalman filters can be made to work in real-data situations, contrary to what some professionals in the aerospace industry believe. Development of the extended Kalman filter requires some proper selection of parameters that define the probability density of the initial state vector, and any other parameters required by modifications to the extended Kalman filter algorithm. More recent research uses numerical methods such as the 4th order Runge-Kutta method to conclude that propogation errors are

not a problem for 15-day solutions for orbits. This paper can be used as foundation for further investigations in the area of orbit prediction.

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open-source publication: a strategic choice for private companies

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Open-source has become a mainstream asset in most parts of today's economy, including space. It's use is widespread and in every projects it is customary to look what existing open-source tools could be used as building blocks. Both private companies and public organizations are heavy users of open-source. The benefits are well-known by now (reliability, vendor-neutrality, standardization, maintenability, ...). Creating an open-source project on the other hand is quite a different topic. For the entity that does the initial work, what could be the incentive to publish freely some cutting edge know-how? If it is a private company in a highly competitive environment and it has to pay for the development, what could be the rationale behind this move?
This presentation relates such a story. It describes the history of the Orekit project, from its inception as an internal tool in a private company, CS-SI, up to its wide adoption in the space domain by now. It explains the reasons for the move to open-source and the rationale behind the license selection. We dicuss the expectations at project publications, what worked and what did not work. We describe the various phases from closed source to full open governance with a meritocratic model. At the end, we assess the current status of the project (spoiler: it is a success).

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The Horizon 2020 project ReDSHIFT: Revolutionary Design of Spacecraft through Holistic Integration of Future Technologies

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The ReDSHIFT (Revolutionary Design of Spacecraft through Holistic Integration of Future Technologies) project has been approved by the European Community in the framework of the H2020 Protec 2015 call, focused on passive means to reduce the impact of Space Debris by prevention, mitigation and protection.
ReDSHIFT will address barriers to compliance for spacecraft manufacturers and operators presented currently and in the future by requirements and technologies for de-orbiting and disposal of space objects. In ReDSHIFT these goals will be achieved through a holistic approach that considers, from the outset, opposing and challenging constraints for the safety of humans on ground, when these objects re-enter the atmosphere, designed for demise, and for their survivability in the harsh space environment while on orbit.
Ensuring robustness into the future, ReDSHIFT will take advantage of disruptive opportunities offered by 3D printing to develop highly innovative, low-cost spacecraft solutions, exploiting synergies with electric propulsion, atmospheric and solar radiation pressure drag. Inherent to these solutions will be structures to enhance the spacecraft protection, by fracture along

intended breakup planes, and re-entry demise characteristics. These structures will be subjected to functional tests as well as specific hypervelocity impact tests and material demise wind tunnel tests to demonstrate the capabilities of the 3D printed structures. Modern celestial mechanics and astrodynamics tools will be exploited to find "de-orbiting highways", (i.e., fast trajectories to de-orbit) able to meet de-orbit and disposal needs, coupled with the above-mentioned technical solutions.
At the same time, novel and complex technical, economic and legal issues of adapting the technologies to different vehicles, and implementing them widely across low Earth orbit will be tackled through the development of a hierarchical, web-based tool aimed at a variety of space actors. This will provide a complete debris mitigation analysis of a mission, using existing debris evolution models and lessons learned from theoretical and experimental work. It will output safe, scalable and cost-effective satellite and mission designs in response to operational constraints.
Through its activities, ReDSHIFT will recommend new space debris mitigation guidelines taking into account novel spacecraft designs, materials, manufacturing and mission solutions. In the talk, the technical description of the project will be given, along with the first progresses made within the study.

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The Fate of Highly Inclined Earth Satellites: From Order to Chaos

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We consider Earth satellite orbits in the region where the perturbing effects due to Earth's oblateness and lunisolar gravitational forces are of comparable order. This range covers the medium-Earth orbits (MEOs) of the Global Navigation Satellite Systems and the geosynchronous orbits of the communication satellites. There is no complete general solution for the long-term behavior of such satellites even in the averaged problem under the quadrupolar approximation; that is, when the disturbing function, consisting of the dominate oblateness term in the geopotential together with the lowest-order term (second harmonic) in the Legendre expansion of both the solar and lunar potentials, is averaged over the periods of the satellite and the disturbing bodies. Under the further approximation that the the lunar orbit lies in the ecliptic, reducing the averaged system to two degrees of freedom, Lidov and Yarskaya (1974, CosRe 12:139) indicate the known integrable cases and investigate the geometrical behavior of the resulting motion. The analogous problem of the dynamics of planetary satellites about an oblate planet perturbed by the Sun, studied for over two centuries beginning with Laplace, was treated recently in great detail by Tremaine et al. (2009, AJ 137:3706) who found all possible equilibrium solutions and determined their stability. But despite these important contributions, we cannot yet construct a complete, logically ordered picture of the global dynamics of gravitationally dominated orbits. It becomes increasingly more difficult when the complexities of the Earth-Moon-Sun system are taken into account, giving rise to secular resonances involving commensurabilities amongst the slow frequencies of orbital precession (Rosengren et al., 2015, MNRAS 449:3522). While the existence of some of these resonances has been known since the early 1960s, the implications of their dynamical effects are still being fully probed and understood (Daquin et al., CeMDA, 2015 doi:10.1007/s10569-015-9665-9). There remain many questions, fundamental and cardinal ones, in this complex subject that are of great practical importance and call for the need to develop new, simple, and reliable models and simulation capabilities.
Here, we will recall a first-order averaged model, based on the Milankovitch vector formulation of perturbation theory, which governs the long-term evolution of orbits subject to the the predominant gravitational forces (Tremaine et al., op. cit.). The averaged equations of motion

hold rigorously for all Keplerian orbits with nonzero angular momentum, and, along with their variational equations, will be given in a concise analytical vector form, which also intrinsically account for the Moon's perturbed motion. In Daquin et al. (op. cit.), it was shown that the devious network of lunisolar secular resonances that permeate the phase space of the highly inclined navigation satellites can interact to produce chaotic and diffusive motions. Using the Fast Lyapunov Indicator, they constructed dynamical stability maps that revealed a transition from a mostly stable region at three Earth radii, where regular orbits dominate, to a resonance overlapping and chaotically connected one at five Earth radii. The goal of this paper will be to explore in more detail this transition from order to chaos in Earth satellite orbits using the vector formulation, and to extend the inclination-eccentricity phase space study beyond MEO to seven Earth radii. Emphasis will be placed upon the phase-space structures near secular resonances which are of first importance to the space debris community. We will show that even the simple and deterministic equations of Tremaine et al. (op. cit.) can possess an extraordinarily rich spectrum of dynamical behaviors, answering why even without the destabilizing influence of atmospheric drag many Earth satellites eventually fall down (wt1190f). Further studies in this area may lead to deeper insights in celestial mechanics as well as provide practical results for satellite technology.

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STAVOR: Transition from desktop to new mobile platforms

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Software environments have changed a lot in the last few years, since the release of the first smartphones. From the typical Windows-Unix market, where languages like Java emerged as multi-platform solutions, new very different systems have appeared leaving the term multi-platform obsolete. To refer to an application that can run in different systems and platforms (Desktop, mobile, web. . .) we use the term Cross-platform.
CS Systèmes d'Information is the main developer of an open-source space dynamics library called Orekit. This library is coded in Java, to benefit of its multi-platform capabilities. Since this language is still largely used for space applications, the operational use of the library in a near future is not at risk. On the other hand, its utilization in new environments like mobile applications is restrained. Many different solutions have been tested to integrate the library in such environments. The main problems of these new platforms and devices are the heterogeneity of hardware and frameworks: very different screen sizes and resolutions, storage capacity, new GPU architectures with not fully implemented standards, battery consumption, multiple switching connections, variable signal, application and device life cycle. . .
The library is already able to run in Android, due to the fact that it embeds a Java virtual machine, similar to the one in Desktop environments. For the integration into other devices like Apple smartphones and tablets, mobile devices based on web browsers and others, two approaches were considered: convert the code or binary of the library to a language recognized by each device, or convert it to a Web language and use it everywhere.
Many source code and binary converters have been used without success due to the data interface of Orekit. Such a solution needs of many modifications in the library code to success the conversion. A solution that obliges to maintain different versions of the code and binaries was not desired, and the conversion to JavaScript requires the modification of all the data interface of Orekit, so this was saved as a possible solution in the future.
The last possible solution is to connect remotely Orekit, which demands a server and a fast internet connection, not very common in mobile devices.
To help with this study, the application STAVOR has been produced as an exemple. It uses Orekit as a space mission simulator, wrapping it in a touch-ready UI with some 3D and 2D visualization modules to represent the simulation results. This application has been implemented in an Android-native UI + embedded browser for the 3D and 2D models (based on WebGL and OpenLayers respectively), and in a pure embedded web solution, including all the UI.

The study concludes that Native solutions prevail over Cross-platform alternatives for the moment due to performances issues. In a near future, all mobile web technologies will be mature and stable enough to run heavy 3D and simulation software like any Desktop platform. At this point, cross-platform solutions will take the place to ease the development with adaptive interfaces and single code implementations.

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A new Mars EDL mission design and simulation tool - MEDLMDST

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Mars entry, descent and landing (EDL) begins when the vehicle reaches Mars atmospheric interface (about 125km altitude) and ends when the supersonic parachute is deployed. EDL phase is considered the critically important sub-phase, and largely determines the success or not of entire Mars EDL. A new Mars EDL mission design and simulation tool, named as MEDLMDST, has been developed at the Space New Technology Laboratory, Nanjing University of Aeronautics and Astronautics. In this paper, the components and architecture of MEDLMDST are firstly introduced. Then, the progress of MEDLMDST and some new results in EDL trajectory optimization, navigation and guidance are reported in detailed.

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Using accurate ephemerides of solar system objects for autonomous navigation

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The future of space missions will be dependent on the improvement of autonomous celestial space navigation methods. Nowadays the deep space exploration uses the starry sky background for attitude determination by imaging and determining the coordinates with an embedded star catalog. Two different approaches are necessary in order to find the position of an interplanetary spacecraft. The first one when you have an approximation of the position using the latest position known, and second, the most complicated when you are lost in space. To achieve these goals we need to have an accurate model of the space environment with usable targets at any time, namely planets and asteroids in the good frame reference. The ephemerides must be computed before the mission with the goal to estimate the number of objects usable for the position determination by the use of a 3D positioning algorithm. In a first step, our method proposes to use the virtual observatory ephemeris to know the number of visible useful objects and to determine the resulting accuracy of the calculated position of the space probe. The next step will be the making of a simulation allowing to determine the parameters to be improved in order to get a more accurate position of the probe.

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The NAROO project for overcoming past, current, and future ephemeris errors

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Accurate orbit determination require a large amount of observations dispatched over a large time span to allow for the best precision and extrapolation. This latter is of high importance in the context of future space missions.
In practice, most orbital models of the Solar System objects are fitted to data covering typically about one century. Even if the conditions are required for precise dynamical modeling, we emphasize an important caveat : ephemerides can sometimes be significantly biased while their extrapolation quickly diverges. Even though this could be due to various reasons, we found that an important one consists in the imprecision of past observations that are introduced in the adjustments. These observations were processed a long time ago with inaccurate star catalogs and with inaccurate methods compared to recent ones. No real efforts have been attempted to reanalyze these data a new time, considering the amount of time, mean and energy required.
Using photographic plates of planetary satellites, we demonstrate that a new reduction of old observations can improve significantly the ephemerides. In this framework and with support of the Gaia mission, the NAROO project has been initiated at Paris Observatory with the primary aim to reprocess the old astrometric observations with the best instrumental, algorithmic and numerical techniques. We discuss the impact of the project on future planetary and satellite ephemerides.

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Modular Fuzzy Interacting Multiple Model for Maneuvering Target Tracking

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As a branch of the field of target tracking, maneuvering target tracking plays an increasing role in military and civilian fields. A novel maneuvering target tracking algorithm is investigated. Drawing on the experience of combination idea of the modular structure and the fuzzy interacting multiple model algorithm (FIMM), a modular fuzzy interacting multiple model algorithm (MFIMM) is presented. The MFIMM algorithm consists of three independent modules working in parallel, called non-maneuver, weak maneuver and strong maneuver respectively. And the change of a target motion is also divided into three levels: no, small, and big. The motion of a target is detected by a fuzzy control motion detector, which imitates the thoughts of human beings to detect a target's motion. Once the maneuver is detected, the MFIMM algorithm selects one of the three modules matching the actual movement of the target every moment according to maneuver condition, and the state vector and covariance matrix is compensated, so that the modified state can suit the actual motion well. Afterwards, the MFIMM algorithm estimates the state of the target through interactive multiple model algorithm (IMM) based on square root unscented Kalman Filter (SRUKF) of the selected module. Therefore, under the architecture of the proposed algorithm, the fuzzy motion detector deals with the level of motion and the modules switching, whereas the IMM-SRUKF accounts for the estimation of the dynamic system. At the end of the paper, simulation is performed on the problem of maneuvering target tracking in two-dimensional space. In order to evaluate the effectiveness of the MFIMM method, Root Mean Square Error (RMSE) of the estimated state is used. The simulation test tracks a same target with determined trajectory by MATLAB emulation comparing the two algorithms, MFIMM and IMM. Results demonstrate that the proposed MFIMM algorithm improves the tracking precision and reduces the computational burden compared with traditional IMM.

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Mercury rotational state estimation applied to the BepiColombo Mission: an optimized approach on the selection of optical observations

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The BepiColombo mission is finally approaching the last phase, preceding its launch, which will bring the spacecraft to its final destination, planet Mercury, around which the probe will start its one year nominal mission in 2024. In-orbit investigations are a privileged condition for exploring a planet, and Mercury has demonstrated to be a fascinating destination since Mariner 10 first unveiled some of its mysteries. MESSENGER disclosed several questions but also arose other interrogatives and much is still to be discovered. One of the objectives of the BepiColombo Radio science Experiment is the study of Mercury's core. Since a strict connection exists between core and rotational state, measurements of Mercury's obliquity and librations at unprecedented accuracies became one the main purposes of MORE (Mercury Orbiter Radio science Experiment) rotation experiment. The rotation experiment will be accomplished thanks to the cooperation among different payloads: precise orbit determination data will be derived from MORE and high resolution images will be provided by HRIC, part of the SYMBIO-SYS payload. An end-to-end simulator has been built up employing the camera images as the primary observables with the final aim of defining their optimal acquisition scheduling. In this simulator the images are employed to correlate surface landmarks extrapolated by pictures of the same area taken at different epochs: their displacement in time represents the observable to be fed into an estimation process for deriving Mercury's rotation parameters. An extensive simulation campaign has been performed leading to the identification of the most favorable observational strategy and location of the landmarks on the surface so as to fulfill accuracies lower than 1 arcsecond for both obliquity and libration estimation. Concurrently an Orbit Determination simulation scenario for MPO has been set, in which high fidelity, but still evolving, models are implemented to consider all possible degradation effects. The setup consider an on-board accelerometer error model, simulating ISA instrument, errors in the radiometric and optical observables and a full gravity field implementation to degree and order 30 and tidal love number k_2. A pure multiarc orbit determination filter provides the simulated trajectory, the computation of simulated observables and then estimates the whole global and local parameters in the play providing direct insight on mission performances. This architecture has been exploited for the evaluation of the difference in performance by using a random generated set of observation locations or an optimized set for optical observables focusing in the achievable uncertainty in the orientation parameters, in particular for what concern the Mercury obliquity ε and amplitude in longitudinal libration Φ_0 validating the optimization approach.

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The ESPaCE consortium as a European producer of spacecraft and natural moon ephemerides

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The consortium ESPaCE (European Satellite Partnership for Computing Ephemerides) is composed of seven European institutes: IMCCE (Institut de Mécanique Céleste et de Calcul des Ephémérides, Paris Obs.), ROB (Royal Observatory of Belgium), TUB (Technical University of Berlin), ERIC (European Research Infrastructure Consortium formerly known as JIVE : Joint Institute for VLBI in Europe), TUD (Delft University of Technology), French space agency (CNES) in France and German Aerospace Center (DLR) in Germany. The objective of the consortium, initiated under an FP7-European project is to provide new accurate ephemerides of natural satellites and spacecraft. For this goal astrometric data issued from ground-based observations as well as from space observations are being analyzed and reduced. On the other hand emerging technologies, specifically VLBI and interplanetary laser ranging, applied to the positioning of spacecraft are also studied. The ESPaCE project addresses also data related to gravity and shape modeling, control point network and rotational parameters of natural satellites. The accuracy improvement of these ephemerides makes them a powerful tool for the analysis of space missions, the preparation of future missions, or for the determination of planetary physical parameters. Among relevant sub-products for space missions, we note the delivery of updated ephemerides of the Mars moons Phobos and Deimos derived from data by the Mars Express mission. In addition, the ESPaCE ephemerides of the Galilean moons are regularly updated in the context of the upcoming JUICE mission.

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Hybrid SGP4: tools and methods

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The propagation of an orbit which is subject to perturbation forces is a non-integrable 6-degree-of-freedom problem that has been classically tackled in three different ways. General Perturbation Theories attempt to deduce an analytical expression for the future position and velocity of the orbiter as a function of its initial conditions and time. Nevertheless, the non-integrability of the problem makes it necessary to consider simplified models for the perturbing phenomena, as well as certain approximations that have a negative effect on long-term accuracy. Another classical approach, Special Perturbation Theories, consist in applying numerical integration methods to the problem, which allows for the consideration of very complex mathematical models of perturbing phenomena. Nevertheless, in order to obtain highly accurate results, small integration steps must be taken, which implies long computational time. The third way to handle the problem, Semi-analytical techniques, apply analytical transformations so as to remove the short-period dynamics from the equations of motion, which can be numerically integrated then through longer integration steps, and hence more reduced computational time. By doing so, long-term propagation can be performed very efficiently. Short-period dynamics can be recovered at the final epoch in order to complement the propagated mean elements and thus provide the osculating elements. More recently, we have proposed a new approach, Hybrid Perturbation Theories, which consist of an integration method followed by a forecasting technique. The former, which can be any of the aforementioned techniques, is intended to generate an initial approximation, whereas the latter, which might be a statistical time series model or computational intelligence method, complements that approximation by forecasting its error at the final epoch. In order to achieve it, the second stage must model the dynamics corresponding to the difference between the output of the first stage and the real behaviour of the orbiter. An initial control period containing such dynamics is, therefore, necessary. We will consider a hybrid propagator composed of SGP4 plus an additive Holt-Winters method in order to describe this methodology and the software it involves.

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Proposed algorithm for on-board maneouvres calculation

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The orbit control for LEO missions is becoming more and more demanding in terms of manoeuvring. This paper proposes a simplified algorithm in order to calculated on board the drag make-up manoeuvres.
It is not pretended to give a full solution of the problem, but a starting point for future implementations. There are two main rationale of calculating this manoeuvres on-board, the first one is to mitigate the operational load to prepare and execute them, the second one is to remove the uncertainty on the orbit evolution between the manoeuvre calculation to its execution time, mainly due to the atmospheric drag. The resulting operational concept allows to keep very tight orbit control with high accuracy and low operational load.
The proposed algorithm compares the actual time of ascending node crossing with respect to the reference one. The delta-time provides the shift with respect to the reference ground-track while its increment allows to determine the altitude of the satellite. When a threshold in the ground-track shift is reached a manoeuvre is triggered based on the current altitude.
Alternative more complex calculations can be implemented allowing applications for formation flying. For example improving the on-board knowledge of the position of the other satellites in the constellation.
The algorithm is based on the one used to control the ground-track of GOCE, therefore it is not only applicable to chemical propulsion or cold gas, but on the contrary it is highly appropriate to low thrust missions. It introduces also the concept of micro-manoeuvres, that allows to emulate a low thrust control with medium thrust propulsion.

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Impact of Solar Spin on Planetary Orbits

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Abstract. This paper examines the possible effect of solar spin in the planetary orbit. We show in our earlier paper that if we incorporate the contribution of spin of the central gravitating body in orbital calculations, a residual slight perturbation on the standard constant areal velocity should exist. In particular, the second law of planetary motion requires a revision. However, it turns out that the classical result of Kepler is recoverable from our result as a special case. To be able to appreciate the need for the revision suggested by the new perturbation considered here, this paper looks into the genesis of orbital theory. We herein propose to reduce spin theory to non-relativistic regime. In fact, we consider restricted three-body problem. Confining, for the moment, again to the specific context of solar system, our initial calculations show that the transverse component the transverse component of the force field is nonzero, in contrast to the GN-physics (Galilei-newtonian physics) wherein such a component vanishes. In particular, the transverse component of the central force field does vanish if we neglect the spin of the gravitating star. This situation is radically different from that of GN-theory (where linearisation often does result). However, if we set the spin equal to zero, we retrieve the orbit equation of GN-physics. As regards solution, we apply numerical schemes to determine solution of nonlinear orbit equation for Earth. Our results exhibit that the new light on issue in relativistic celestial mechanics and models of planetary motion.

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Fast Low Earth Orbit Acquisition Plan Optimiser

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Earth observation data is key for a more efficient use of land and natural resources, better land and sea monitoring, more informed political decisions, and better understanding of the weather, climate, and land changes.
Many Earth observation satellites are operated in low Earth orbits which provide a good trade-off between revisit time and spatial resolution. Repeat ground track orbits are of particular interest as they allow the acquisition of the same scene at fixed time intervals. Additionally, either for calibration or nominal operation proposes, many satellites are required to overpass an exact location on Earth.
For missions comprising spacecraft constellations, accurate orbit phasing is also needed. Hence, the satellite manoeuvres should be carefully planned to control the ground track drift so that the desired longitude at ascending node crossing are achieved.
This paper describes a tool and the associated mathematical framework that automatically computes an orbit acquisition plan that minimize the duration of the orbit acquisition phase or the required Delta-V given the spacecraft characteristics and mission constraints. The automatic algorithms are built upon on a perturbation analysis of the nominal orbit and provide the necessary information to perform preliminary analysis of orbit acquisition phases of Earth observation satellites.
Two kinds of simplified orbit acquisition plans can be computed: i) continuous semi-major axis change and ii) impulsive semi-major axis change. In the former, constant Delta-V per time interval is applied to the spacecraft, which is suitable when several small impulsive manoeuvres can be approximated by a continuous manoeuvre and also when there is no detailed information about the manoeuvring capacities and constraints. The latter is based on impulsive semi-major axis changes, which allows a more detailed plan where practical consideration such as the existence of calibration and touch-up manoeuvres are taken into account.
The tool also allows the analysis of several semi-major axis launch dispersions and launch dates providing for each case the resulting acquisition duration, required Delta-V, and mean local solar time drift. In particular, to insert another spacecraft within a constellation, a trade-off needs to be made regarding the requested orbital elements offset at orbit injection (in terms of semi-major axis, inclination or mean local solar time). Taking into account the spacecraft manoeuvre capabilities, the operational constraints of LEOP, the constraints preparing and implementing manoeuvres, launch date constraints and the agreed (or expected) launch dispersions, the tool identifies the consequences of all the possible scenarios (all cases are analysed together), which can then be used to define the orbital offset to be requested.
Some examples are given that illustrate the potential applications of this tool.

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An implementation of SGP4 in non-singular variables using a functional paradigm

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The SGP4 (Simplified General Perturbations 4) orbit propagator is a widely used tool for the fast, short term propagation of space orbits. The algorithms in which it is based are thoroughly

described in the SPACETRACK report #3, as well as in Vallado et al. update. Current implementations of SGP4 are based on Brouwer's gravity solution and Lane atmospheric model, but using Lyddane's modifications for avoiding loss of precision in the evaluation of the periodic corrections, which are, besides, notably simplified for improving evaluation efficiency. Different alternatives in the literature discuss other variable sets, either canonical or not, that can be used in the computation of periodic corrections (see Izsak, Aksnes, Hoots, or Lara).
This work presents a new implementation of the SGP4 algorithm in Scala that offers a choice about the variable set used for the computation of the periodic corrections. Scala is a hybrid functional/object oriented programming language running in the Java Virtual Machine that allows for incorporating functional features in the design. Validation of the new implementations is made by carrying out different tests based on Vallado's results. Finally, applicability for massive data processing tasks like prediction of orbital collision events and performance are discussed.
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New tool for finding periodic Halo orbits: the solver of a Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation)

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Halo orbits and other periodic orbits in the restricted circular 3 body problem has always been very well explained in the literature since their discovery by Farquhar in the 60 of last century. However for finding the numerical values of such orbits, the availability of the tools dedicated for such tasks is not obvious. Due to the fact that the differential equations are quite simple, those days most of the time tools used are based on some computer listings written within an US based mathematical framework, which is clearly dedicated for people highly involved in computer and computer language rather than for general purpose Engineers. Hence the approach chosen in the paper is to rely on a tool largely used by Engineers (and not computer guys) for taking advantage of the capabilities of solving dynamic problems: the tool used is a European tool which is a object oriented solver of differential equations which is the cornerstone of the Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation) largely used by engineers. The paper presents the mathematical problem in simple words and the method used to solve it. The major advantages of the approach proposed and used successfully is to benefit of a real simulation framework based on models and on experiments where there are no mixing between the inputs\outputs needs and the real problem being to be solved. Hence the full model can be clearly and explicitly described while the results coming from the experiments can be extensively assessed and analysed with simple monitor outputs.

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A Series for the Collision Probability in the Short-Encounter Model

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The increase of conjunctions between active satellites in Low Earth Orbit (LEO) and other objects, either space debris or other satellites, has made necessary to evaluate the risk posed by these conjunctions in order to decide if evasive maneuvering is needed. The calculation of the collision probability between a pair of objects must done by a precise and fast algorithm because of the enormous and growing LEO population and because numerical methods for collision avoidance maneuver optimization may need to evaluate this probability several times during its execution.
In this article, a series to compute the collision probability of two spheres under the assumptions of short-term encounter, which generally hold in LEO, has been derived. It is valid for both Gaussian and non-Gaussian distributions of the position of the spheres, and in the particular case of a Gaussian distribution the use of Hermite polynomials yields a simple form for the series.
The parameters that appear in our formula, or in others which address the same issue, are the axes of the projection of some incertitude ellipsoid on the collision plane and the projection on the same plane of the relative coordinates of the objects which might collide. A region of practical interest in this parameter space has been carefully defined based on satellites' real data, and a representative sampling set was chosen. On this sampling set a comparison between the new series and previous algorithms has been performed for the Gaussian case to measure the performance of the proposed method. The presented series is found to be faster than any other algorithm in every explored case. Numerical evidence suggests that if the series for the Gaussian case is truncated when the last term is smaller than the computed probability times a tolerance of 0.1, then the last term is an upper bound for the error. This article also presents very strong evidence for the case that the first two terms of the series are sufficient for the computation of the probability of collision, and that the absolute value of its second term is an upper bound for the error made when using it.

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Probabilistic orbit lifetime assessment with ESA's DRAMA/OSCAR tool

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ESA's space debris mitigation policy has come into force in March 2014 and adopts the space debris mitigation technical requirements from the ECSS adoption notice of ISO 24113. Those requirements include recommendations on the disposal of systems that have reached the end of their useful life, and were driving the development of OSCAR (Orbital SpaceCraft Active Removal). Specifically, OSCAR is the component of DRAMA designed to address disposal manoeuvres at end-of-life and assess the compliance of the later stages of a mission with the mitigation requirements, where mission planners have to decide on their implementation at early stages in the project (typically around SRR/PDR). In its current version, published and freely available within the DRAMA software suite, OSCAR allows for the analysis of the orbit evolution subject to different possible future scenarios for solar and geomagnetic activity, which are the main drivers in the estimation of the residual lifetime for a specific orbit.
Based on standardised and widely accepted methods for the prediction of those scenarios, the remaining orbit lifetime is computed via a semi-analytical propagation taking into account all relevant perturbations.

However, long-term forecasts of the orbit evolution are very sensitive to several quantities, most of them very difficult to forecast, including the solar and geomagnetic activity, the object's cross-section, its attitude state and mass, the drag and solar radiation pressure coefficient of the object, as well as physical model limitations. Moreover, the uncertainties in the injection manoeuvre transferring the spacecraft into its disposal orbit and uncertainty in the disposal epoch cannot be neglected. With OSCAR being used in the compliance verification process with respect to the mitigation requirements, the current approach to also assess the uncertainty associated with a lifetime estimate shall be discussed. By accounting for the various sources of uncertainty, OSCAR will allow for a more probabilistic and thus more realistic estimate, which is beneficial in the compliance analysis.
This paper will give a brief overview on the core functionalities of OSCAR and then focus on the uncertainties considered in the latest update of OSCAR. The propagation of the uncertainties results in a probabilistic assessment of the orbit lifetime. For example, the 25-year-rule compliance can then be based on an assessment in how many cases the orbit lifetime would be below or above 25 years.
Finally, some exemplary results will be provided, addressing different orbital regimes.

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Debris operational support through the web applications

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With the growing number of debris and satellites which ESA operate, the necessity for operators to mitigate collisions with debris is a reality. Meanwhile atmospheric re-entry of objects is also gaining lots of media interest.
In order to enhance ESA's Space Debris Office's (SDO) operational support to missions and to national alert centers, CGI has been developing web based tools which improve decision making speeds, decision quality and end user experience in general. The tools also allow to scale up operational support to more missions, be it ESA's or external ones.
What are being developed are essentially front-ends for displaying data from the SDO's databases or generated from SDO's computations. The display of data are in tables, graphic plots or 3D visualisations and is split in views which are specifically adapted to the role of the user, e.g. satellite operator, debris analyst or manager. Among the multiple functionalities, an important one is the possibility to launch the SDO's avoidance manoeuver optimiser CORAM and to display the resulting trajectory and close approach geometry.
The applications were developed with powerful web technologies, have a common visually appealing look-and-feel and have been developed with continuous user feedback, such that the ergonomics are adapted to the SDO's work flow as well as to its own varied customers. The latest version of the tools will be demoed during the presentation.

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Tools and Techniques Supporting the Operational Collision Avoidance Process at ESOC

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ESA's Space Debris Office provides a service to support operational collision avoidance activities. This support currently covers ESA's missions Cryosat-2, Sentinel-1A, Sentinel-2A, Sentinel-3A, and the constellation of Swarm-A/B/C in low-Earth orbit (LEO). The support process is provided to third party customers, too.
We provide an overview on tools used in the mission design phase and during the operational phase. In addition, we briefly introduce the process control and data handling in the ESA process.
During the mission design phase collision avoidance is studied. Here the focus is at the effect of warning thresholds on the risk reduction and manoeuvre rates. For such analysis ESA's DRAMA tool suite with the module ARES is available.
During operations collision avoidance needs to address conjunction event detection, collision risk assessment, orbit determination, orbit and covariance propagation. ESA's process based on the central tools CRASS and CORAM is implemented following a database-centric approach through a temporary local "mini-catalogue". This catalogue is based on Conjunction Data messages (CDM) and own operational orbits. Each forecasted conjunction event is analysed in an automated way, returning the approach details and an estimate of the associated collision probability. A wide range of contemporary collision risk estimation algorithms with covariance scaling is supported. Identified high-risk conjunction events are further assessed and mission-specific processes are in place for decision-taking and manoeuvre recommendation. CORAM is able to assess optimised manoeuvres considering various constraints.
The database is also used as the backbone for a web-based tool (SCARF), which consists of a visualisation component and a collaboration tool that facilitates both, the status monitoring and task allocation within the support team, as well as the communication with the control team. The web-based solution optimally meets the needs for a concise and easy-to-use way to obtain a situation picture in very short time, and the support of third party missions not operated from ESOC.

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Sensor fusion analysis for HEO space debris using BAS3E

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One of the main missions of a Space Surveillance system is the detection and cataloguing of space objects having a size compatible with the detection constraints of its sensors. While radars are used to observe objects placed at Low Earth Orbits (LEO), and Telescopes to observe objects orbiting in Medium (MEO) and Geostationary Orbits (GEO), for objects orbiting in Highly Elliptical Orbits (HEO), both types of sensors are suited for observations. In particular, the passage through the perigee can be observed from radar stations, while in the high-altitude orbit portion telescopes are intended to be the source of observations. In this way, combining data derived from different type of sensor seems advantageous for tracking HEO objects.
BAS3E (Banc d'Analyse et de Simulation d'un Systeme de Surveillance de l'Espace) is a space surveillance system simulation bench that includes the "real world" simulation (objects and sensors) and the operational surveillance system (e.g. Cataloguing and catalogue maintenance, collision risk assessment, re-entry prediction and fragmentation detection). This tool enables the performance analysis of the surveillance network depending on its features, as well as the algorithms involved in the catalogue maintenance, analysis and planning systems. BAS3E is composed of different subsystems including:

- Observations simulation of surveillance network sensors: reference orbits generation, considered as "real" orbits, and the associated sensors observations generation.
- Catalogue maintenance system: sensors observations treatment and correlation, objects orbit determination, and objects database maintenance (the catalogue).
- Catalogue analysis system: orbits and covariances propagation, from catalogue objects; collision risk and reentry risk assessment using the propagated orbits; and fragmentation analysis and detection.
- Planning of sensor observations depending on previous analysis systems requests.

BAS3E makes use of parallel computing techniques, allowing, in a reasonable time effort, the simulation of thousands of space debris objects and the operation of a dedicated space surveillance network.
This paper describes an application using BAS3E, focused on the performance analysis of different space surveillance networks in the detection and cataloguing of a HEO population of objects. The use of BAS3E is described, including the concatenation of computation stages, the databases management and the persistence layers. Sensor networks considered within this study contain different combinations, in quantity and quality, of ground-based telescopes and radars. These sensor networks have been selected attending to criteria such as viable number of stations for a mid-term deployment and quality (measurement noise, observability constraints . . .) in line with already-operating sensors. This study tends to evaluate the advantages of fusing data from different sources (only optical measurements, or optical measurements along with radial distance and/or radial velocity data) concerning HEO orbit determination and, besides, conclusions are presented on the gain of incorporating additional sensors in a hypothetical space surveillance system ground network.

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Fragmentation Event Model and Assessment Tool (FREMAT) supporting in-orbit fragmentation analysis

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The future sustainability of the near Earth environment requires continuing efforts to increase our knowledge of the current and future debris population. Possible on-orbit fragmentation events are a major concern nowadays. The Fragmentation Event Model and Assessment Tool (FREMAT) project for ESA was carried out with the objectives of simulating on-orbit fragmentations, assessing their impact on the space population and evaluating the capability of identification of fragmentation events from existing surveillance networks. In the frame of the FREMAT activity, the implementation of several algorithms related to on-orbit fragmentation events was carried out. FREMAT encompasses three individual tools: Fragmentation Event Generator (FREG), Impact of Fragmentation Events on Spatial density Tool (IFEST) and (Simulation of On-Orbit Fragmentation Tool) SOFT. Fragmentation Event Generator (FREG) has been conceived to simulate fragmentation events (explosion and collisions). A breakup model based on recent models was the baseline for this tool .We have enhanced the baseline NASA break up model, in order to ensure the consistency of mass and momentum in the created fragment clouds. Its output is one or two clouds of fragments that can later be fed into IFEST or SOFT, or to any other propagator. The second tool, IFEST (Impact of Fragmentation Events on Spatial density Tool) allows the evaluation of the impact of on-orbit fragmentations in the space debris population. This tool employs a fast semianalytic propagator for computing the long-term evolution of the clouds of fragments (up to hundreds of years) obtained from FREG, and computes the spatial density caused by those fragments as well as the percent increase in the background spatial density obtained from MASTER. The computation of the spatial density within this tool is validated against results provided by Esa's POEM tool. Finally, the third tool, SOFT (Simulation of On-Orbit Fragmentation Tool), has been created to simulate the determination of the type of fragmentation and the objects involved in a fragmentation event when a space surveillance network detects a number of unexpected new objects and a fragmentation event is considered a possible cause. It can process a cloud from FREG, and clouds from other sources can be adapted to be processed by SOFT. Uncertainties in the knowledge of the orbits of the fragments and the presence of foreign objects is also considered. . The tool determines the type of fragmentation, calculates the time and location of the event and identifies the parent objects. This paper presents a description of the algorithms implemented in this toolkit, a brief description of each tool and a brief summary of their main functionalities. Furthermore, study cases are presented including parametric analysis by means of introducing variations in the input parameters of the fragmentation model. Short and long-term evolution of the clouds are studied, as well as the feasibility of determining the location and time of the fragmentation event. Additionally, the influence on the increase of collision risk is assessed.

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Launch vehicle multibody dynamics modeling framework for preliminary design studies

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Launch vehicle dynamics modeling is quite challenging mainly because of the highly interconnected disciplines involved: propulsion, aerodynamics, structures, mechanisms, and GNC among others. Discipline experts perform their respective design often independently and with separate dedicated tools. Consequently, during launcher preliminary design studies, numerous iterations are required in order to keep mission objectives synchronized.
These preliminary design efforts can potentially be reduced by using a multidisciplinary launch vehicle model integrated in one single tool. Because this allows to reduce the number of iterations and the associated costs, a launch vehicle multibody dynamics modeling framework is a key technology to aim for.
Dedicated developments of multidisciplinary modeling tools for launch vehicle multibody dynamics have been presented in the relevant literature. However, none fully profits from an object-oriented, equation-based, and acausal modeling language like Modelica. As yet, such an approach is still missing. It is therefore the objective of this paper to introduce such an alternative approach employing this modeling framework.
This framework enables object-oriented and physics-based modeling of subsystems and components related to most key analyses of a launcher system. These include among others: launcher configuration, staging and separation dynamics, end-to-end trajectories, performance, controllability and stability. Moreover, all this can be done within a single simulation environment.
The paper gives an overview on the first building blocks leading to an integrated and multidisciplinary tool for launcher preliminary design studies. Particularly, its easiness of implementation is demonstrated along with the benefits of this approach.

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Launcher mission analysis platform for fast and accurate mission domain performance assessment

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Facing the civil launcher market evolution toward higher versatility of missions and shorter development cycles, Airbus Defence & Space has improved its mission analysis platform for

Ariane 6 needs. This platform enables fast and accurate assessments of launcher architectures throughout the development phases. It contributes to launcher staging optimization, tuning of engine characteristics and sizing trajectories delivery to design teams. The mission domain covers a large range of injection orbits often requiring multiple firings of the upper stage. The trajectories have to satisfy numerous design and operational constraints, especially related to safety rules. The upper stage disposal scenario is also part of the mission analysis since it must comply with the space law while inducing the lowest possible performance penalty.
The mission analysis platform is based on Airbus Defence & Space optimization trajectory tool. An automatic and robust convergence toward the optimal payload injection trajectory must be guaranteed on the whole Ariane 6 mission domain. The core of the optimization process is a solver for the upper stage flight. This solver based on an indirect method yields quasi instantaneously the upper stage optimal leg reaching the targeted orbit whatever the initial state. An automatic initialization procedure using the in-house nonlinear programming libraries guarantees the fast convergence over the Ariane 6 mission domain.
The Flight Performance Reserves (FPR) ensure a given mission success probability regarding flight dispersions and design uncertainties. FPR must be carefully estimated at the early stages of the design process since they directly influence the launcher performance. The FPR are assessed on-line along the optimal trajectory by a semi-analytical method yielding the optimal sharing between several stages. The results are checked by a Monte-Carlo simulation modelling a simplified on board guidance. The FPR assessment process has thus been significantly fastened.
Close-range safety is analysed by a preliminary modelling taking into account the features of the debris generated in case of an in-flight launcher destruction. The safety constraints are thus accounted at the early stages of the launcher design. Refined assessments are later performed by statistical methods to check the results.
The upper stage disposal scenario is also systematically assessed within the optimization process whenever the injection perigee lies within the protected LEO region. The required propellant budget is minimized taking into account the fall down safety and the boil-off penalty involved by ballistic legs. The scenario is a posteriori validated during the development by usual Ariane5 methods.
The entire process is integrated in an automated workflow enabling thousands of computations while checking data transfers between engineering domains. Realistic propellant budget models enable to adapt, when necessary, the upper stage propellant loading. Various architectures can thus be assessed very quickly yielding performance derivatives and dispersed trajectories.
This paper will present the whole process, the main technical new developments and illustrative cases.

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Launch Vehicle Design and GNC Sizing with ASTOS

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The European Space Agency is currently involved in several activities related to launch vehicle designs (FLPP, Ariane 6, VEGA evolutions, etc). Within these activities ESA has identified the importance of developing a simulation infrastructure capable of supporting the multi-disciplinary design and preliminary GNC design of different launch vehicle configurations. Astos Solutions has developed the Multi-Disciplinary Optimization (MDO) and Launcher GNC Simulation and Sizing Tool (LGSST) under ESA contract. The functionality is integrated in the ASTOS software and is intended to be used from early design phases up to Phase B1 activities. ASTOS shall enable the user to perform detailed vehicle design tasks and assessment of GNC systems, covering all aspects of rapid configuration and scenario management, sizing of stages, trajectory dependent estimation of structural masses, rigid and flexible body dynamics, navigation, guidance and control, worst case analysis, launch safety analysis, performance analysis and reporting. This paper will present

how the workflow for the vehicle optimization and design of launcher GNC algorithms is realized in ASTOS®, DCAP, ODIN and Matlab/Simulink®. The first step comprises the definition and design of the launch vehicle including the structural mass estimation on substructure level as function of the dimensioning load case. The reference solution is used for the GNC design in Matlab/Simulink® and first performance analysis tasks with rigid body dynamics. In a second step ODIN is used for the finite-element model export, which again can be used as input to the mode-shape export computed by DCAP. Alternatively a free-free-beam model can be utilized in combination with spring damper elements between stages and at engine suspensions. The mode shapes are used for linearization of the ASTOS flexible dynamics and for controller design in MATLAB. In a third step Matlab/Simulink® is used for the controller design re-using as much information as possible from ASTOS and DCAP . Finally, the impact of the launch safety analysis on the launch vehicle design is discussed.

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Model Validation Framework for Launchers: Post-Flight Performance Analysis

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The launchers GNC performance is strictly depending upon its knowledge of the system properties and its evolution throughout the flight. The GNC design and verification integrates mathematical models derived at sub-system level to predict system level behaviour. Their improvement is a key task through the development and validation of a new launch vehicle. Hence the exploitation of flight data shall be maximized in order to reduce at minimum the lack of accuracy in the mathematical models. It is clear that the exploitation of in-flight data during qualification flights in order to improve the launch vehicle models plays a key role in the tuning of the GNC system algorithms and architecture for future flights.
This paper presents the Model Validation Framework for Launchers which is a set of tools and algorithms for the determination and validation of accurate mathematical models and parameters based on pre-, in- and post-flight measurement data.
The Model Validation Framework for Launchers is made up of the following components:

- Measurement Pre-processing component in charge of preparing the flight measurements in order to be used by the Trajectory Reconstruction and Parameter Estimation components.
- Trajectory Reconstruction component in charge of computing the launch vehicle trajectory using the pre-processed flight measurements.
- Parameter Estimation component in charge of characterizing the selected launch vehicle models based on flight measurements, after pre-processing, and vehicle states based on a-priori postulated models.
- Model Validation component in charge of confirming the correctness, accuracy, adequacy and applicability of the identified models with their corresponding estimated parameters.

The target application of the Model Validation Framework for Launchers in this activity has been the VEGA Flight Programme Software Alternative (FPSA).

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Wave-Based Attitude Control of Launch vehicle with Structural Flexibility and Fuel Sloshing Dynamics

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ABSTRACT
The dynamics of a typical launch vehicle are non-linear, non-uniform and time variant. In addition, liquid fuel systems introduce sloshing dynamics which can be difficult or impossible to model, especially if required for real time control. It is a difficult problem, with consequences for the stability, trajectory tracking and efficiency of the launcher. Wave based control has been developed to deal with such difficulties in a natural way and has been shown to cope well with under actuated flexible space systems.
The key concept is that the motion of a flexible body can be separated into two notional components, one travelling from the actuator, for example a TVC engine, into the system, the other leaving the system through the actuator. The actuator, simultaneously launches mechanical waves into a system while it absorbs returning waves. When the launching and absorbing is finished, vibrations launcher have been damped and the desired reference motion is left behind. A wave-based controller is designed for a launcher similar to the European launcher Vega, where the input data to build the launcher model has been taken from publicly available information. In numerical simulations the controller successfully suppresses the disturbance of the launcher body due to structural flexibility and fuel sloshing motion. A major advantage of the strategy is that no direct measurement of the sloshing motion is required. Only measurements of launcher attitude and actuator torque are needed. The work has been carried out as part of ESA Future Launchers Preparatory Programme (FLPP)
Keywords: Spacecraft Dynamics, Launchers, Attitude Control, Flexible Systems, Sloshing, Mechanical Waves
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Indirect Planetary Capture via Periodic Orbits about Libration Points

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The libration points and periodic orbits possess unique dynamics properties in multibody system, which have been exploited to design low-energy transfer for space mission. In this paper, we investigate the periodic orbits for planetary capture and propose an indirect planetary capture method. In new method, the periodic orbit is consider as a park orbit during the capture, which connects with the interplanetary trajectory and the target orbit by stable and unstable manifolds, respectively, at the corresponding periapsides respective to the planet (see in Fig. 1).
The indirect capture method is researched under the background of Mars. Firstly, the dynamic model of CRTBP is established. The libration points and halo orbits for Sun-Mars system is established. Secondly, the candidate halo orbit for capture is determined according to the periapsis distance of invariant manifolds. The amplitudes of halo orbits that suitable for park orbits are given. L1 and L2 halo orbits provide different capture opportunity. The phase angles of invariant manifolds are discussed. Finally, the efficiency of indirect capture method is compared with the direct periapsis capture. The results show that the cost for indirect capture is approximately constant regardless of the periapsis distance (see in Fig. 2). The indirect capture method could save velocity increment significantly at the cost of long transfer time. The new method for

planetary capture can also applied to other planets and provide reference for future exploration mission.

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Coupled dynamics of large space structures in Lagrangian points

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Nowadays interest on large structures, ISS like, to serve for a long time as orbiting outposts place in strategic, possibly long-term stable locations, is increasing. They can serve as a support for far target robotic\manned missions, for planetary tele-operated robotic surface activities, as scientific labs for sample return missions in preserved environment avoiding contamination, for astronauts training, for refueling and maintenance of deep space vessels. Whatever the exploitation is such large structure would undergo numerous docking\undocking activities with a time dependent matrix of inertia; it should require a large lifetime along with orbital stability and, being the structure extended, a strongly coupled attitude\orbital dynamics is expected. Lagrangian points are an evident appealing location for such an infrastructure, offering stable trajectories together with well suited relative positioning with respect to the Sun and the other planets to be considered in the three body system 1.

The investigation of Large Structures coupled dynamics, whenever located in Lagrangian points proximity, is the topic of the paper. The configuration design\6 DOF dynamics coupling is deeply investigated to, eventually, drive the infrastructure system and operational design. Because of the wideness of possible practical applications in the incoming decade, the Earth Moon Lagrangian points system is here considered.

The paper firstly shows the natural periodic orbit-attitude solutions, introducing maps to visually identify the regions where those solutions exist, under the CR3BP approach. The maps are parametrized over the infrastructure inertia properties, and solutions are classified with respect to the number of attitude rotations per orbit. This taxonomy supports preliminary mission design and operations analysis, in verifying the impact of variations in inertia properties (e. g. after docking/undocking of a module) on the attitude and orbital motion. Practical applications of such solutions are discussed, with attention to future missions (e.g. ARM, HERACLES) which could benefit from a full exploitation of the coupled orbit-attitude dynamics. Distant retrograde orbits (DRO) are investigated with greater detail, since their stability properties are appealing for numerous applications. Extension to other orbit classes is then briefly discussed, underlining differences and similarities with the presented results and suggesting new fields of investigation.

The aforementioned discussion introduces then to the rigid body finite extension enhancement in the model, to assess its effect on the orbital motion and analyze modifications in the stability regions. Solar radiation pressure (SRP) disturbance is part of the model enhancing as well, assessing its effects on stability regions as a disturbing action on the whole coupled 6DOF dynamics. Design parameters and drivers are obtained from SRP analysis, with some conclusions on critical aspects and possible requirements for a large structure in the Earth-Moon system. The need of an active attitude control is preliminarily discussed, with attention devoted to requirements needed to guide the control strategy and system design.

Effects of flexibility in the large infrastructure are then introduced in the model; energy principles and linear modal analysis are the mathematical approaches exploited to assess whether orbital and attitude motions could excite/be excited by small vibrations of flexible structures.

Preliminary considerations are deduced from the classical flexible dumbbell model, and are extended to lumped mass models of given complexity.

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Orbit Determination through Global Positioning Systems: A Literature survey of Past and Present Investigations

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Positioning a spacecraft or a satellite in a predefined orbit was and is still considered as one of the most important process of any space mission design for military or commercial purposes. A need to prepare a comprehensive review of investigations conducted in this area was realized. Therefore present research was narrowed down to most significant contributions in the area of Global Positioning Systems applications only. Research work from 1988 to 2015 only has been included in this paper. It was observed that findings from 1988 start off fruitfully, with sub-meter orbit accuracy. Data from orbit determination agrees with Very Long Baseline Inferometry (VLBI) solutions. Most investigations focused on increasing accuracy of prediction through minimization of errors. Various orbit estimation strategies, process noise models for atmospheric fluctuations, combine processing of GPS phase and pseudo-range data were investigated in this period. It was also observed that Orbit modeling is not restricted to one type of orbit rather it included distinct categories. Such as low circular orbiters with altitudes up to a few thousand kilometers; Elliptic orbiters whose perigees are as low as a few hundred kilometers, and apogee's are as high as tens of thousands of kilometers. Finally, highly circular orbits were investigated with altitudes over tens of thousands of kilometers resembling geosynchronous satellites. This paper can be used as foundation for further investigations in Global Positioning Systems.

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The true nature of the equilibrium for geostationary objects, applications to the high area-to-mass ratio debris

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The long-term dynamics of the geostationary (GEO) region has been studied both numerically (Chao, 2005; Anselmo and Pardini, 2007) and analytically (Chao and Baker, 1983; Chao, 2006; Valk et al., 2008; Rosengren and Scheeres, 2013), and some of these results contributed to the IADC guidelines for disposal of objects in the GEO region.
In this work, we revisit the dynamics of this region through the application of canonical perturbation theory, and we apply our results to study the peculiar dynamical behavior of high area-to-mass ratio space debris. More specifically, previous works focused on the evolution of objects *around* a nominal solution called the forced equilibrium solution. Here, instead we focus on the nature of the equilibrium solution itself. Thanks to a higher order normal form, we demonstrate that this equilibrium is actually a lower dimensional object containing slow frequencies. This means that even placed at this pseudo-equilibrium, an object will exhibit periodic variations of its elements, which can be large. We give an analytical expression of these variations, valid for long time scales. To this end, we considered the Hamiltonian of the system accounting for all major perturbations in GEO : the Earth gravitational potential at order and degree 2, the third body perturbations from the Sun and the Moon from Montenbruck and Gill (2000), and the solar radiation pressure. Using canonical perturbation theory, we perform a rigorous averaging of the 8 degrees of freedom Hamiltonian by the method of normal forms via Lie Series (Hori, 1966; Deprit, 1969). The fast terms are then eliminated by a series of canonical transformation, revealing the long-period evolution of the different elements. This allows us to derive the forced equilibrium of this averaged Hamiltonian which is a lower dimensional object containing 5 slow frequencies defining a quasi-periodic orbit, which shows the actual nature of this pseudo–equilibrium. We obtain through

a back-transformation of the canonical transformation made from the forced equilibrium, the analytical time-explicit evolution of all elements at this equilibrium. This analytical result is compared to the numerical integration of the full model before averaging, and gives satisfying accuracy. The long term evolution of the inclination and eccentricity for an object at the equilibrium are particularly analyzed showing strong dependence on the area-to-mass ratio.
We highlight that in addition to the geopotential at order and degree 2, we use a realistic model for the Sun and the Moon from Montenbruck and Gill (2000), where the Moon and the Sun are on elliptical, inclined orbits with a variation of their argument of perigee and right ascension of the ascending node. As noted in Valk et al. (2008), having a fixed Sun-Earth distance in the estimation of solar radiation pressure (an assumption made in previous studies, such as Chao (2006)) would induce spurious long-period terms in eccentricity and inclination evolution. This also ensures that the solar radiation pressure which derives from the position of the Sun is correctly modeled. Another novelty of the approach is that the Hamiltonian is derived in cylindrical coordinates since the geometry of the GEO region is very suitable for this coordinate system, therefore our approach does not need the disturbing function expansions making it simple to develop, and our results are directly translatable in Keplerian elements without singularity. The time-explicit solutions also give direct access to the equilibrium without scanning the whole phase space, and the long-term behavior described by these formulas can be used for disposal studies.

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Orbit prediction of high eccentricity satellites using KS elements

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Highly elliptical Earth satellite orbits are used for communication services, geostationary transfer and space and planetary exploration missions. A major requirement for the efficient mission planning is the precise computation of orbit ephemeris. The orbit evolution mainly pertains to the complex interaction of the Earth's oblateness, Atmospheric drag and luni-solar gravity, which cause variation in the perigee altitude leading to increase or decrease in the satellite lifetime. Classical Newtonian equations yield singularity at the collision of two bodies, which render them unstable for long-term orbit propagation. KS total energy element equations are less sensitive to round off and truncation errors in the numerical integration algorithm and offer a very powerful method to obtain numerical solution with respect to a complex force model. Better accuracy is obtained for in-orbit computations since an orbital frequency is based on total energy and the equations are regular everywhere in the unperturbed case. The equations are smoothed for eccentric orbits because eccentric anomaly is an independent variable.
This paper concerns with the development of an orbit prediction package for high eccentricity satellites using KS elements. Plataforma Solar de Almería (PSA) algorithm and a Fourier series algorithm are used to obtain the accurate position vectors of the Sun and the Moon, respectively. An oblate diurnally varying atmospheric model for drag and zonal harmonic terms up to J6 for oblateness, are considered. Orbit computations for few test cases are carried out using a fourth-order Runge-Kutta-Gill method and the resulting orbital parameters are found to be satisfactory when compared with the observed values. Due to low memory requirement, this package can be used in multiple applications including on-board navigation and guidance software.

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A Simulation Tool to Design of Satellite Formations

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EXTENDED ABSTRACT
A simulation and analysis tool for satellite formation is presented. The tool models the relative motion of a chief-deputy satellite formation, offers ways of selecting proper initial conditions for the formation, and helps examine the results obtained from linear and nonlinear models.
Simulation tool is developed in MATLAB environment providing visualization of the formation. User interface of the tool is created by MATLAB GUI, giving a user friendly environment (Figure 1). In this frame, this tool computes the relative dynamics of the deputy satellite with respect to the chief satellite, calculates the required initial conditions for the deputy satellite for keeping the satellites in the desired formation, and offers the required orbital corrections. Nonlinear relative dynamic model and commonly used linear models are also included in this tool, providing an environment to compare the results of the linear and nonlinear formation design approaches.
Figure 1: Main window of the simulation tool
The tool has three main parts: Pre-processing, Processing and Post-processing. In pre-processing part, user must define the chief's orbital parameters, initial conditions and the inputs required for the simulation. The main window for the deputy has two parts to compute its motion using Keplerian relative motion and using orbital parameters. These two main computations run two separate simulations.
After setting up the initial parameters, the processing part is run in MATLAB SIMULINK environment, to compute the relative motion of the deputy generating the outputs needed for the analysis of the formation. The block diagrams parts of the code are given in Figures 2 and 3.
Figure 2: Block diagram of model that use the Keplerian equations of motion
Figure 3: Block diagram of model with orbital elements.
An orbit propagator based on the Keplarian two-body equations of motion are used to simulate the orbital motion of the spacecraft. The chief's position is computed using an orbit propagator including the variation of the mean classical orbital elements. The perturbations due to non-spherical earth (J2), due to moon and sun are included on the computations. The Kepler's equation states that:
(1) where, M is the mean anomaly, E is the eccentric anomaly, is the angular velocity of the orbital mean motion, is the epoch time, and is the mean anomaly at epoch. The differential equations of motion used to obtain relative dynamics of the deputy satellite with respect to chief satellite are given in Equation-2.
(2a)
(2b)
(2c)
In the above equation, are the components of the disturbance vector, and are the control forces, and and are the relative position of the follower satellite with respect to the leader satellite expressed in leader perifocal frame.
Figure 4: Position Vectors and LVLH Axes Frame
The orbital-period commensurability, energy matching condition and initial orbital conditions are the main terms that should be taken into consideration in for developing an optimal formation keeping scheme. The required initial conditions and the required orbital corrections in order to maintain the formation are computed by using this energy matching concept (Alfriend et. al., 2010).
The relative position of the deputy is also expressed using orbital elements. This method, originally suggested by G.W. Hill, and it has been widely employed in the analysis of relative satellite motion. One of the main advantages of the orbital elements approach is to obtain a non-differential relative position equation and incorporate the orbital perturbations. The deputy's relative position defined using orbital elements is given in Equation-3 and various parameters are shown in Figure 5.
(3)
Figure 5: Orbital Elements
The post-processing part is common for both models. It provides to visualize all the parameters computed by the processing part. Some of the results that may be displayed are the relative distance between two satellites, the projection view of the relative motion, the orbital parameters

(semimajor axis, inclination, etc.), thrust implementation and its value. Here, the orbital motion in a 2D Earth map and in a 3D Earth model are also used to visualize the motion (Figure 6) Furthermore, it is possible to visualize the motion by running 3D Animation. Here, the 3D motion is given with respect to ECI and ECEF frames, both in space view or satellite view (Figure 7).
Figure 6: 2D Mapping
Figure 7: 3D Animation
In the final manuscript, the details of the simulation tool will be provided. Final manuscript will focus especially on the selection of the initial conditions defined in terms of orbital elements of the chief and deputy satellites. The linear procedures used for determining the initial conditions for relative motion consider near circular orbit and they are valid for orbits with small eccentricities and even for close formations (i.e., the distance between satellites is about 1 km). However, for formations with relative distances greater than 10 km, it is not possible to obtain a stable formation by using the initial conditions computed using linearized schemes. In this case, nonlinear effects should be added to the initial conditions predictions. In the final manuscript, we will present the orbit analysis tool in detail. Also to be presented is a new method that incorporates the nonlinear effects in the selection of initial conditions in terms of orbital parameters. The approach taken will be described in detail, and the success of the approach will be demonstrated through simulations.
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Space Situational Awareness Capabilities of the Draper Semi-analytical Satellite Theory

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The fundamental requirement of space situational awareness (SSA) is to provide actionable knowledge about events and activities in Earth orbit. A key component of SSA is space surveillance – determining the present position of space objects and the ability to predict their future orbital paths. Related requirements are the detection of new space objects, the detection of spacecraft maneuvers, and the prediction of when one space object may interfere with another space object. Such interference may be physical or electromagnetic in nature. The Draper Semi-analytical Satellite Theory (DSST) is based on based on a synergistic combination of techniques from analytical satellite theory and from traditional numerical integration. The conventional equations of motion and the associated numerical processes are replaced with: 1. Equations of motion for the mean elements, 2. Expressions for the short periodic motion, 3. Semi-analytical theory for the partial derivatives, 4. Semi-analytical theory truncation algorithms, and 5. Interpolation for the mean elements, mean element state transition matrix, mean, mean element partial derivatives, and the Fourier coefficients in the short-periodic expansions. There are three comprehensive implementations of the DSST: 1. GTDS Orbit Determination system (Fortran 77) 2. DSST

Standalone Orbit Propagator (re-factored Fortran 77) 3. Orekit open source DSST (java) GTDS was the original development platform for the DSST. The DSST Standalone Orbit Propagator (F77) makes the DSST algorithms available without the overhead of GTDS. The Orekit DSST demonstrates portability of the DSST algorithms to an object-oriented software platform. The DSST is very desirable for Space Situational Awareness due to: Comprehensive force models The large grid size employed to integrate the mean element equations of motion and to compute the short-period Fourier coefficients The capability to tailor the semi-analytical theory force models at execution time. The compactness of finite time interval ephemeris representations for multiple space objects orbit estimation algorithms (both batch and recursive) built to estimate the mean elements directly from the tracking data The near linearity of the mean element equations of motion is very desirable from the perspective of propagating realistic uncertainties The structure of the DSST is very amenable to parallelization via modern day computing hardware and software techniques The present study analyzes the DSST for the purpose of Space Situational Awareness. The study is divided into two parts. The first part deals with the error analysis of a semi-analytical batch least square orbit determination process. The second part of the study deals with uncertainty realism and the propagation of uncertainties over time using the DSST. Uncertainty realism for SSA requires a proper characterization of a space object's full state probability density function (pdf) in order to faithfully represent the statistical errors. DSST covariance propagation using the unscented transformation is discussed.

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Space dynamics software ELECTRA

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The objective of this article is to present the current state of the CNES flight dynamics software ELECTRA maintained by Capgemini and some of the main principles used for validating its scientific and computational facets in an industrial context.
First of all, we will describe the development history. Some pieces of software have been created by industry. Other parts have been developed initially by research teams, based on the ELECTRA method the CNES developed, followed by an industrialization process to meet industrial standards and code quality requirements. Following the prototype phase, the development of the operational tool ELECTRA was decided by CNES Steering Committee of Safeguard Working Group in July 2006 and started in 2007. At the beginning, ELECTRA was implemented for internal CNES safety needs, but it was soon provided to space operators to assess victim risk associated with their operations and comply with the French Space Operations Act. Since December 2010, ELECTRA has been deployed and used operationally to monitor the risk associated with each launch from Guyana Space Centre in place of the existing "SUZHANE" tool. ELECTRA was written in FORTRAN. In 2015, it was decided to port ELECTRA in JAVA.
Secondly, we will describe the functional coverage of ELECTRA. The ELECTRA software computes the risk of making a victim during atmospheric reentries, with or without taking into account protection coefficients in several contexts (Random (or Uncontrolled) reentry, Controlled reentry, launch failure, Reentry on final orbit). ELECTRA computes two complementary estimations of the risk, the probability of causing at least one victim and the expected value of the number of victims, taking into account fragmentation of the spatial vehicle after atmosphere re-entry. The risk computation is done by assessment of fragment impact location and probability of occurrence and consideration of population distribution and habitat protection.
Third, a description of the underlying architecture of the Fortran and JAVA software components will be proposed: different libraries, the role of the different libraries, how each library has been constructed to meet needs of modularity and re-use. We will also point out some limitations of the existing architecture, and pit-falls which should be avoided in JAVA future developments.
Then, we will explain the methods and principles used for validating the new version of ELECTRA (JAVA). Two different and complementary approaches are used: non-regression based on comparison with the previous version developed in Fortran language, and "from scratch" validation in case of major evolutions or implementation of new functionalities. Using examples, we will

present the mechanisms, tools and documents used for following the validation of the tool through different versions, and some principles and feed-back of validating new functionalities.
As a conclusion, we will present what seems to us to be the key topics for developing and maintaining the flight dynamics software ELECTRA in terms of architecture and validation, and how this could be taken into account in developing new products.

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Computer Graphics for Space Debris

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The number of resident space objects re-entering the atmosphere is expected to rise with increased space activity over recent years and future projections. Predicting the probable survival and impact location of the medium to large sized re-entering objects becomes important as they can cause on-ground casualties. We present development and application of a new tool for quick estimation of aerodynamics and aerothermodynamics properties of the re-entering objects. The novel method uses primitive geometries to develop a complex object and voxelization (voxels are the 3D equivalent of pixels in 2D images) for computing the shading/visibility factors to quickly estimate the aerodynamic and aerothermodynamics properties of the re-entering object. The tool can be used as a module of object oriented codes to support preliminary End of Life analyses.

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Debris de-tumbling & de-orbiting by elastic tether & wave-based control

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We consider the problem of guidance and control of a completely passive, target, piece of debris, using an actively-controlled, "chaser" spacecraft, connected to the debris by an elastic tether. Compared with robotic capture, the use of an elastic tether reduces the need for precise chaser-target coordination during capture, reduces collision risk during capture, simplifies subsequent de-tumbling, makes for lighter and mechanically simpler systems, and limits subsequent jerk with its potentially destructive consequences. The target debris is likely to be tumbling initially, so the chaser must achieve active de-tumbling before, or during, further manoeuvres of the "stack" (debris-tether-chaser), such as imposing a delta-V in order to de-orbit. The preceding task, of attachment of the tether to the debris, could be achieved in various ways, including by net capture or harpoon. A novel net capture technique will be briefly presented, but the techniques considered here will apply regardless of the attachment method. Note that some proposed capture methods require a separate de-tumbling prior to capture, whereas methods using an elastic tether can de-tumble after capture, thereby reducing mission complexity, uncertainty and mass. The first control challenge, then, is precisely due to debris tumbling prior to capture. If there are energy loss mechanisms on board, the debris rotation will probably be mainly around the axis of maximum moment of inertia. If so, and if the dominant debris rotation axis is approximately aligned with the tether axis, its main effect after capture will be a twisting action on the tether. If the rotation axis is approximately perpendicular to the tether, the debris will tend to wind up along the tether. In either case, the chaser needs to respond so as to reduce the tumbling towards zero over time, while

imposing control and avoiding tether entanglement and debris-chaser collisions. The general case can be a dynamic combination of these two extreme cases (with spin, nutation and precession), with possible slippage during any wrapping. Aside from de-tumbling, a further control requirement is to change the velocity of the debris-tether-chaser stack by a target mount in a target direction, while again avoiding entanglement, collisions, and uncontrolled pendular and libation motions, both during and after this delta-V manoeuvre. There may also be a subsequent requirement to allow the stack to coast for a relatively long period while ensuring the tether stays fully extended, to avoid entanglement during the period, and to be ready for a subsequent stack manoeuvre. The paper will show how wave-based control easily meets all these requirements using a single control strategy, combining position control and active vibration damping. The chaser-debris interaction is modelled as two-way wave motion, longitudinal (stretching) or rotational (twisting), travelling back and forth along the tether between chaser and debris. It can be shown that the effect of the control is to make the actuator behave as a viscous damper (motion absorber) to wave motion travelling in the tether from the debris towards the chaser. In this way when the tether is under tension the debris experiences a force and torque due to the tether which appears to have a viscous damper to ground at the far end, causing it to tend exponentially towards its natural (unstretched) length, and/or zero-twist condition. Meanwhile the chaser can simultaneously impose a desired velocity (or attitude) using an outgoing wave. The details of this combined control action and its implementation will be explained. Its advantages over classical control approaches include being simple to implement; very robust to wide variations in the debris parameters (which may be unknown); requiring minimal sensing; and all sensing can be done at the chaser end. The technique is tested in a detailed computer simulation of a complete de-tumbling of an arbitrarily shaped structure undergoing arbitrary tumbling motion. Then a complete de-orbit of the Envisat satellite to a target location is simulated and tested, and parameters such as fuel requirements assessed. The control technique was also tested on hardware using two model hovercrafts, one remotely controlled. Keywords: Debris removal, elastic tether, stack control, de-tumbling, mechanical waves, GNC

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An Access Point to ESA's Space Debris Data: The Space Debris Office Web Based Tools

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With DISCOS (Database and Information System Characterising Objects in Space) ESA's Space Debris Office has a very powerful database in hands when it comes to space debris related analyses. It serves as a single-source reference for launch information, object registration details, launch vehicle descriptions, spacecraft information (e.g. size, mass, shape, mission objectives, owner), as

well as orbital data histories for all trackable objects, which sum up to more than 40000 object entries.
Based on DISCOS and USSTRATCOM TLEs, the Space Debris Office routinely predicts upcoming re-entries as well as performs detailed analyses on high interest re-entries and ad-hoc risk assessments to missions after severe fragmentation events. To support these processes, the Space Debris Office also does their own solar activity prediction, based on publicly available solar activity data, with the SOLMAG tool.
All this data is of high interest not only within ESA but to the whole space flight community. It can be a valuable asset for analyses and operational processes, including but not limited to Space Debris related studies and collision avoidance. A reliable and controlled access to this information with maintained data quality is thus fundamental for the community. To accomplish this, the Space Debris Office is currently developing web based tools for DISCOS data access, including a machine friendly REST API, fragmentation analyses, and re-entry predictions. These will be complemented by web pages of more static nature, like the SOLMAG solar activity predictions, and of course by the already established Space Debris User Portal (https://sdup.esoc.esa.int) serving as distribution point for ESA's risk and mitigation analysis tools MASTER (Meteoroid and Space Debris Terrestrial Environment Reference), DRAMA (Debris Risk Assessment and Mitigation Analysis), and ORIUNDO (On-ground Risk Estimation for Uncontrolled Re-entries Tool).
This paper will introduce all web based tools under development and outline their function. The design will be addressed with a focus on user friendliness, function and harmonised look-and-feel. Special emphasis is put on security to not only protect ESA's data and server infrastructure, but also to implement ESA's data access and usage policy.

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From end-of-life to impact on ground: An overview of ESA's tools and techniques to predicted re-entries from the operational orbit down to the Earth's surface

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Currently, there are about 17 000 tracked objects in Earth orbit, out of which approximately 7500 are expected to have a remaining orbital lifetime of less than 100 years. Out of those 7500, about 1250 have a mass of more than 1 kg, but the vast majority are smaller pieces of space debris.
Once an object in Earth orbit has reached the end of its operational life, or in case of a space debris object after its genesis, it enters into the re-entry prediction system of ESA's Space Debris Office. This system automatically predicts the remaining orbit lifetime; In case of a short remaining orbit lifetime it automatically predicts the impact location, risk and associated uncertainties; In case of a high risk re-entry event it enables the in-depth analysis of the affected regions and atmospheric break-up of the object; And tools are available for post-event processing of the observational data. The results of this analysis chain are provided to the relevant actors, e.g. national alert centres or operators, either automatically or on-demand.
In this paper we present the status of the re-entry prediction system, and its orbit determination, orbit propagation, environment forecasting, and risk assessment methodologies related to the

orbital lifetime, re-entry location, and atmospheric break-up predictions. Uncertainties depending on the orbit regions, object type, and step in the re-entry prediction system are derived and used to tune the service to provide the best possible results over the entire population. In the last step of the system, these automatically generated results are complemented with an operator review of the available data to provide re-entry and break-up prediction for individual objects, occasionally complemented by processing dedicated observations of the re-entry object. Post-mortem analyses, e.g. after a confirmed re-entry, are performed for selected objects in order to explain potential observations of the re-entry event and retrieved samples on-ground. The entire data collection is a relevant source which serves as input for break-up modelling tools, and improvement of the entire prediction service.

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Optimization and tools for deployment and reconfiguration of formation flying Missions

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Formation flying plays an important role in future space missions. Formation deployment and reconfiguration are key technologies of formation flying missions. Optimal multiple impulsive formation deployment and reconfiguration are studied in this paper, software design tool is developed.Firstly, Efficiency of tangential, radial and normal direction maneuvers is analyzed. Secondly, optimal tangential maneuvers for formation deployment and reconfiguration are studied, mathematical models of three and four impulsive deployment and reconfiguration are derived. Initial guesses are studied for solutions of three and four impulsive maneuver model. Three impulsive maneuver problem is resolved by differential corrections method. Optimal three impulsive maneuver solution are obtained using the continuation method. Four impulsive maneuver problem is resolved by analytical method. Global optimal four impulsive maneuver solution is obtained by genetic algorithm. Symmetric deployment trajectories are proposed to avoid the attitude disturbance of mother satellite.Thirdly, multiple impulsive maneuver model without maneuver direction constraints is derived and optimized using genetic algorithm.Fourthly, optimal semi-analytical solutions for formation deployment and reconfiguration are studied.Finally, the algorithms are integrated in a design software tools. we won the first prize in the 7th Chinese Space Trajectory Design Competition with the aid of the design tool.

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An Object-Oriented multidisciplinary simulation framework for space dynamics and space tether simulation

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Tethers have been used in space for a number of different purposes: generation of artificial gravity, formation flying, propulsion, etc. Lately, space situational awareness has fostered the study of tethers in the context of space debris mitigation and removal, as is the case of the promising capabilities of electrodynamic tethers to de-orbit a satellite efficiently. In other approaches, tethers are considered as parts of more complex devices such as harpoons or hooks.

The dynamics of tethered spacecraft involve nonlinear effects with coupling between orbital motion and longitudinal and lateral modes, and show a extremely rich behavior including periodic

and unstable motions. Moreover, tether dynamics can be coupled with other spacecraft systems such as AOCS and the power system.
Although different approaches have been proposed to address the simulation of space tethers, there is still a need to simulate the dynamic behavior with the required accuracy and an acceptable time consumption. There is also a need for modeling flexibility for simulating different complex multi-tethered configurations or the coupling with other spacecraft systems.
This work is intended to meet this need by using a new tether dynamical model and a non-casual and object-oriented modelling approach. To this end, the SDG is developing a set of simulation libraries for the multidisciplinary simulation tool EcosimPro. Although the focus of the libraries is the simulation of tether space systems, common space dynamics functionalities such as orbit propagation, satellite attitude dynamics, formation flying, AOCS simulation, etc. are also included in the libraries.
The tether dynamical model, which was presented in previous works, is briefly described. It is based on the discretization of the tether in a number of elastic rods, and allows simulating multi-tethered satellite systems with central and ending masses modeled as point masses or rigid bodies.
After the description of the structure of the libraries, the orbit propagators, and the validation process, several simulation cases are presented to show the capabilities of the simulator for trajectory propagation, AOCS simulation, coupled simulation of satellite motion with other systems, and multi-tethered satellite simulation. The work concludes by presenting a complete de-orbiting mission of an electrodynamic tethered system.

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Joint Optimization of Main Design Parameters of Electric Propulsion System and Spacecraft Trajectory

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The problems of joint optimization of the main design parameters of electric propulsion, power supply systems and trajectory of the spacecraft is considered in two formulations: minimum-thrust problem and maximum payload mass problem under the optimal parameters of the electric propulsion system (thrust, exhaust velocity, power). The first problem allows solving one of the fundamental problems in optimization of trajectories for the spacecraft with finite thrust related with the existence of the solution which is one of the reasons for the complexity of the constructing robust and efficient numerical optimization technique. Indeed, if the numerical scheme does not converge to a solution then the real reason is unknown since it can be either the absence of a solution or the numerical scheme failure. Therefore, identification of the boundary of the solution region is an actual problem. In this way, solving the minimum thrust problem gives necessary assessment. The minimum thrust problem is formulated similar to any other low thrust trajectory optimization problem but the functional to minimize is the thrust. It is supposed that the thrust can be either maximal or zero, the maximal thrust magnitude is constant along the trajectory and its direction is unconstrained. The transfer time is fixed. The second problem is directly related to the optimization of the parameters of the propulsion system in the region of the existence of solutions. It is well known that for every space transportation operation there is an optimal value of specific impulse corresponding to the minimum total mass of electric propulsion system, power supply system providing electric propulsion operation and propellant. It is easy to show that there is also an optimal value of electric power of the electric propulsion system that being associated with the growth of the required characteristic velocity while reducing the thrust. It is obvious that the optimal values of specific impulse and electric power of the main electric propulsion system can be found by joint optimization of the trajectory and design parameters. An approximated design model of the spacecraft equipped with main electric propulsion system and the Pontryagin's maximum principle are used for optimization. The necessary conditions for the optimality of specific impulse and electric power of the electric propulsion system are derived. Numerical examples of the joint optimization for the interplanetary spacecraft trajectory are presented.

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A Self-Boundary Fall Free Algorithm for 2D Open Dimension Rectangle Packing Problem of Satellite Module

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Layout design of spacecraft module belongs to scheme design problem, which has been proved to be NP hard. This problem has not only computing complexity but also engineering complexity, and it is more difficult to tackle the challenge of practical application in engineering. In practical engineering, the dimension of satellite configuration is usually unknown and needs to be optimized (generally minimized) as well, while the dimensions of satellite components are known. Assumed that the material densities of all modules are same, the layout design of satellite module can be simplified to a 2D open dimension rectangle packing problem when the satellite configuration is cuboid. In this problem, given a set of rectangles with known dimensions, the arrangement of these rectangles should be determined without overlapping and inside a predefined enveloping rectangle. This paper proposed a self-boundary fall free genetic algorithm (GA&SBFFA) for the open dimension rectangle packing problem, which is used to optimally arrange the rectangles densely and minimize the area of enveloping rectangle. Meanwhile, the shape of the enveloping rectangle is maintained as square as possible so as to satisfy the static equilibrium requirement in some complex system design, e.g. satellite. The main procedure of SBFFA is as follows. First, the dimension of enveloping rectangle is determined by the packing items, which is different from the methods published previously. Next, the information of feasible space where the next item can be placed is recorded, which includes the widths of the feasible spaces and the coordinates of the feasible points where the bottom left vertex of the next item can be placed. Finally, the next item is placed according to the principle of minimal potential energy. Based on SBFFA, the minimum enveloping rectangle space is calculated with given sequence of packing items. GA is used to solve the packing optimization problem by searching the optimal item sequence. Two experiments are used to testify the proposed method and the efficacy is demonstrated. The computational expenses were reduced to about 30 seconds when there are 50 items, which is much less than the reported methods. Keywords: Packing problem; Layout optimization; Area minimization. Layout design of spacecraft module belongs to scheme design problem, which has been proved to be NP hard. This problem has not only computing complexity but also engineering complexity, and it is more difficult to tackle the challenge of practical application in engineering. In practical engineering, the dimension of satellite configuration is usually unknown and needs to be optimized (generally minimized) as well, while the dimensions of satellite components are known. Assumed that the material densities of all modules are same, the layout design of satellite module can be simplified to a 2D open dimension rectangle packing problem when the satellite configuration is cuboid. In this problem, given a set of rectangles with known dimensions, the arrangement of these rectangles should be determined without overlapping and inside a predefined enveloping rectangle. This paper proposed a self-boundary fall free genetic algorithm (GA&SBFFA) for the open dimension rectangle packing problem, which is used to optimally arrange the rectangles densely and minimize the area of enveloping rectangle. Meanwhile, the shape of the enveloping rectangle is maintained as square as possible so as to satisfy the static equilibrium requirement in some complex system design, e.g. satellite. The main procedure of SBFFA is as follows. First, the dimension of enveloping rectangle is determined by the packing items, which is different from the methods published previously. Next, the information of feasible space where the next item can be placed is recorded, which includes the widths of the feasible space and the coordinates of the feasible points where the bottom left vertex of the next item can be placed. Finally, the next item is placed according to the principle of minimal potential energy. Based on SBFFA, the minimum enveloping rectangle space is calculated with given sequence of packing items. GA is used to solve the packing optimization problem by searching the optimal item sequence. Two experiments are used to testify the proposed method and the efficacy is demonstrated. The computational expenses were reduced to about 30 seconds when there are 50 items, which is much less than the reported methods. Keywords: Packing problem; Layout optimization; Area minimization.

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Trajectory and Systems Design for Low-Thrust Interplane-

tary Missions via Multi-Objective Hybrid Optimal Control

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Preliminary design of interplanetary missions is a highly complex process. The designer must choose the launch date, flight time, propulsive maneuvers, and possibly a sequence of planetary flybys as well as altitudes and velocity vectors for each of those flybys. Low-thrust missions add an additional degree of complexity because the designer must also choose a time history of control variables, i.e. thrust magnitude and direction, which define the trajectory. For some types of missions, such as missions to asteroids and comets, the designer is also responsible for choosing the destination because the customer is interested in a population of bodies rather than a specific body. In addition, low-thrust trajectory design is tightly coupled with spacecraft systems design because the propulsion and power capabilities of the spacecraft strongly drive the available trajectory options. Therefore, the choice of system parameters is an integral part of the low-thrust preliminary design problem. Furthermore, a mission designer does not work in isolation - the customer who commissions the work, usually a scientist, does not just want a point solution even if that solution is globally optimal in propellant use, time, or some other metric. Rather, an exploration of the trade space between several metrics of the customer's choice is the true goal of preliminary mission design.
The method of multi-objective hybrid optimal control (MOHOCP) will be presented in this work. In MOHOCP, the mission design problem is separated into two nested loops – an "outer-loop" which chooses the sequence of bodies to be visited and also the systems parameters of the spacecraft, and an "inner-loop" which searches for the globally optimal trajectory corresponding to each candidate outer-loop solution. A "cap and optimize" approach is used where one of the outer-loop's objective functions becomes the objective of the inner-loop solver and the other outer-loop objective functions are represented in the inner-loop as constraints or problem assumptions. In this work a version of Nondominated Sorting Genetic Algorithm II (NSGA-II) is used as the multi-objective outer-loop solver and Monotonic Basin Hopping (MBH) coupled with a nonlinear programming (NLP) solver is used to solve the inner-loop.
The algorithm presented here is the core of NASA Goddard's Evolutionary Mission Trajectory Generator (EMTG), an open-source tool for preliminary design of interplanetary missions. In this work the MOHOCP algorithm will be described in detail and a worked example of a combined interplanetary trajectory and systems design problem will be presented.

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Spiderman Spacecraft: Tethered Asteroid Hopping in the Main Belt

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The use of the gravity of celestial bodies for gravity assist maneuvers is quite common in astrodynamics. The spacecraft gravitationally interacts with the celestial body in such a way as to provide the desired delta-v to the spacecraft. While this works well for large bodies such as planets, the gravitational attraction of small bodies, such as asteroids, is typically too small to perform such maneuvers.
In this work we analyze a different type of fly-by orbit using a variable length tether. During a fly-by of the spacecraft at an asteroid, a tether is attached to the asteroid which is then reeled out maintaining a tension within the limits of the tether. A dynamical model describing such

a tethered flyby is developed. Using this model, it is possible to perform fly-by maneuvers that conceptually are very similar to traditional gravity assist maneuvers. While current tether technology is not yet able to provide the required tension, these fly-bys provide a new class of orbits that can be utilized in future for missions.
We demonstrate a potential use of such tethered fly-by maneuvers for multiple rendezvous orbits. A spacecraft in the main asteroid belt has to visit a long sequence of asteroids via tethered fly-bys. The optimization is performed with the goal of maximizing the number of asteroids visited in a given time frame. The computation is performed using the PyKEP toolbox developed by the Advanced Concepts Team. Due to its flexible design, this Python based toolbox can easily be to allow for optimization of missions using these novel fly-by dynamics. The global optimization is performed using either evolutionary algorithms or a tree search strategy.

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Design Formation Flying in the small satellite class

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The exploitation of cooperative satellites flying in close formation is an appealing technology with potential to overcome some limitations intrinsic to the single platform satellite systems. Scientific applications go from the effective implementation of single pass Synthetic Aperture Radar (SAR) interferometers for high accurate planetary digital elevation models generation, to the implementation of dual(multi)-spacecraft telescopes. Moreover, the satellite formation concept brought the key benefit of distributing the scientific payload among the satellites of the formation, with mass and power demand reduction on each single space segment; this offers the chance to exploit small and even micro/nano-satellite satellite class . Those classes offer an economical benefit from one side, being the currently available technology performance the main limitation as far as the micro/nano-sats are considered.
In this framework, the authors aim to set up a software tool to support the satellite formation architecture definition, depending on the specific scientific/commercial mission requirements. Flexibility is one of the key drivers of the tool here presented, to concurrently handle the constraints related to the specific satellite class of interest together with the imposed mission objectives requirements.
The assessment among possible formation architectures takes into account a wide range of design variables, as for example: the platform technology, the reference orbit, the number of satellites to set up the formation, the geometry for the spacecraft relative configuration. Moreover, the tool leans on a spacecraft 6 DOF dynamics modeling, fast and accurate, to exploit at the best the environmental perturbation to drive the formation flying architecture design to minimize the active control. This assessment procedure is integrated in a multi-objective optimization that takes as input the high-level mission requirements and constraints, and gives as output the formation sizing parameters that maximize the required performances.
Considering the SAR interferometry application. The vector of decision variables to be optimized (x^*) is composed, for example: by the most relevant sizing parameters of the Radar system (e.g. antenna dimensions, frequency of transmission and associated bandwidth, maximum angle of incidence of the Radar signal at the ground, the pulse repetition frequency of the system), by the formation geometry sizing parameters (e.g. number of satellites, initial conditions of the relative motion) and by some satellite characteristics (e.g. ballistic coefficients of the different elements of the formation). Other parameters (p) that characterize either the Radar system or the satellites are considered fixed (e.g. the Radar system temperature, ground back scattering coefficient, reflectivity property of the satellite surfaces). The cost function vector (J) is composed by the most relevant resolution performance indexes of both cross-track and along-track SAR interferometers (e.g. phase height sensitivity factor, ambiguity height, variety of baselines provided by the formation over a specific planetary area of interest). Finally, the fundamental constraints (g) are imposed (e.g. minimum antenna area, maximum admissible cross-track and along-track

baselines to not incur in decorrelation, minimum admissible safe separation of the satellites, maximum power available).
As a first case study, useful to identify the key variables for the tool, a formation flying concept around the Saturn's moon Titan for complete bi-static SAR mapping is here presented and critically discussed. The solution exploits the natural perturbations around Titan to provide a big variety of baselines at all planetary latitudes and therefore, high resolution and unambiguous digital elevation model of the whole Titan surface, while minimizing the fuel consumption necessary for the formation control (45 m/s of total ΔV during 600 days of planetary mapping for both formation and station keeping manoeuvers).
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Spacecraft formation control using analytical integration of Gauss variational equations

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This paper derives a control concept for far range Formation Flight (FF) applications assuming circular reference orbits. The paper focuses on a general impulsive control concept for FF which is then extended to the more realistic case of non-impulsive thrust maneuvers. The control concept uses a description of the FF in relative orbital elements (ROE) instead of the classical Cartesian description since the ROE provide a direct access to key aspects of the relative motion and is particularly suitable for relative orbit control purposes and collision avoidance analyses.
Although Gauss' planetary equations have been first derived to offer a mathematical tool for processing orbit perturbations, they are suitable for several different applications. If the perturbation acceleration is due to a control thrust, Gauss' variational equations show what effect such a control thrust would have on the keplerian orbital elements. Integrating the Gauss' variational equations offers a direct relation between velocity increments in the local vertical local horizontal (LVLH) frame and the subsequent change of keplerian orbital elements.
For proximity operations, these equations can be generalized from describing the motion of single spacecraft to the description of the relative motion of two spacecraft. This will be shown for impulsive and finite-duration maneuvers. Based on that, an analytical tool to estimate the error induced through impulsive planning is presented. The resulting control schemes are simple and effective and thus also suitable for on-board implementation. Simulation results show that the proposed concept improves the timing of the thrust maneuver executions and thus reduces the residual error of the formation control.

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ASTOS 8.1 - Mission Performance Analysis, System Concept Analysis and Other New Features

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The ASTOS software was originally designed as a trajectory optimization tool for launchers. This was more than 25 years ago. Since then the software had been continuously developed towards a multi-purpose analysis, simulation and optimization software for nearly all kind of space scenarios. This paper shall present the latest extensions of ASTOS incorporated into release 8.1. The paper will first give an overview on ASTOS and its workflow. Then it will present the new capabilities. Four new major developments have been made for the release 8.1 of ASTOS focussing on new markets and increased usability: 1) An integrated set of mission performance analysis features that cover almost all aspects required to fill a mission analysis report according to DOPS-GS-RM-1002-OPS-OSA. 2) Models and techniques for system concept analysis considering the power system, thermal control system as well as data management and communication systems. 3) A built-in tool to import CAD models, to easily texturize them and to use them for the built-in animation tool. 4) New wizards that ease the set-up of an ASTOS scenario and that help migrating a scenario from one to another application. The mission analysis capabilities of ASTOS cover launch, LEOP, operational phase and post-mission disposal, whereas mission analysis aspects like operational life-time prediction, eclipse analysis, collision avoidance, deorbit and graveyard manoeuvres, station keeping, etc. are considered. The results are automatically written into the mission analysis report template. For the system concept analysis dedicated

models for solar arrays, batteries, PCDUs, data storages have been implemented. On the other hand all existing models, e.g. for actuators or sensors have been extended by thermal, data and power system aspects. These individual components can be linked into a system using a node model. In case of the thermal system each node can be linked to a surface element or another node, whereas each connection is characterized by its thermal resistance. For a realistic animation of a scenario it is essential to have a detailed geometry model of the vehicle. These models are typically available as CAD files. Those models lack texture and relevant surface material properties. These features are typically added using expert tools like Maya, Cinema4D or Blender. In order to reduce this effort ASTOS 8.1 provides an easy-to-use texturing tool that comprises a database with typical spacecraft surface materials. Due to the flexibility of ASTOS, it requires several steps to create a scenario in the classical way. This effort is significantly reduced by the new wizards. Based a question tree these wizards either create a new scenario or reduce the effort to migrate a scenario, e.g. from an optimization towards a guidance reference scenario. The paper will detail on each of the above mentioned new functionalities, showing user input and results taken from typical example scenarios.

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Capitalizing on Relative Motion in Electrostatic Detumble of Axi-Symmetric GEO Objects

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Touchless methods of actuating and detumbling large Earth-orbiting objects is of increasing importance for active debris mitigation strategies. Previously developed are electrostatic detumble dynamics and simulations for deep-space and lead-follower formations. This study investigates the influence of the instantaneous position and relative formation on electrostatic detumble performance. Developed are the mathematical sensitivities to relative position to enable optimal relative guidance studies. The newly developed Linearized Relative Orbit Elements (LROE) formation flying controller is exhibited and applied for formation maintenance. The benefits of formation flying in electrostatic detumble scenarios and the advantages of the LROE controller for electrostatic actuation applications are demonstrated through numerical simulation.

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Analysis of Electric Propulsion Capabilities in Establishment and Keeping of Formation Flying Nanosatellites

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Nanosatellite clusters constitute one of the current and future trends in space technology. 1 These clusters can be used in a variety of applications including search and rescue, communication, earth monitoring, etc. In order to maintain a satellite cluster over a long period of time it is required that the inter-satellite distances between the cluster agents will be controlled. 1 The nanosatellites need to mitigate the along-track drift created by the initial orbit injection, using the very limited resources available onboard. In the mass range of $1 - 10$ kg, Cubesats have strict constraints on allowed mass, volume, electrical power, and carry only limited sensor and actuator capability. 1 This paper explores the capability of state of the art miniaturized electric propulsion (EP) systems to establish and maintain a nanosatellite cluster mission. Different EP technologies will be considered. Due to the low thrust provided by an EP system, long orbit control maneuvers are required. Therefore, mission design is highly effected by long-term attitude and power constraints. In contrast to previous work that either ignored the attitude limitations or

assumed that the satellite orientation is dedicated to the orbit control maneuver, 1 the proposed paper will focus on developing an autonomous controller that can work under realistic attitude and mission constraints. The paper compares 6 types of satellite body forms, with deployable and body fixed solar panels, each with its own constraints on mass, volume, and power. The paper will present the results of the analysis and provide a conclusion regarding the possibility of current EP technology to be applied for maintaining a nanosatellite cluster.
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Robust Control Design Methodology of the 6DoF Flight-Formation of PROBA-3

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PROBA-3 is one of ESA's technology demonstrators, which consists of two spacecraft flying in close formation in a highly elliptic orbit. Its aim is to create a Sun coronagraph instrument with the optical payload and the external occulter disc mounted on different spacecraft. One of its main challenges is to develop the GNC that maintains a flight-formation during the apogee arc with very stringent attitude and relative position requirements. The flight-formation consists of aligning both spacecraft with the Sun vector, while maintaining a fixed distance with sub-millimetre precision. In addition, various technology demonstration manoeuvres will be performed for virtual structure simulation (rotating the formation, while maintaining a fixed distance), and virtual telescope focusing (resizing the formation, while keeping the Sun vector aligned).
The control problem has been posed as a coupled 6DoF multi-input multi-output (MIMO) design task, which simultaneously takes care of both the attitude and the position. To meet the design specifications, state-of-the-art H_∞- and μ-synthesis techniques have been utilized to obtain a robust controller. In addition, in-house developed simulation tools have been employed for validation purposes in a non-linear environment.
In this paper the step-by-step design techniques and accompanying tools used by SENER are presented, starting with the formulation of the control problem, working towards the justification of the designed controller, and resulting in a fully validated product both in frequency and time domain for integration with the full PROBA-3 system.

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Evaluation of satellite aerodynamic and radiation pressure acceleration models using accelerometer data

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Models of non-gravitational accelerations, of which satellite aerodynamics and radiation pressure are the most important examples, are critical for many orbit determination and prediction

applications. Such models typically consist of three parts, each of which can be implemented at various levels of sophistication, depending on the required accuracy of the application. The first part consists of a model of the environment, such as the density of the atmospheric particles, or the direction and magnitude of the photon flux coming from the Sun and Earth. The second part is a model of the geometry and material properties of the satellite's outer surfaces, while the third part is a representation of the interaction between the particles and the surfaces. In its most simple form, these last two parts combined can be expressed in terms of a constant satellite ballistic coefficient.
Traditionally, the implementation of non-gravitational models in astrodynamics tools is based on a semi-empirical approach, and their assessment is based on an evaluation of tracking data residuals. The accelerometers on the CHAMP, GRACE, GOCE and Swarm satellites, however, measure the sum of the non-gravitational accelerations directly. The combination of these observations with non-gravitational acceleration models has led to the availability of thermospheric data sets with many scientific applications in the field of aeronomy.
In this paper, the experience obtained with the processing of accelerometer data and the use of non-gravitational force models for aeronomy applications is demonstrated, and applied in order to provide useful pointers for the implementation of such models in orbit determination and prediction tools at various levels of complexity and accuracy.

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Using the attitude response of aerostable spacecraft to determine thermospheric wind

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The residual atmospheric density present at orbital altitudes produces an aerodynamic disturbance, both to satellite orbits and attitude, that at Very Low Earth Orbits (<450 km) can be significant, usually being the strongest disturbance. Certain spacecraft shapes are considered to be aerostable as a restoring aerodynamic torque appears when the spacecraft aerodynamic equilibrium attitude - where no aerodynamic torque is produced - is lost. The equilibrium attitude is defined with respect to the relative flow which is composed of the spacecraft's inertial velocity, the atmospheric co-rotation and the atmospheric wind. A cone is a simple example of an aerostable shape, with its aerodynamic equilibrium attitude achieved when the cone's axis of symmetry is aligned with the relative flow.
The proposed method is able to measure the atmospheric wind by observing the spacecraft's attitude motion when it is allowed to freely react to the aerodynamic torques caused by the relative flow. Estimates of the other attitude disturbance sources are required to isolate the response cause by the aerodynamic torque and knowledge of the spacecraft's aerodynamic properties and atmospheric density is needed to determine the contribution from the wind magnitude and direction in the observed aerodynamic torque.
Aerostable spacecraft behave as an undamped oscillators with their natural frequency depending on the dynamic pressure and their aerostable properties (i.e. the aerodynamic stiffness). If the spacecraft aerostability is strong enough the attitude motion will remain a bounded oscillation around the aerodynamic equilibrium point and thus no additional control input is required. The natural frequency of the system along with the velocity of the spacecraft, determines the achievable spatial resolution of the wind measurements. As the atmospheric density increases exponentially with decreasing altitudes, improved spatial resolutions at lower altitudes are achieved by using the same aerostable properties. A high spacecraft inertia to aerodynamic stiffness ratio also increases the natural frequency and implies that small spacecraft (with high area per unit of mass) are better suited to be used in the proposed method.
Aerostable spacecraft only provide an aerodynamic torque that is normal to the relative flow direction (i.e. pitch and yaw). So the cross-track wind components have a stronger effect on

the spacecraft's attitude than its in-track wind counterpart. The dynamic pressure is a function of the in-track wind and thus the in-track wind component is also observable although it has a much weaker effect and thus it imposes more stringent requirements.
The method described in this paper can provide global cross-track and in-track wind measurements. The measurements accuracy and spatial resolution with respect to the system parameters (i.e. altitude, aerostable properties and its uncertainty, inertia and uncertainty on the knowledge of atmospheric density) are also analyzed.

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The SPENVIS Next Generation

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ESA's Space Environment Information System (SPENVIS) is an on-line resource for evaluating the space environment and its effects on spacecraft components and astronauts. The SPENVIS system provides access to a large number of space environment models and related analysis tools, allowing the users to combine and chain results between different models. SPENVIS has a long and acclaimed history. Since its first development at the Belgian Institute for Space Aeronomy (BIRA-IASB) in 1996, it has been successfully operational for more than fifteen years. As a result, SPENVIS has established a mature user community from all over the globe that is using the system for various purposes including mission analysis and planning, education and scientific research. Recently, a new system known as SPENVIS Next Generation has been developed under the ESA/GSTP-5 programme. The key objective was to upgrade SPENVIS into a new web-based service-oriented distributed framework supporting plug-in of models related to the hazardous space environment, and including both a user-friendly interface for rapid analysis and a machine-to-machine interface for interoperability with other software tools. The purpose of this talk is to introduce the new SPENVIS system and its capabilities.

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Planetary Orbital Dynamics (PlanODyn) suite for long term propagation in perturbed environment

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Trajectory design and orbit maintenance are a challenging task when multi-body dynamics is involved or in the vicinity of a planet, where the effect of orbit perturbations is relevant. This is the case of many applications in Space Situation Awareness (SSA), for example in the design of disposal trajectories from Medium Earth Orbits, or Highly Elliptical Orbits or Libration Point Orbits, or in the prediction of spacecraft re-entry, or in the modelling of the evolution of high area-to-mass ratio objects. On the other hand, the natural dynamics can be leveraged to reduce the propellant requirements, thus creating new opportunities. Orbit perturbations due to solar radiation pressure, atmospheric drag, third body effects, non-spherical gravity field, etc., play an important role in SSA. The semi-analytical technique based on averaging is an elegant

approach to analyse the effect of orbit perturbations. It separates the constant, short periodic and long-periodic terms of the disturbing function. The short-term effect of perturbations is eliminated by averaging the variational equations, or the corresponding potential, over one orbit revolution of the small body. Indeed, averaging corresponds to filtering the higher frequencies of the motion (periodic over one orbit revolution), which typically have small amplitudes. The resulting system allows a deeper understanding of the dynamics. Moreover, using the average dynamics reduces the computational time for numerical integration as the stiffness of the problem is reduced, while maintaining sufficient accuracy compatible with problem requirements also for long-term integrations. This paper presents the Planetary Orbital Dynamics suite for long term propagation in perturbed environment. PlanODyn implements the orbital dynamics written in orbital elements by using semi- analytical averaging techniques. The perturbed dynamics is propagated in the Earth-centred dynamics by means of the single and double averaged variation of the disturbing potential. The single averaged disturbing potential due to luni-solar perturbations is developed in series of Taylor of the ratio between the orbit semi-major axis and the distance to the third body, following the approach by Kaufmann and Dasenbrock. The effect of other orbit perturbations such as the zonal harmonics up to order 10, the effect of solar radiation pressure and aerodynamics drag are also modelled. The double averaged potential is also implemented by averaging on the variable describing the orbital motion of the perturbing body (i.e. Sun or Moon) around the Earth, but the different inclination of the perturbing bodies is retained. Different application scenarios of PlanODyn will be shown: the behaviour of quasi-frozen solutions appearing for high inclination and high eccentricity orbits (HEO) can be reproduced and the re-entry of geostationary transfer orbits can be studied. In addition, to allow meeting specific mission constraints, stable conditions for quasi-frozen orbits can be selected as graveyard orbits for the end- of-life of HEO missions, such as XMM-Newton. On the opposite side, unstable conditions can be exploited to target an Earth re-entry; this is the case of the end-of-life of INTEGRAL mission, requiring a small delta-v manoeuvre for achieving a natural re-entry assisted by perturbations. Maps of stable and unstable HEOs are built, to be used as preliminary design tool for graveyard or frozen orbit design or natural re-entry trajectories at the end-of-life. Moreover, the application of PlanODyn to design end-of-life disposal from medium Earth orbits through passive solar sailing will be demonstrated.

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Application of the Attitude Analysis of Dynamics and Disturbances Tool in EUMETSAT's study on thruster's allocation and momentum management for meteorological spacecrafts

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The AADDTool - Analysis of Attitude Disturbances and Dynamics Tool – was developed to analyse the disturbances impacts on the dynamics of the spacecraft, start tracker blinding, wheels momentum unloading schemes, and solar power supply, for long term missions in meteorological missions for EUMETSAT. The configuration and setup of the mission is supported by a dedicated graphical interface, for fast configuration and seamless interaction with the simulator.
The paper presents its main features: it combines tiled 3D models for a cylindrical spacecraft (e.g. MSG) or any spacecraft with a central body and solar panels (e.g. MTG and METOP), with accurate models of space environment torques in line with ECSS standard. These were validated with independent tools or available flight/sensor data, and their tile-by-tile analysis of the disturbances allows the user to refine the contributions and take into consideration better approximations for shadowing. The implementation relies on developed libraries implemented in Matlab/Simulink, with a modular architecture to enable a modular design and progressive sophistication of the tool. All modules were validated unitarily using dedicated test setups with reference data (from independent tools, ground and flight data).
Foreseen continuous upgrading activity brought into AADDTool attitude dynamics propagation, spun guidance schemes, guidance programming, and the implementation of an elliptical field-of-view for the Star-Tracker analysis. Additional off-loading schemes were added for performance

assessment of improved laws for wheels momentum management such as maximization of time between off-loading or scheduled off-loading with defined set points of the wheels speeds (or momenta).
For closed loop analysis and study of thrust firing parameterisation, the control loop MetOp was also implemented, including a pulse frequency modulator, dedicated to the thrusters' triggering management where the required torque is achieved by modulating the inhibition duration following a period of constant actuation. Before the study, implementations in the tool of two of the MetOp safe modes were successfully validated in terms of propellant mass and number of firing pulses, against simulated telemetry from the real flight software for Sun Safe Mode, and against data coming from LEOP telemetry (flight data) for Earth Safe mode.
The tool has been used for cases studies in LEO and GEO, to size disturbances and momentum management. The upgraded off-loading schemes have been compared with previous results in terms of offloading frequency and propellant consumption. Some results are presented.
The most recent study was for the MetOp scenario: analyse the impact of re-tuning the default thrusters grouping numbers according to mission phases or scenarios for improvement of propellant consumption, varying the lifecycle, solar activity, and initial conditions. The study used the on-board closed loop for two operating controlled modes: Earth pointing (FAM2) and Sun pointing (PRO). For this study, the MetOp spacecraft was modelled with a tiled 3D-model using a parallelepiped for main body and rotating solar array, used to perform a tile-by-tile contribution of the disturbance torques in LEO. The impact of the closed loop settings were analysed, and the main conclusions are summarized in terms of overall mass consumption, and firing and attitude histories.

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DELTA (Debris Environment Long-Term Analysis)

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In this paper, we present the ESA Debris Environment Long Term Analysis (DELTA) tool, used to analyse the long term propagation and evolution of the future debris environment. DELTA is one of the models that contribute to the IADC studies on long term evolution, which have already been used to derive the mitigation guidelines and have also underlined the need for Active Debris Removal (ADR). DELTA is a three-dimensional, semi-deterministic model, which allows a user to investigate the evolution of the space debris environment and the associated mission collision risks in the low, medium and geosynchronous Earth orbit regions over user defined timespans. DELTA is able to examine the long-term effects of different future traffic profiles and debris mitigation measures, such as passivation and disposal at end-of-life, and also to take into account remediation measures, with the possibility to perform active debris removal in a variety of scenarios with different criteria. DELTA uses an initial space object population as input and forecasts the evolution of all objects larger than a user-defined size. The population is described by representative objects, evolved with a fast analytical orbit propagator which takes into account the main perturbations. The initial population is usually extracted from ESA's MASTER-2009 (Meteoroid and Space Debris Terrestrial Environment Reference) model at a given epoch, and can consider objects down to 1 mm in size. DELTA uses a set of detailed future traffic models for launch, explosion and solid rocket motor firing activity. They are each based on the historical activity of the preceding years. The collision event prediction is done by using a target centred approach, developed to stochastically predict impacts between all objects within the DELTA population. The fragmentation, or break-up, model used is based on the EVOLVE 4.0 (NASA) break-up model. In this paper, we show in detail the architecture of DELTA. Furthermore, we explain its singular way of computing the probability of collision, which is flux-based, different to the majority of the long term evolution tools which use a CUBE method. Finally, some sample results of simulations performed with DELTA are shown in order to display the large range of possible scenarios and applications that such a tool has.

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Evaluation of Iterative Analytical Techniques for Interplanetary Orbiter Missions

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The preliminary design of interplanetary direct transfer trajectories is generally done using the patched conic technique. This design consists of hyperbolic excess velocity vectors at both the ends, say Earth and Mars. For an orbiter mission, a particular inclination and periapsis altitude of the approach hyperbola must be achieved. For a given departure date and a fixed flight duration, there are two hyperbolic transfer trajectory options at each end that matches the excess velocity vector for a specified inclination. Further, each of the two Earth side options can be mapped to each of the Mars side options, thus resulting in four distinct hyperbolic transfer trajectories. Two iterative analytical techniques that generate four distinct design options are introduced in this paper. The iterative analytical techniques are based on the concepts of patched conic and pseudostate methods. The distinct design options are achieved using an analytical tuning strategy. This strategy arrives at suitable hyperbolic orbit characteristics that achieves the excess velocity vector after certain duration, which is a fraction of the total flight duration, in the departure phase. The iterative method based on patched conic technique considers the gravity field of the planet alone within its sphere of influence and that of the Sun alone outside. So, the design obtained by this technique results in large deviations in the target parameters such as the closest approach altitude (CAA) and the related time on numerical propagation under a realistic force model. To improve the achievable accuracies on the target parameters, an iterative method based on pseudostate technique is used and an improved design is generated. A comparison on the deviations in the target parameters that are obtained upon numerical propagation of these designs under similar force model is made. The benefits derived while attempting numerical refinement of these analytical designs are quantified.

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Tube Dynamics and Low Energy Trajectory from the Earth to the Moon in the Coupled Three-Body System with Perturbations

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We develop a low energy trajectory from the Earth to the Moon by extensively using the framework of tube dynamics. We assume that a spacecraft is under the influence of gravity of the Sun, the Earth as well as the Moon and also that the spacecraft and the planets move in the same plane. In this situation, we model the Sun-Earth-Moon-Spacecraft (S/C) 4 body system as a coupled PRC3B system with perturbations, where the Sun-Earth-S/C 3 body system with the Moon perturbation and the Earth-Moon-S/C 3 body system with the Sun perturbation may be coupled at an appropriate patch point. First, we consider various boundary conditions such that the spacecraft departs from the low Earth orbit (LEO) and arrives at the low lunar orbit (LLO). Then we want to find a low energy trajectory connected at a patch point, so that the trajectories from the LEO to the patch point and from the patch point to the LLO can be connected with less maneuver. To do this, we compute the Finite Time Lyapunov Exponent to detect stable and unstable invariant manifolds called "tubes" at a section and we construct the family of departure orbits in the Sun-Earth-spacecraft system with the Moon perturbation, which may be outside of the unstable manifolds associated with the Lyapunov orbit around L2. The family of arrival orbits is also obtained to be inside of the stable manifolds in the Earth-Moon-spacecraft system with the Sun perturbation. With a trajectory correction maneuver, then we obtain a low energy trajectory at the patch point. Finally, we will show that the proposed trajectory may be more

efficient in the sense that the total maneuver is to be less than other trajectories such as the Hohmann trajectory.

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Efficient numerical propagation of planetary close encounters with regularized element methods

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In Solar System dynamics, a close encounter with a major body is the only natural phenomenon capable of modifying the orbital elements of a body on a very short timescale. If not properly taken into account during orbit propagation close encounters may heavily degrade the quality of a solution, or even completely compromise it. When numerically integrating the equations of motion of the body in a heliocentric reference frame, a close encounter will introduce an impulsive perturbation which has to be dealt with either by decreasing the step size close to the perturbing body or by some other device. Also, a close encounter introduces *gravitational scattering*: trajectories which are close before the close encounter may diverge afterwards due to different post-encounter major axes and therefore different orbital periods 1.
The accumulation of numerical error can be reduced by integrating regularized equations of motion, whose characteristics are more advantageous for numerical integrations. When perturbations are weak it is especially convenient to integrate regularized equations which describe the variation of orbital elements, since they will follow an almost-linear behaviour 2. At first, this fact seems to rule out orbital elements for the propagation of close encounters, which by definition introduce a strong perturbation in heliocentric orbits. Here, we circumvent this problem by switching the primary body from the Sun to the planet during the close encounter phase, along the lines of the patched conics method of preliminary orbit design 2. Thus, the propagation is split in three weakly-perturbed legs: heliocentric pre-encounter, planetocentric, and heliocentric post-encounter. A particularly delicate aspect is the definition of the point at which to switch primary bodies during the propagation. Ideally, this has to be chosen according to a criterion which minimizes the final propagation error and the computational cost.
We tested our approach by performing large-scale numerical simulations of close encounters in the planar Sun-Earth CR3BP. Each simulation is parametrized in the conditions at the point of minimum approach distance. The propagation performance is evaluated for encounters taking place in a wide range of asymptotic velocities, and for different kinds of heliocentric orbits. As an additional parameter we choose the geocentric distance at which the switch between the dynamics is executed, as to study the influence of the switch point on the propagation efficiency. We employ different formulations of the Dromo family of element methods [4, 5], and we compare them against the integration of the equations of motion in Cartesian coordinates (Cowell's method) and the Kustaanheimo-Stiefel method 2. The integrator used is an implicit multistep with variable step size and order, which automatically alternates between Adams-Bashforth-Moulton and BDF numerical schemes [6]. For each propagation, the accuracy is estimated with respect to a reference solution computed in quadruple precision using Cowell's method, while the computational effort is measured by the number of calls to the right-hand side of the equations.
Adopting regularized element methods and switching the primary bodies increases propagation efficiency, especially for relatively low minimum approach distances. Figure 1 depicts results for close encounters with a minimum approach distance of 5.03 Earth radii. Even with a sophisticated integration scheme with variable step size and order, regularized element methods guarantee a gain of up to three orders of magnitude in accuracy for about the same computational cost as Cowell's method. Varying the geocentric distance at which the dynamics are switched does not have an effect on the final propagation error, but it does influence the number of function calls. An optimal range of switch distances exists in which the function calls reach a minimum.

Preliminary tests with different integrators for a limited set of initial conditions have shown that the existence and extension of an optimal switch distance range depend on the characteristics of the numerical scheme.
Work which is currently being carried out includes a comprehensive study aimed at the definition of a criterion for switching between heliocentric and planetocentric dynamics, with the objective of maximizing the propagation efficiency with regularized element methods. The simulations will be extended and re-parametrized for the 3D case, and the propagation efficiency will be estimated for case studies modelled on objects of particular significance for Space Situational Awareness activities, such as 99942 Apophis. Open-source software tools for the propagation of close encounters with regularized element methods will be made available through an online repository.

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Coupling High Fidelity Body Modeling with Non-Keplerian Dynamics to Design AIM-MASCOT-2 Landing Trajectories on Didymos Binary Asteroid

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The Asteroid Impact Mission (AIM) is a mission by ESA, planned to be the first to rendezvous with a binary asteroid. AIM mission objectives includes both scientific investigations and technological demonstrations. The mission is part of the Asteroid Impact & Deflection Assessment (AIDA), a joint cooperation between ESA and NASA, devoted to assess the effectiveness in deflecting the heliocentric path of a threatening Near Earth Asteroid (NEA) for planetary defense purpose. The target of the mission is near-Earth binary asteroid 65803 Didymos. The goal of AIDA is to study the effects of a kinetic impact on the surface of the smaller secondary asteroid of Didymos couple, informally called Didymoon. To this purpose, together with AIM, the AIDA mission includes DART (Double Asteroid Redirection Test), the kinetic impactor, designed by NASA. The primary objectives of AIM include the detailed study and characterization of the binary couple. Among these, the internal composition of the smaller asteroid will be determined by means of low frequency radar tomography. Similarly to what done with the CONSERT instrument, on board the ESA's Rosetta mission, the radar will include a lander-orbiter architecture to host both transmitters and receivers. Rosetta mission highlighted the challenges of designing close proximity trajectories and to land a probe on the surface of an extremely irregular body such as comet 67P/Churyumov-Gerasimenko, whose shape and mass distribution were completely unknown and unexpected during the mission design phase. In that case, the Philae lander release was challenged by the highly perturbed dynamical environment in the proximity of the comet and its very low and irregular gravity field. In analogy with the Rosetta mission, AIM will deploy a small and passive probe (MASCOT-2, with clear heritage from MASCOT, Hayabusa 2 mission) that will reach the surface of a largely unknown object after a purely ballistic descent. MASCOT-2 lander do not feature any anchoring mechanism and this makes the release even more challenging since Didymos system's gravity field is expected to be weaker, with an escape velocity from Didymoon's surface of about 4-6 cm/s, being the asteroids estimated to be nearly two orders of magnitude less massive than comet 67P/Churyumov-Gerasimenko. In addition, the presence of two gravitational attractors makes the gravity field in the close proximity of the couple highly unstable and chaotic. The paper proposes an effective strategy for MASCOT-2 release, beneficial for the mission analysis and operations design points of view. The AIM scenraio is presented as a perfect case of study, but the methodology applies to any asteroid/small body scenario. In particular, the landing trajectory and dynamics of MASCOT-2 is studied during close-proximity operations using the highest up-to-date fidelity model of Didymos. The paper presents some updates on the work the authors are currently performing during the phase A/B1 design of AIM, under ESA contract, in consortium with OHB System AG, and Spin.Works. From the orbital mechanics point of view, the binary system is naturally modeled as a three-body system and solutions are studied within the frame of the Restricted Three-Body Problem modeling. Shape-based models are used to model the gravitational contribution of the two asteroids refined models are built by combining them to reproduce the gravity field in the proximity of the binary couple. The purpose of the design strategy is to take advantage of the presence of two gravitational attractors to find effective landing solutions. The increased complexity because of the two gravity sources is here read as a potential opportunity to be exploited through the three-body problem modeling, which opens to a variety of dynamical solutions not available whenever a single attractor is dealt with. Three-body solutions are computed for Didymos binary system and suitable trajectories to land MASCOT-2 on the surface of the secondary are selected. More in detail, the motion close to the Lagrangian points is exploited: stable manifolds associated to Halo and Lyapunov orbits, have been propagated in the high-fidelity dynamical environment and suitable solutions are selected to guarantee soft landing on the secondary asteroid. The dynamics of the lander is propagated from release up to rest on the surface of the asteroid. Results show that the extremely low gravity environment does not guarantee the lander to stay on the surface after touch down, but MASCOT-2 will most likely bounce until reaching a stable position of Didymoon. Successful landing probability is assessed for the case of study and landing dispersion is evaluated. Compared to classical Keplerian solutions, three-body dynamics are shown to be effective to lower the risk of rebounding on the surface of the secondary, and to increase the safety of the overall release maneuver to be performed by AIM.

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Dynamics in the center manifold around equilibrium points in Periodically Perturbed Three-Body Problems

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Please find attached the corresponding abstract.
Best regards,
Bastien Le Bihan.

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Simulation of autonomous landing near a plume source in a tiger stripe canyon on the south pole of Enceladus

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The Institute for Space Technology and Space Applications (ISTA) of Bundeswehr University Munich is investigating mission concepts for the in-situ astrobiological exploration of the icy moons of the outer solar system. A concept studied in the context of the DLR funded Enceladus Explorer project (EnEx) aimed to place a lander near one of the plume sources on the bottom of a "tiger stripe" canyon on the south pole of Saturn's moon Enceladus. Once there, the lander would deploy a melting probe to sample relatively shallow liquid water in the ice under the plume source. The lander would have to achieve a landing accuracy of 50 m and manage to land safely on an extremely challenging terrain interspersed with landing hazards like ice blocks, but also uncertain terrain like soft and unconsolidated snow or hard ice. To achieve this, a landing Guidance, Navigation, and Control (GN&C) system would be necessary to allow for autonomous landing operations. To achieve the required accuracy, terrain relative navigation can use sensors such as optical and thermal cameras, LIDAR, etc. to navigate relative to detected terrain features. To ensure a safe landing the system must be able to assess if the originally planned landing site is safe and if not, to then autonomously command a retargeting to another, safer spot. The guidance and control function must then calculate a viable trajectory and thrust arc to the newly chosen landing site.
To validate that the landing satisfies the accuracy and reliability requirements we are developing a tool in Matlab/Simulink to simulate the operation of the autonomous landing GN&C system. In this paper we present a first version of this tool, used to simulate the final phase of landing operations as described above. It comprises the following parts:
• Terrain simulation block: Generates a simple terrain model based on a given Digital Elevation Model (DEM) file. The topology can be modified using fractal algorithms and arbitrary simple shapes can be added to represent hazards. Terrain texture is also simulated.
• Simultaneous Localization and Mapping (SLAM) block: Simulates Terrain Relative Navigation (TRN)/feature matching, whereby features are extracted from the simulated terrain and a SLAM approach is followed for accurate navigation.
• Hazard Detection and Avoidance (HDA) block: Creates fused hazard maps from the output of camera and LIDAR sensors based on the simulated interaction of these sensors with the terrain. A fuzzy logic approach is used to evaluate landing safety and command a retargeting if necessary.
• Guidance and control block: Implements E-guidance and D'Souza guidance algorithms to generate a thrust arc and trajectory to a new landing spot if commanded by HDA.
Using this tool we will attempt to show the feasibility of an adequately accurate and safe landing near a plume source on the bottom of a tiger stripe canyon on the south pole of Enceladus.

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An Interactive Trajectory Design Environment Leveraging Dynamical Structures in Multi-Body Regimes

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With the increasing complexity of cislunar and interplanetary missions, as well as the introduction of a wide range of scenarios involving small spacecraft, there is significant motivation to design trajectories that require fewer resources and may be sustainable over longer time intervals. Progress toward such goals is achievable by leveraging the natural dynamical structures available within multi-body regimes to guide the selection of a baseline trajectory. As demonstrated by previous missions, such as ARTEMIS, three-body dynamical structures can provide innovative trajectory design options. Such analysis can be particularly beneficial for upcoming mission concepts including exoplanet observatories, the redirection of asteroids, as well as lunar and interplanetary CubeSat missions. Tools that enable active selection of dynamical structures available in a multi-body regime may also supply the guidance necessary for rapid and well-informed trajectory design in chaotic environments.

The accessibility of regions of interest in a multi-body regime is a challenging metric to represent and the efficient selection of specific solutions is nontrivial for dynamically sensitive environments. In higher-fidelity multi-body models, the generation of a large set of solutions demands significant, and often prohibitive, time and computational resources. However, the Circular Restricted Three-Body Problem (CR3BP) is well known and this model offers a reasonable approximation to the dynamical behavior, e.g., in the Earth-Moon and Sun-Earth systems; periodic and quasi-periodic orbits in these dynamical environments also tend to persist in higher-fidelity models. Associated with these ordered motions are natural manifolds, which are useful in assembling low-cost transfers. In fact, researchers in the astrodynamics community continue to investigate a large number of solutions and techniques that may be useful for orbit design and operation within multi-body dynamical environments.
To incorporate knowledge of the dynamical accessibility of specific regions in the Earth-Moon and Sun-Earth systems into the trajectory design process, Purdue University, in partnership with NASA Goddard Space Flight Center, has developed a graphical and interactive design environment that enables rapid and well-informed construction of complex trajectories that leverage natural solutions. This trajectory design tool is comprised of several modules that offer guidance into the leveraging of known dynamical structures for the active selection of trajectory arcs. For instance, to support various mission scenarios, an interactive catalog of periodic and quasi-periodic solutions in the CR3BP supplies the capability for straightforward design trades and orbital selection. Pre-computed libration point orbits are also available in some systems, with the option for on-demand manifold generation. Additional modules offer the capability to identify alternative periodic and quasi-periodic orbits using point-and-click selection on a Poincaré map. Incorporating maps (and/or surfaces of section) into the trajectory design process enables detection of additional orbit options and connections, and provides insight into the dynamical sensitivity. This interactive design tool enables rapid and well-informed construction of complex trajectories in a user-friendly environment that offers intuitive access to dynamical systems theory techniques including Poincaré maps. The end-to-end trajectories are then available for transition to higher-fidelity models, for refinement via the addition of various constraints, for input to other tools such as GMAT.

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High-fidelity small body lander simulations

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Recent missions to the small bodies of our Solar system have established a core understanding of the origins, characteristics, and dynamics of these pristine bodies. Furthermore, the feasibility and benefit of including lander platforms capable of performing in-situ (sub-)surface measurements was demonstrated most notably by Rosetta's Philae lander. The inclusion of such landers on small body mission motivates the need for capability to generate high-fidelity simulations of the landers' trajectories. In this work, we discuss and demonstrate the SAL tool, which provides that capability.
Our tool models small body environments at three distinct scales: at the large scale, the coarse shape and gravity field of the target body are represented using a constant-density polyhedron. The intermediate scale consists of rocks and boulders on its surface, which are recreated following statistics observed on asteroid Itokawa. Finally, at the small scale, interactions between the target body surface and a lander are captured using a contact dynamics model based on the normal, friction, and rolling resistance forces and torques.
The major challenge in simulating this environment results from the high resolution of the applied models. The computational load of detecting collisions between a lander and a small body shape model with hundreds of thousands of facets is reduced through the use of bounding boxes and the subdivision of the target body shape into local worlds. By constructing simplified gravity models from the high-resolution shape model, we reduce the cost of gravity field evaluations in exchange for only a small reduction in the gravity field accuracy. Generating the full distribution of millions

of surface rocks is computationally unfeasible; instead we apply a procedural generation strategy that creates these surface features “on the go,” and only on the active local world when the bounding-box collision module detects an impending collision. This provides a reproducible, low-cost technique for creating rock distributions. Finally, regional variations in the properties of the small body surface are captured by locally varying the coefficients of restitution, friction, and rolling resistance, which govern the energy dissipation during contact interactions.
Using this tool, we may carry out sets of Monte Carlo deployment simulations in which the release conditions, rock field parameters, and surface interaction coefficients are varied. The resulting trajectories allow mission designers to analyze the feasibility of a given deployment strategy, and establish related hardware requirements, such as the on-board battery capacity, which follows from the expected time-to-land. The trajectories’ geographical spread provides information on the illumination, thermal, and scientific characteristics of the expected landing site. Finally, our tool may also be used to simulate surface mobility operations, in which a lander uses mobility devices such as reaction wheels to generate impulses, which allow it to travel on and over the small body surface in a series of “hops,” enabling a single lander to obtain scientific measurements at multiple sites.

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NEO Threat Mitigation Software Tools

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Nowadays, there are a number of institutions worldwide that contribute to the discovery, tracking, identification, cataloguing and risk characterisation of asteroids in general, and NEOs in particular. However, there is no currently an integrated set of tools that cover, in a complete manner, the assessment of the impact risk mitigation actions that can be taken to prevent the impact of a NEO on Earth and to allow helping the dimensioning of space missions to address such problem. In that context the following set of utilities have been developed within the European Commission funded NEOShield project to allow covering the abovementioned activities: - NEO Impact Risk Assessment Tool (NIRAT) - NEO Deflection Evaluation Tool (NEODET) - Risk Mitigation Strategies Evaluation Tool (RIMISET).
NIRAT, the first tool, allows evaluating, for possible impactors, the projection of the b-plane dispersion at the dates of possible impact and also the presence of keyholes that would enable future collision opportunities. This tool allows characterizing the impact probability for the different opportunities and, together with the knowledge of the asteroid features, the evaluation of the risk. This tool resembles current performances achieved by NEODyS and Sentry, but does not intend to represent the same level of accuracy in the obtained results. The services provided by this tool are required by the next other tools.
The second tool, NEODET, allows assessing the required optimal change in asteroid velocity (modulus and direction) at any given instant prior to the possible impact epoch that would allow shifting the dispersion ellipse out of the contact with the Earth. This would represent the effect of impulsive mitigation options (one or several impacts). It also allows evaluating the accumulated effect that slow-push techniques (e.g. gravity tractor) would impose on the asteroid orbit to achieve optimal deflection by those other means.
Finally, the RIMISET tool allows evaluating how some of the most relevant impulsive and slow-push mitigation techniques would meet the required changes in asteroid state to obtain the searched for deflection and the requirements that this could impose on the design of the mitigation mission. Following mitigation methods are included: explosive, kinetic impact, gravity tractor and ion beam shepherd. Each technological solution is simulated to allow ascertaining the efficiency in achieving the deflection goal by any of the proposed means (impact, explosive, gravity tractor and possible combinations of those). Ultimately, it shall serve to dimension the required mitigation space systems and solutions.

Validation cases have been executed over the now no-threat cases of asteroids 2011 AG5 and 2007 VK184.

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Monte Carlo Simulation of a Triple Flyby Capture at Jupiter Using Paramat

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In 2014, Thinking Systems began work on a parallel processing tool that incorporates the numerical engine from the General Mission Analysis Tool (GMAT) into a system, Paramat, designed to use the processing capabilities of modern, multi-core computer platforms. This paper opens with a brief description of recent changes to Paramat. Paramat is then used to model an orbital capture at Jupiter that uses gravity assists at Callisto, Io, and Ganymede to reduce the orbital insertion costs for the capture. That mission segment is presented as a baseline trajectory for Monte Carlo analysis of the costs of the insertion sequence. Perturbations are applied to the nominal capture trajectory and to parameters related to the course correction maneuvers in order evaluate the maneuver contingency costs for the capture. Analysis of these data provide insight into the total delta-V costs and margins for the capture phase of the mission, along with an estimate of the orbit determination requirements for each phase of the capture trajectory.

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Navigation Tools at ESOC Mission Analysis Section

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Software able to study and simulate the guidance and navigation process is essential when dealing with complex interplanetary missions. The outputs required from such software are multiple, ranging from the evaluation of stochastic ΔV required to keep the dispersion errors with respect to the nominal trajectory small, to the identification of critical phases of the missions for the guidance and navigation process, to the assessment of the optimal ground station network selection and many others. In the ESOC Mission Analysis section the two most complete tools available in this context are IntNav (Interplanetary Navigation software tool), developed by GMV and LoTNav (Low Thrust Navigation software tool), developed by DEIMOS Space. Both tools are composed of several modules and they share a common structure; the core modules of both software are: measurements generation module, parameters estimation module and guidance module. The first module computes the measurements selected for the orbit determination, given the trajectory of the spacecraft and the measurement schedule (availability of observations and ground stations are always taken into account). The parameters estimation module is used to compute partial derivatives of the estimated states with respect to the selected parameters and also to conduct the covariance analysis for knowledge and dispersion. The guidance module finally is used to determine the stochastic manoeuvres to be executed in the guidance process. Additional modules are also present in both tools (or were added to the core software afterwards), to deal with other functions or analyses to be performed in the navigation context: for example modules for the trajectory computation and sectioning are available in both tools, a reconstruction module is available in IntNav to simulate the trajectory reconstruction of particularly interesting phases in the trajectory (e.g. flybys), and many others are available. A general overview of this structure will be presented, followed by examples obtained with both tools. JUICE moons tour navigation results obtained with IntNav and Bepi-Colombo interplanetary navigation results obtained with LoTNav will be presented. The major assumptions will be explained and the encountered issues will be highlighted: flybys with high uncertainties and large navigation manoeuvres, parameters

selection choices taken after the results evaluation, the non-trivial optimization process for the manoeuvres targeting and other aspects. Final results for both cases will be then detailed.

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SNAPPShot: Suite for the Numerical Analysis of Planetary Protection

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When interplanetary missions depart from the Earth, the rocket bodies used for their launch may be inserted into a resonant orbit with the Earth or into trajectories that may cross other orbits. A suite of numerical tools has been developed as part of a European Space Agency contract to study the compliance of the launcher with planetary protection requirements.
For each mission, the launcher injection dispersion, the uncertainty in the spacecraft physical parameters, or potential spacecraft failures are treated with a Monte Carlo (MC) analysis. The uncertainty in the velocity vector components with respect to the nominal manoeuvre is defined and the distribution of velocity is sampled to obtain a discrete number of initial conditions. If required, the dispersion of other parameters (such as the area-to-mass ratio) can be included. Each initial condition is propagated through a high accuracy propagator, which describes the evolution of the launcher trajectory in Cartesian coordinates with respect to the solar system barycentre. Once each trajectory is computed, it is analysed to detect if it enters a planet's sphere of influence. If so, the b-plane representation is used to characterise the close encounter.
The suite includes several Runge-Kutta propagators (both with adaptive step size strategy and regularised step size), providing also the possibility of dense output and the inclusion of user-provided events in the propagation (e.g. the propagation can be stopped if an impact is registered). Different options in the ephemeris routines (including the NASA SPICE toolkit) are available, which were validated against the data on the JPL Horizon system. In addition, SNAPPShot provides a tool for the analysis of the b-plane to distinguish between conditions of impact, gravitational focussing, and resonances. Fly-bys of different bodies and subsequent close encounters can be handled. The tool can identify the most critical close encounter, which will be used for the statistics generated from the MC analysis. The number of required MC runs to validate the requirements with a given confidence level is automatically estimated Wilson confidence intervals. If impacts are detected, the number of MC runs is automatically incremented.
All codes in the suite have been written in Fortran 90, exploiting the advantages of modern Fortran, such as array operations, dynamic memory allocation, modules and procedure pointers. The possibility of using parallel computations in the MC analysis will be discussed. As an example, an analysis for planetary protection compliance verification for the launcher of the BepiColombo mission requires four hours of computational time on a PC to run a MC analysis with more than 50000 runs.

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from low level toolbox to orbit determination: handling users requests in Orekit

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Orekit is a core space flight dynamics library published as free software under the terms of the permissive Apache Software License V2. Since its inception in 2002, Orekit was designed as a low level layer providing the foundation objects for operational applications. From the very beginning, the foundation objects included time, frames, orbits, but also attitude, maneuvers and a rich framework for state propagation with continuous output and on the fly detection of discrete events. As new versions were published, this set of features has been extended, with new propagators (the semi-analytical DSST being a major example), new predefined events (16 as of end 2015 and counting), new frames and various improvements.
As an open-source project, Orekit is essentially driven by its users requests and contributions. Looking back at the project evolution, one notices that after the initial stabilization phase during which the core features were completed, users requests led to introduce more and more intermediate features. These features where more mission or operation-oriented, showing us the library was used in various contexts for real problems solving. Many features added in the last two or three versions were really not envisaged at project start. These features clearly show the benefits we get from an open community. Some users have a problem to solve that first appears to be really mission-specific, but often as they notify the project about it, it appears more general than expected and can benefit other users after some reformulation. One typical example is the ellipsoid tessellation. This strange feature was needed for one project in early 2015, but just one month after its design, a second project raised a similar need and a few weeks later an independent user opened a feature request on the same topic.
Some other features have been on our plans for a few years without being implemented, both because of lacking resources to do the job and because they were considered at the boundary of Orekit scope. This was the status of orbit determination. Here again, as more and more users were demanding for it, we finally embarked on it and added it.
This presentation focus on how an open-source project can interact with its users while still maintaining a general orientation, using some examples from the last few releases of the library, up to the latest addition of orbit determination.

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WORHP Multi-Core Interface, Parallelisation Approaches for an NLP Solver

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The goal of this talk is to present current research activities aiming at improved efficiency and stability within the ESA NLP-Solver WORHP. It is designed to solve high dimensional sparse non-linear optimisation problems.
The underlying SQP method is inherently sequential, therefore parallelism cannot be exploited straightforwardly.
In order to obtain the best solver performance the parameter configuration should be adapted accordingly for every problem itself. An approach running several solver instances using different parameter sets in parallel has been developed and proven highly beneficial on a given set of problems. The First-Across-The-Line approach stops all instances when the first local optimum has been found, thus improving the solver's speed and stability as well. Furthermore, the approach allows the user to experiment with specialised algorithms within the optimisation as the threads using basic parameter settings serve as safeguards guaranteeing the solver to converge as usual.
In order to improve the solver's efficiency for one special problem, the new operational mode can be used to automatically attune the solver's parameters accordingly. Again the solver is started with several instances at once, but this time the Best-Of-All mode is used in order to obtain the best local optimum and the corresponding parameter settings. Additionally, the mode enables users to perform parameter sweeps to further improve the solver's configuration.
The results presented show the improvement of the solver's performance on the state-of-the-art CUTEst test set that has been solved faster and with more optimal solutions found compared

to the traditional single-core approach. The parameter attunement is applied to specific single problems from the test set as well.

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Non-Keplerian Trajectory Planning via Heuristic-Guided Objective Reachability Analysis

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In many space exploration scenarios of great interest, such as close-range study of asteroids and comets, spacecraft motion cannot be effectively approximated using Kepler's laws. Furthermore, special dynamical structures such as periodic orbits are not inherently associated with specific science requirements and serve only as a limited framework for facilitating operations. To broaden the mission design domain for pursuing abstract objectives in non-Keplerian systems, we instead formulate a reachability analysis tool that maps a domain of available single-impulse maneuvers onto a set of high-level outcomes. As this process can only be conducted with numerical sampling, heuristics are used to guide iterative refinement of map features or the search for a performance metric's global maximum. The reachability data product can be visualized to aid preliminary mission design or efficiently computed onboard the spacecraft to enable opportunistic and robust online planning. Both modes of use will be demonstrated for planning trajectories for close-range imaging of potential lander deployment sites on the highly irregularly shaped comet 67/P.

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Analytical Approximation for the Multiple Revolution Lambert's Problem

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An approximate analytical solution of the multiple revolution Lambert's problem is presented. The solution is obtained based on a variation of parameters approach and offers remarkable accuracy near the minimum energy condition. Consequently, the method is useful for rapidly obtaining low delta-V solutions for interplanetary trajectory optimization. In addition, the method can be employed to provide a first guess solution for enhancing the convergence speed of an accurate numerical Lambert solver.

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MOSQP: an SQP Type Method for Constrained Multiobjective Optimization

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We propose an SQP type method for constrained nonlinear multiobjective optimization. The proposed algorithm maintains a list of nondominated points that is improved both for spread along

the Pareto front and optimality by solving single-objective constrained optimization problems. We provide numerical results for some trajectory optimisation problems as well as for a test set of unconstrained and constrained multiobjective optimization problems. The numerical results confirm the superiority of the proposed algorithm against a state-of-the-art multiobjective solver, either in the quality of the approximated Pareto front or in the computational effort necessary to compute the approximation. Moreover, we discuss convergence to local optimal Pareto points under appropriate differentiability assumptions

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MP2OC: Multi Phase Multi Purpose Optimal Control Toolbox

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The paper will present a Multi Phase Multi Purpose Optimal Control (MP2OC) code developed at Strathclyde University in the framework of the UKSA National Space Technology Programe (NSTP-2). The grant received for the development of the tool was part of a bigger project aiming at building an integrated design platform for quickly assessing design parameters based on the simulated performance and operations for future space access vehicles. The trajectory module here presented optimises the entire mission, from ascent to insertion into final orbit and from de-orbiting to final landing. The future development of the tool will also take into account model and operational uncertainties and evaluate the robustness of the descent performance against them.
The design of spaceplane mission delivering a payload to a certain orbit and reentering is in itself a multiphase problem. Moreover the atmospheric trajectories, can be further divided in multiple segments. Among segments, the elements defining the problem can differ though disciplinary models (e.g., propulsion modes for a hybrid engine, or in a multi-stage propulsion system), problem objectives and constraints, and level of fidelity needed within the models. To allow such a flexibility in the design, the simulation needs to been structured in multiple phases, with interchangeable disciplinary models. Hence the tool has been designed to allow for each phase, to define the set of models used and all the variables involved in the parameterisation of the controls and propagation of the trajectory. The continuity between phases is guaranteed at convergence of the optimization process by the optimiser and the formulation through constraints functions. Different optimisation algorithms have been included and tested to tackle a variety of problem definitions – from single to multi-objective, deterministic and stochastic, from constrained to unconstrained – and to offer both global exploration capabilities, and local refinement. Fixed and adaptive step size integration techniques and interpolation methods are also included. The platform has been entirely developed in Matlab and it has been interfaced with C disciplinary code for testing purposes. The tool has been designed in such a way that can be easily extended in the future to integrate different optimization/integration techniques.
Finally the possibility of approximating disciplinary models through Kriging metamodeling technique has also been integrated in the platform. A preprocessing is performed before optimisation to evaluate the model in a number of sampling points and build the surrogate The use of surrogate is particularly useful when expensive disciplinary models are involved in the design. The possibility of developing an automatic procedure for multifidelity optimisation is part of the future development for the tool.
The toolbox capabilities and architecture will be presented together with a real application where the trajectory control is optimised, based on different mission objectives and constraints, for the ascent and descent mission segments of a conceptual single stage to orbit vehicle, to a circular low Earth orbits from fixed take-off and landing site.

Interplanetary Flight and Non-Earth Orbits (II) / 155

Trajectory Design Tools for Libration and Cis-Lunar Environments

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The Sun-Earth libration and Cis-Lunar environments are challenging regimes for trajectory designers with complex multi-body dynamics, perturbation modeling, and integration of propulsion influences. Beginning with libration orbits and research on dynamical systems (aka manifolds), several tools with application to libration orbits and Cis-lunar regions have been developed in cooperation between NASA's Goddard Space Flight Center and Purdue University. One of these innovative tools, the Adaptive Trajectory Design (ATD) tool is being used in conjunction with commercial software to design both a multi-body trajectory for the upcoming Lunar IceCube (L-IC) Cubesat mission and the Wide-Field Infrared Survey Telescope (WFIRST) Sun-Earth L2 mission transfer. As a payload deployed by the Exploration Mission-1 (EM-1) on the maiden flight of NASA's Space Launch System (SLS), L-IC will use a lunar gravity assisted, multi-body transfer trajectory with an innovative RF Ion engine to achieve lunar capture and delivery to the science orbit. WFIRST trajectory deign is based on an optimal direct transfer trajectory to an L2 orbit. In the paper, ATD utilities that permit the designer to categorize orbits by energy and amplitudes among other numerous design variables, Circular Restricted Three Body methods, and manifold generation are described along with the transfer trajectory design process for both missions. Based on the constrained L-IC EM-1 architecture and deployment, an assessment using ATD and dynamical system research tools has uncovered Euclidian regions of Cis-lunar space which permit a transition onto stable/unstable manifolds that encounter the Moon at the prerequisite arrival conditions, resulting in an innovative process. Using ATD's powerful Poincare mapping tools and libration orbit generation via energy or orbit amplitudes, feasible WFIRST science orbits are generated that feed into the selection of optimal transfer manifolds from the low Earth orbit injection condition. These ATD utilities for both missions permit the interweave mapping of manifolds and conics to complete any Cis-lunar or Libration orbit design. ATD's innovative applications will be fully defined and the basic operations and its interface to GSFC's General Mission Analysis Tool (GMAT) for high fidelity modeling are presented.

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Asteroid Rendezvous Uncertainty Propagation

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Most methods of propagating orbit uncertainty assume posteriori Gaussian distributions, require an intrusive implementation or suffer from the curse of dimensionality associated with high-dimensonal random inputs. Although Monte Carlo techniques avoid these draw backs, the approach has a slow convergence rate. This paper considers the application of separated representations for orbit uncertainty propagation and discusses the theory behind their generation. The computation cost of a separated representation is largely linear with respect to dimension, thereby improving tractability when compared to methods that suffer from the curse of dimensionality. Generation of a separated representation requires the propagation of a small number of samples and yields an approximate solution, or surrogate, to a given stochastic differential equation describing the propagated orbit. This surrogate provides information on the moments and spatial density of possible solutions, as well as the sensitivity of the quantities of interest with respect to random inputs. This paper presents the case of spacecraft targeting an asteroid for a rendezvous, for

which the initial conditions of each and the components of the interceptor maneuver are uncertain. Separated representations is used to estimate the probability of a successful rendezvous and analyze the resulting probability distribution functions.

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Some validation checks of "TriaXOrbitaL" tool : Earth-Moon L2 orbit, Sun-Moon perturbations.

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The modelling becomes year after year a priority for each agency, each company in order to better foresee the features of space devices and spacecrafts. The paper presents a flight dynamic tool so-called "TriaXOrbitaL", freely distributed as promotional release, that has been used since 1989 with continuous improvements for constellation design, North-South station keeping manoeuvre, orbit transfers GTO or better Super-GTO to GEO, interplanetary flight and travel to Moon. The tool has been fully focused on electric propulsion long thrust arcs but can manage as well of course chemical propulsion shorter arcs. The tool is accessible and has been used by engineers or students. The main advantage of using such tool is for sure a simplification of the preliminary studies because the tool has been developed for being accessible for every engineer willing to improve his knowledge in the orbital manoeuvres field. But one of the drawbacks when using such tools is to be able to state about the validity and accuracy of the results provided by the tool. Since the beginning of its development, a great care has been taken with respect to the fundamental checks of the tool. Hence the paper exhibits some of the tests performed in order to reproduce the "every body know" specific behaviors of the flight dynamic: for example the evolution of the GEO due to real Sun, Moon and earth potential perturbations (J2) with inclination up to 15° for 54 years; the stability of the Earth-Moon Lagrangian point L2, and several other checks. The description of the tests performed may constitute a good starting point for using the tool and for the knowledge of its features.

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Extended Tisserand graph and multiple lunar swing-by design with Sun perturbation

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The use of multiple lunar swing-bys to pump up the hyperbolic escape velocity of interplanetary trajectories has been proposed in literature and repeatedly put into practice in real missions (*Kawaguchi et. al., 1995, Dunham et. al., 2007*). JAXA's technology demonstrator mission DESTINY (*Kawakatsu et. al., 2013*), with its new main mission objective to fly by asteroid Phaethon, is planning to make use of a similar strategy to obtain the required escape velocity after a low-thrust spiralling phase from its launcher injection orbit.
This paper presents a systematic approach to design multiple lunar swing-by sequences that can be applied for this purpose. The Sun third-body perturbation plays an important role essentially providing free Δv between lunar swing-bys. As a first step, an extension of the classical Tisserand graph in perigee-apogee radius is presented, in which the potential gains by solar perturbation between flybys can be estimated. Secondly, following an approach proposed by *Lantoine & McElrath (2014)*, a database of Moon-to-Moon transfers is generated with a continuation method. A simplified planar circular restricted three-body problem is assumed. The families of transfers are stored parametrised as a function of initial Sun-Earth-Moon angle, lunar hyperbolic escape velocity modulus, and direction. In addition to the families calculated by *Lantoine & McElrath*,

new families with multiple revolutions and families with energies close to libration point orbits are found and generated. The database can be accessed and transfers retrieved to quickly generate sequences in a similar fashion to a multiple swing-by classic Lambert problem solver, but including the effect of the Sun third body perturbation.
Two particular practical examples are presented: a sequence to be used for the DESTINY mission to escape the Earth-Moon system and initiate the transfer to asteroid Phaethon, and a solution to obtain a transfer to the Earth-Moon L_2 point.

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Solar System geometry tools with SPICE for ESA's planetary missions

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ESA has a number of science missions under development and in operation that are dedicated to the study of our Solar System (i.e. MEX, Rosetta, ExoMars, BepiColombo, Solar Orbiter and JUICE). The Science Operations Centres for these missions, located at the European Space Astronomy Centre (ESAC) in Spain, are responsible for all science operations planning, data processing and archiving tasks, being the essential interface between the science instruments and the spacecraft, and with the scientific community. From the concept study phase to the day-to-day science operations, these missions produce and use auxiliary data (spacecraft orbital state information, attitude, event information and relevant spacecraft housekeeping data) to assist science planning, data processing, analysis and archiving.
SPICE is an information system that uses auxiliary data to provide Solar System geometry information to scientists and engineers for planetary missions in order to plan and analize scientific observations from space-born instruments. SPICE was originally developed and maintained by the Navigation and Ancillary Information Facility (NAIF) team of the Jet Propulsion Laboratory (NASA).
This article outlines the different set of tools that are dedicated to geometry handling, visualisation and analysis software using SPICE from mission concept development though the entire analysis of mission data for ESA planetary missions.

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ESA's Asteroid Impact Mission: Mission Analysis and Payload Operations state of the art

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The Asteroid Impact Mission (AIM) is an ESA mission whose goal is the exploration and study of binary asteroid 65803 Didymos, which is expected to transit close to the Earth (less than 0.1 AU) in late 2022. AIM is planned to be the first spacecraft to rendezvous with a binary asteroid: its mission objectives include the highly relevant scientific return of the exploration as well as innovative technological demonstrations. In addition, AIM is part of a joint collaboration with NASA in the AIDA (Asteroid Impact & Deflection Assessment) mission. The primary goal of AIDA is to assess the feasibility of deflecting the heliocentric path of a Near Earth Asteroid (NEA) binary system, by impacting on the surface of the smaller (or secondary) asteroid of the couple. To this aim, AIDA includes the kinetic impactor, DART (Double Asteroid Redirection Test) by NASA and the observer, AIM (Asteroid Impact Mission) by ESA. The work presented in this paper has been performed by the authors under ESA contract within the phase A/B1 design of AIM mission. The paper presents some updates on the ongoing design of the mission. Each phase of the operative life of AIM spacecraft is detailed with information and results on the solutions adopted for Mission Analysis design and on the strategies to suitably operate payloads. The selected interplanetary trajectory is presented, including the available launch window to reach Didymos on time. Suitable transfer solutions are selected based on Δv constraints imposed by the launcher and further requirements imposed by spacecraft design. More in detail, AIM is planned to be launched in late 2020 and to arrive at Didymos system in middle 2022. As the spacecraft approaches the asteroid system, it will go through far- and close- approaching maneuvers. The far-approaching maneuver is presented in detail: the final Δv to stop AIM at Didymos is split into five smaller maneuvers, performed at one week distance between each other, to decrease the overall maneuver cost and to allow for precise tracking and rendezvous with the binary system. Close proximity operations at the asteroid are then described. During this phase, AIM mission analysis is driven mainly by observational requirements coming from scientific payload on board, to study the asteroid system before and after DART impact (expected for late 2022), such to accomplish mission objectives. Observation stations are selected for AIM spacecraft to study Didymos by operating scientific payloads. Close proximity operations include the release of a lander on the surface of the smaller asteroid of the couple (secondary) and the release of a network of cubesat opportunity payloads (COPINS). The deployment strategies are described from the operational and maneuvering point of view. In addition, payload operations include technological demonstration of deep space laser communications, low frequency radar tomography of the smaller asteroid of the couple (called Didymoon) and high frequency radar subsurface investigation. The Earth-Spacecraft-Sun-Asteroid geometry is presented into detail during all mission phases. The paper includes the analysis of coverage and illumination conditions during all phases of the missions, to provide inputs to the planning of scientific payload operations and ground segment operations. Both AIM ground and asteroid coverage is analyzed. Constraints imposed by natural illumination of the asteroids are highlighted, to identify poles visibility and to assess visible latitude bands, during the mission time line and payload operations at asteroid. The results and analyses presented here are part of the phase A design of the AIM spacecraft. The project is currently ongoing and the mission analysis will be further iterated and refined through the design phase.

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Optimal real-time landing using deep networks

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Optimal trajectories for spacecraft guidance, be it during orbital transfers or landing sequences are often pre-computed on ground and used as nominal desired solutions later tracked by a secondary control system. Linearization of the dynamics around such nominal profiles allows to cancel the error during the actual navigation phase when the trajectory is executed.
In this study, we assess the possibility of having the optimal guidance profile be represented, instead, by a deep artificial neural network trained, using supervised learning, to represent the optimal control structure. We show how the deep network is able to learn the structure of the optimal state-feedback outside of the training data and with great precision. We apply our method to different interesting optimal control problems, including the inverted pendulum swing-up and stabilization problem, a quadrocopter time and power optimal pin-point landing control problem and a time and mass optimal spacecraft landing problem. In all cases the deep network is able to safely learn the optimal state-feedback also outside of the training data making it a viable candidate for the implementation of a reactive real-time optimal control architecture.

We perform our study making use of the open source projects Space-AMPL, to compute the supervision signal, and Theano to train the deep network.

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Massively parallel optimization of low-thrust trajectories on GPUs

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The optimization of low-thrust trajectories is a difficult task. While techniques such as Sims-Flanagan transcription gives good results for short transfer arcs with at most a few revolutions, solving the low-thrust problem for orbits with large numbers of revolutions is much more difficult. We developed a massively parallel genetic optimization algorithm to obtain low-thrust solutions for targeting of a celestial body, such as the Moon or a planet. The solution is not limited to simple arcs or few rotations, but is capable of solving the problem also in the case of many revolutions where other classical methods fail. While we perform the propagation in a two-body model, in principle the propagation can also be performed in more complete models such as the circular restricted three body problem.
The optimization algorithm chosen is a genetic algorithm with large population size. Due to its massively parallel nature, this type of problem is a natural fit for implementation on a GPU. Modern GPUs are capable of running thousands of computation threads in parallel, allowing for very efficient evaluation of the fitness function over a large population. In particular, we optimize the shape of the control function as well as the departure time. Propagation of the spacecraft is then performed in massively parallel fashion on the GPU before the results of the fitness function are read back into CPU memory for preparation of the next iteration of the algorithm.
We demonstrate with various examples how this algorithm can provide good initial guesses for a following local optimization to compute very accurate low-thrust orbits.

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Development, validation and test of optical based algorithms for autonomous planetary landing

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In recent years, a renewed interest in space exploration has induced investing a growing amount of human and financial resources to provide next generation spacecraft with enhanced autonomous navigation and landing capabilities. Complex missions in which close approach to and landing on uncooperative objects play a major role are being developed by numerous space agencies: in particular, ESA is working together with ROSCOSMOS on a cooperative programme for Mars, Phobos and Moon exploration: as our satellite is concerned Luna-Resurs Lander (Luna-27) mission planned for 2020 and the Luna 29, the Lunar Sample Return mission to follow are involved, strongly focused on the South Pole landing to collect icy volatiles located in a very precise region of the huge Aitken crater. Part of the European contribution for the Luna-27 mission is the PILOT (Precise and Intelligent Landing using Onboard Technologies) subsystem, for enhancing autonomous landing capabilities in terms of high precision landing and hazard detection and avoidance functionalities. An analogous collaboration has been established for the

ExoMars programme, which include a lander delivery first, followed by a rover release on Mars, to be launched in 2016 and 2018 respectively and for the exploration of the small Mars moon with the Phootprint Mission, a Phobos Sample Return mission planned for the Twenties.
Since 2006, technologies for autonomous landing are studied by NASA in the frame of the Autonomous Landing Hazard Avoidance Technology (ALHAT) program. ALHAT technologies, tested at the end of 2014 in free flight on the Morpheus lander demonstrator, are going to be integrated in future exploration missions. Recently CNSA has performed its first lunar landing with a lander/rover system with the Chang'e 3 mission (with missions 4 and 5 already scheduled), while ISRO is planning to put a lander carrying a rover on the lunar surface by the early 2018 in the Chandrayaan 2 mission. Among the various technologies under study, vision-based systems represent one of the most promising tools to provide the required level of accuracy, unattainable by classical technologies. In this paper the research carried out at the Department of Aerospace Science and Technology (DAER) of Politecnico di Milano about algorithms and tools dedicated to autonomous navigation and hazard detection is presented.
A hazard detection algorithm, based on artificial neural networks, has been developed and extensively tested. A single grayscale image of the landing area is filtered to extract essential information regarding shadows, surface roughness and slopes, and then it is analyzed by a cascade neural network, which provides a hazard map. Hence, hazard information is exploited by a subroutine that computes the most suitable landing site, taking into account safety requirements and trajectory constraints. The achieved computational efficiency allows the system to operate real time.
Also a vision-based relative navigation tool, relying on features tracking between subsequent frames, is currently under development. Camera information is fused together with measures coming from classical sensors, like Inertial Measurement Units and radar/laser altimeters. Collected data are filtered, taking into account the lander dynamics to obtain a convergent estimate of the system status. Collected information are also exploited to build a dynamical semi-dense map of the landing area, exploitable by hazard detection to estimate slopes and to locate the selected target.
To further increase the TRL of the aforementioned algorithms, an experimental facility is under setup at DAER premises. Since the scarce availability of complete real landing imagery datasets, vision-based algorithms development relies widely on synthetic images. To validate such approach, experiments are necessary. Moreover, the whole navigation system performance can be assessed only connecting the composing parts together, to verify mutual influences. The experimental facility under development represents a Hardware-in-the-Loop environment, and it is composed by a 3D mock-up simulating the planetary surface; a 7 DoF robotic arm to carry the sensors suite and simulate the spacecraft dynamics; an illumination system consisting of a LED array and a dimming subsystem to provide a realistic and controllable light. Hence the aim is to simulate the landing maneuver with the robotic arm in a scaled environment with realistic illumination conditions, over a reproduced planetary surface. The system is designed to verify either hardware and software breadboards up to TRL 4, with possible further enhancements to qualify flight models to TRL 5. The development status of the vision-based tools is presented. The design and the first activities for functional verification of facility components are shown, such as first tests of the hazard detector in a lunar environment.

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Understanding concepts of Optimization and Optimal Control with WORHP Lab

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The ESA-NLP solver WORHP is already used in several academic and industrial projects in a wide range of applications, as aerospace, automotive or logistics. Currently over 500 users worldwide code their problem formulations using the the standard interfaces to Fortran, C/C++ and MATLAB.
To simplify the formulation of optimisation problems for demonstration and educational purposes WORHP Lab is developed as a graphical user interface (GUI). With a growing set of applied

examples and visualisation techniques it shows the capabilities of the underlying solver WORHP and opens access to more involved concepts like parametric sensitivity analysis using WORHP Zen.
Furthermore, WORHP Lab provides the possibility to solve optimal control problems using our transcription method TransWORHP. Different approaches like full discretisation with grid refinement or multiple shooting are compared easily within this tool. Additionally, optimal control problems can be solved on reduced time horizons to illustrate concepts of nonlinear model predictive control.
WORHP Lab was already employed successfully in several industrial workshops as well as for educational purposes with pupils and students. In this talk we illustrate its features with aerospatial examples of optimisation and optimal control problems.

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FALCON.m – The free and fast optimal control tool for MATLAB

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No matter if ascent trajectories are to be determined, satellites orbits need to be changed, docking maneuvers must be performed or space debris needs to be collected – in all of those applications optimal trajectories are sought. These examples may be stated as optimal control problems, containing a dynamical model which describes the system behavior and a set of equality and inequality constraints. The optimization is subject to a cost function, which may be minimal energy, minimal time, or any other desired cost function. Since the structure of the optimal control problem remains the same, the constraints, the cost function, and the dynamic model can be changed easily.
In the past, these problems have often been solved using simplifying assumptions and indirect optimal control methods. Today, with the availability of more powerful computers and advancements in numerical optimization theory, direct optimal control has become increasingly important.
The Institute of Flight System Dynamics at the Technische Universität München has developed a new optimal control tool for MATLAB called FALCON.m. The tool is implemented in MATLAB which allows the user to define all required equations within the popular environment. Additionally, exceptional performance is achieved by using automatic analytic derivatives and automatic MEX compilation with multi-threading for the heavy duty parts of the calculations.
FALCON.m implements a direct collocation method for discretizing the optimal control problem and allows the use of any collocation scheme – such as Trapezoidal, Hermite Simpson, or classical explicit Runge Kutta. The resulting numeric parameter optimization problems are solved using state of the art solvers such as IPOPT, SNOPT or WORHP.
The aforementioned solvers use gradient based algorithms, requiring the gradient of the full discretized problem. Its calculation is crucial for the overall performance, in terms of calculation time and convergence robustness. FALCON.m calculates the sparsity of the gradient and utilizes the Symbolic Math Toolbox and MATLAB Coder Toolbox to efficiently and automatically calculate the analytic gradients for dynamic models, constraints, and cost functions. The subsystem derivative approach used by FALCON.m enables a highly performant source code transformation even for large and complex models.
User friendliness is a major focus of FALCON.m. As the tool is delivered as a library of MATLAB classes, a clear information flow allows for a distinct problem formulation. Additionally, checks for common mistakes in the problem formulation are performed, mitigating errors that may be hard to find by hand.
FALCON.m is a state of the art optimal control tool that is available for free. It has been designed for real world applications and an easy access to the world of optimal control. The power of

FALCON.m as well as the variety of its features will be demonstrated in this paper using the example of a space application problem.

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On Ultimately the Most Highly Inclined, the Most Concise Solar Polar Trajectory with Practically the Shortest Period

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This paper presents the extended orbital synthesis results from the author's work in 2009 to achieve ballistic and short period out-of ecliptic trajectories which possess ultimately the most highly and most concise solar polar properties. Those are realized through almost ballistic flight instead of using electric propulsion or solar sail acceleration. The strategy developed utilizes a Jovian gravity assist first, followed by very high speed synchronized multiple polar gravity assists by Earth or Venus. So far, the use of very high speed gravity assist has been conceived not practically useful to control the trajectory energy. However, this paper presents those still effectively contribute to amending the trajectories periods, in other words, to diminishing the size of them, and lead to acquiring small sized out-of-ecliptic ballistic trajectories. The process simply converts orbital energy associated with highly eccentric ellipses to inclination change. The biggest advantage of this strategy is to reduce propellant mass to be carried drastically, even close to zero, like ballistic flight. While the author's work in 2009 presented the trajectories down to almost one year period, this paper will present the further sequences that make the semi-major axis lower than one AU and lower the perihelion distance closer to the Sun for close-up observation of the Sun.

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Processing Two Line Element sets to facilitate re-entry prediction of spent rocket bodies from the geostationary transfer orbit

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Upper stages of rockets are large objects, which contain components that are known to be able to survive atmospheric re-entry. Such surviving material, for example propellant tanks, will impact Earth's surface and might cause ground casualties. Predicting the satellite re-entry, and thus also impact location is notoriously difficult; re-entry prediction is associated with uncertainty in the order of 10% of the remaining lifetime in-orbit. This makes managing the ground casualty risk, by issuing actionable impact warnings, challenging. Thus, the risk posed by spacecraft re-entries will be reduced if the accuracy with which such events can be predicted is improved.
At present, Two Line Element sets (TLEs) are the only publicly available data that can be used for re-entry prediction of a space object. However, there is a number of factors that, if unaddressed, could reduce the accuracy of re-entry prediction based on TLEs:

1. The quality of TLEs of an object is not homogeneous; sometimes TLEs of low quality or even belonging to a different object are published.

2. Occasionally, the object or its orbit can be altered by collisions, fragmentations or space weather phenomena. Such space events render the TLEs of the object from before the event inapplicable to its new, changed state.
3. TLEs do not provide information on space object parameters, such as ballistic coefficient (BC) or solar radiation pressure coefficient (SRPC). TLEs only include the B* parameter that accounts for combined atmospheric drag and solar radiation pressure forces, not BC and SRPC individually.
4. TLEs can only be propagated using the SGP4/SDP4 propagator. However, this propagator is based on the Brouwer theory and, therefore, only models the largest perturbations affecting a satellite. The many assumptions of the theory can severely limit the accuracy of the resulting propagation and thus of the re-entry prediction.
5. TLEs are not supplied with uncertainty information, e.g. a covariance matrix. It is thus challenging to estimate the accuracy with which the re-entry is predicted based on these ephemerides.

In order to overcome these difficulties in TLE-based re-entry prediction, a multi-step procedure is proposed. The first step consists of analysing TLEs, with the goal of identifying outliers, space events, and dynamical phases of re-entry, where the drag is relatively high or low. The filtered TLEs are then used to estimate the unknown spacecraft BC and SRPC. The last step consists of performing an orbit determination in which the TLEs (and derived osculating elements) are used as pseudo-observations.
This paper presents the approach adopted to process the TLEs to improve the accuracy of re-entry prediction. This processing is based on methods previously employed to detect space weather events, which slide a window through the orbital elements contained in the TLEs or derived quantities. Details of the algorithm, which enables TLEs of varying quality and generated in different phases of re-entry to be analysed using the same method, are given. Then, the method used to distinguish between space events and outlying TLEs is described. The trade-off between the number of false positives and negatives, i.e. incorrectly identified and missed outliers, is emphasised. The results of applying the TLE processing methodology to several example rocket bodies are presented in detail and discussed in the context of the accuracy of the resulting re-entry prediction.

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Observation of orbital debris with space-based space surveillance constellations

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Since the first orbital launch back in 1957, the population of space debris in orbit around the Earth has steadily risen.
As the orbital debris population grows, the likelihood of catastrophic phenomena like the collision between two orbiting objects increases. In order to limit the proliferation of space debris in orbit, a great number of standards, guidelines and even laws have been put in place since the end of the 90's. In this scenario, a thorough and accurate bookkeeping of space objects is paramount. Space surveillance has thus become our most reliable ally to safeguard space missions from the threat of collisions.
The BAS3E simulator (Banc d´Analyse et de Simulation d'un Système de Surveillance de l'Espace) is a CNES software tool, developed in collaboration with GMV, with the following capabilities: orbit determination of space objects, generation of optimum observation plans, collision forecast, anticipation of dangerous reentries, and detection of debris fragmentation. Furthermore, BAS3E has the capability to simulate observations of space objects obtained by a given sensor network taking into account sensor visibility constraints. Orbit and attitude ephemerides, quality, precision,

and usage cost; are some of the parameters that shall be defined for each sensor. Originally conceived for ground-based observations (telescope and radar), BAS3E has been recently enhanced to enable the definition of "orbiting" sensor sites, which allow for the simulation of space-based space surveillance sensors.
Using such space surveillance system simulator, this paper evaluates the feasibility to use on-board sensors for both Low Earth Orbit (LEO) and Geostationary Orbit (GEO) object surveillance. The main goal is to assess the ability of a space-based space surveillance constellation, to detect and catalogue the space debris population on these both orbital regimes. The orbit determination accuracies that can be attained when space objects are tracked by different space-based sensor configurations have also been studied and will be presented in this paper.
Different constellations of equispaced spacecraft following quasi-circular, Sun-synchronous dawn-dusk orbits have been analysed, for which the constellation altitudes and the number of satellites were varied. For simplicity reasons, we assumed that spacecraft followed an attitude profile which ensured the pointing of the on-board sensor towards the object. This assumption permits the study of the attitude constraints required by each sensor in order to detect, track and catalogue a given space object.

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Techniques for assessing space object cataloguing performance during design of surveillance systems

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In order to guarantee safe operation of satellites, space object catalogues must be build-up and maintained. The catalogues should be complete, i.e. contain sufficiently accurate and frequently updated orbital states for all required objects. In theory, completeness of the catalogue is achieved by designing the radar in a way that a major fraction of the object population is considered detectable, i.e. covered by the sensor's field-of-regard and within the sensor sensitivity. However, complete coverage does not necessarily guarantee a proper catalogue build-up, yet. If an object is observed once, it must be re-observed in order to verify its existence and improve the accuracy of the determined state. In a next step, individual observation tracks are combined with each other to further improve the accuracy. Consequently, tracks must be associated to each other, i.e. tested if they originate from the same object or not.
The success rate of the association is dependent on the quality of the tracks, the re-observation time and the re-observation geometry. For surveillance radars, the association performance must be considered as a critical design parameter and can be optimized along with the detection rate during the design process. We outline the underlying techniques and present a simulation-based framework for assessing the surveillance system design in terms of association performance and achievable accuracy.
In the tool framework phased-array radars are defined with a detection figure-of-merit (i.e. ratio of detectable object size at a certain distance), a field-of-view, a pointing direction, and a noise estimate. Then, observations are generated with a realistic object population, e.g., based on ESA's MASTER model. The resulting tracks are associated to each other using covariance-based distance metrics. We address several difficulties which arise during the association, e.g. proper treatment of state uncertainties and robust initial orbit determination. The association performance is analysed for different orbital heights and re-observation conditions for a specific radar design concept. Additionally, the typical resulting orbital state accuracies are presented for the initial orbits as well as for the improved ones.

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MONTE: The Next Generation of Mission Design and Navigation Software

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MONTE (Mission Design and Operations Navigation Toolkit Environment) is an astrodynamic toolkit produced by the Mission Design and Navigation Software Group at the Jet Propulsion Laboratory. It supports operational orbit determination and flight path control for deep space and Earth orbiting flight missions, as well as providing an array of tools that can be used in mission design and analysis. Starting with initial development in 1998, it had a primary goal to encompass into a single software system the capabilities of a large suiteof software including the highly successful DPTRAJ/ODP. Since 2009, the Masar project has funded Monte to include development of mission designcapabilities including trajectory optimization and analysis as well as 3d visualization. It was first used in flight operations starting with the launch of Phoenix to Mars (2007) and currently is the prime Orbit Determination software for all JPL missions. Monte has also been used to support missions from ESA, JAXA, and ISRO. The mission design capabilities are being used for the design of future missions including Europa/Clipper, Mars 2020, and InSight. Monte has also been used for non-operations tasks including gravity analysis and satellite ephemeris estimation.
Monte is presented to the user as an importable Python-language module. This allows a simple but powerful user interface via CLUI or script. In addition, the Python interface allows Monte to be used seamlessly with other canonical scientific programming tools such as SciPy, NumPy, and matplotlib. This paper gives an overview of the Monte system, a history of its successes to date, and a preview of what's to come for the software.

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New orbital elements for accurate orbit propagation in the Solar System

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Close encounters with massive bodies, such as planets or Jupiter/Saturn's satellites, make the orbit of any asteroid or spacecraft chaotic. Moreover, in the case of subsequent encounters the Lyapunov time can become very short. Accurate propagation is required in the orbit determination of chaotic bodies, because it mitigates the exponential divergence of nearby orbits. For example, the impact monitoring of natural and artificial objects with the Earth, and the pianification of space missions with several fly-bys, have to be done with mathematical tools that are able to deal with chaos. One of these tools is a reliable and accurate orbit propagator.
We propose new methods to accurately compute elliptic and hyperbolic motion in the Solar System. Our approach roots in the regularization of the two-body equation, which is transformed into a set of linear differential equations with constant coefficients. This result is obtained by introducing a new independent variable (also called fictitious time) and new spatial coordinates in place of the position and velocity.
In the Burdet-Ferrándiz linearization the fictitious time is the true anomaly, and the new state variables are the inverse of the orbital radius, the radial direction and the angular momentum. In this way the motion is decomposed into the radial displacement and the rotation of the radial unit vector. We show that a new linearization of the two-body equation can be obtained with a similar decomposition when either the eccentric or the hyperbolic anomaly is the independent variable. Then, by applying the variation of parameters we introduce six variables that can be used to describe the perturbed motion of the propagated object. The new quantities, together with the total energy and the physical time, constitute the state vector of the special perturbation methods proposed here (the method that works with negative total energy is described in ref. 1). We also investigate the geometrical and physical meaning of the six parameters: they are all related to an intermediate frame which shares with the local-vertical local-orizontal frame the

direction of the angular momentum. This slowly moving frame recalls the ideal frame discovered by the Danish astronomer P. A. Hansen in 1857, which plays a key role in Deprit's (ref. 2) and Peláez's (ref. 3) sets of orbital elements.
The performance of the new formulation has been evaluated for geocentric motion and for trajectories with close encounters. We found a considerable advantage with respect to the traditional integration in Cartesian coordinates.
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Recursive estimation of non-gravitational perturbations from satellite observations

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Thanks to high-fidelity ephemeris and detailed gravitational maps, third-body and non-spherical gravitational perturbations can be modeled with sufficient precision for most applications in low-Earth orbit (LEO). On the contrary, owing to severe uncertainty sources and modeling limitations, mathematical models of the main non-gravitational forces – namely, aerodynamics and solar radiation pressure (SRP) – are generally biased even when advanced formulations are considered.
To date, accurate satellite drag and SRP estimation is only envisaged in challenging missions and the recursive estimation of non-gravitational forces is generally carried out by means of high-sensitivity accelerometers. Nonetheless, unmodeled force estimators using satellite observations only were also proposed. The method of dynamical model compensation (DMC) is arguably the most popular example of this class: first, an underlying parametric model of the unknown perturbation is adopted; then, the parameters of such model are assumed to be first-order Gauss-Markov processes and they are appended to the state vector of a recursive estimator (most often an extended Kalman filter). Provided accurate and sufficiently dense satellite observations, DMC was successfully applied to the estimation of atmospheric force. In that study, no other process noise but the one in the atmospheric force itself was considered.
In the broader context of Bayesian estimation of dynamical systems, sequential Monte Carlo (SMC) algorithms – which include the popular particle filters – are valuable tools to optimally approximate the posterior distribution of hidden Markov processes. Compared to Kalman filtering techniques, particle filters do not require any assumption on neither the linearity of the system nor the nature of the noise. SMC was used to tackle several problems in astrodynamics but, to the best of our knowledge, it was not applied to non-gravitational force estimation, yet.
In this paper, *we propose an SMC algorithm for the recursive inference of non-gravitational perturbations from satellite observations* with no supporting in-situ acceleration measurements. Our approach is conceptually similar to DMC but, on the top of the previously mentioned advantages and drawbacks of SMC, we show that it provides good estimates of the non-gravitational perturbations even when fairly inaccurate measurements and a modest underlying propagator are used. The filter works by updating the empirical distribution of a prescribed number of weighted particles. Each particle consists of one set of orbital elements and some parameters involved in the computation of the forces, e.g., drag and reflectivity coefficients. Weights are assigned to the particles based on the agreement between propagated orbital elements and observations. Secular

effects of the non-gravitational perturbations allow “good” particles to emerge when the weights are updated.
Mean orbital elements are exploited as measurements. They can be obtained by either converting GPS states with a contact transformation or using two-line elements (TLE). This feature allows analytical and semi-analytical propagators, e.g., SGP4, to be naturally integrated in the algorithm to propagate particles. For these reasons, this work can be a valuable resource both for space situational awareness applications, e.g., space debris’ characterization from TLE, and to enhance short-term trajectory predictions on-board small satellites.

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Scilab open-source modeling & simulation platform

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“Share your knowledge. It is a way to achieve immortality.” - the Dalai Lama.
Scientists and Engineers worldwide leverage platforms to model & simulate, to expand knowledge and to collaborate. Open platforms have a key advantage in that they allow a wider and more open collaboration.
Scilab is a powerful open-source modeling and simulation software platform installed every month by more than 100,000 engineers and scientists from around the world.
The goal of this presentation is :

- to introduce Scilab, its history and its governance
- to showcase some applications of Scilab in the Aerospace industry (CNES, Airbus DS, ESA)
- to demonstrate the functional capabilities of the Software

In addition to the Celestlab flight dynamics application developed by CNES, other Scilab-based applications will be covered :

- Use of Scilab for the Rosetta/Philae mission mission
- Sizelab: An Airbus DS application for the mechanical pre-sizing of Ariane 6 launcher
- The Aerospace Blockset developed during the ESA Summer Of Code In Space

Creating knowledge, converting knowledge into applied research and bringing applied research to the industry through sustainable commercial projects is accelerated by connecting these different pools of talent:

- academic experts and students in Universities and Engineering schools
- scientific experts in research centers (public and private)
- research and development teams in industries and companies

The vision we want to share and convey through Scilab and the open-source software is the power of connecting different and complementary talents.
Innovators, inventors and entrepreneurs ultimately prefer to build on open platforms because Open platforms are able to create the most value in the long run.

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An Update on NAIF’s Package of "SPICE" Astrodynamics Tools

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"SPICE" (*) is an information system, comprising both data and software, providing engineers and scientists with the geometry data needed to help design robotic solar system missions, conduct mission engineering operations, plan observations from instruments, and analyze the data returned from those observations. The SPICE system has been used on the majority of worldwide planetary exploration missions since the time of NASA's Magellan mission to Venus, and it appears to be the ancillary information system of choice for most future solar system exploration missions. Along with its "free" price tag, portability and the absence of licensing and export restrictions, its stable, enduring qualities and substantial user support in terms of training and consultation help make it a popular choice.
A description of SPICE was presented at the original ICATT symposium in July 2001. This presentation will bring attendees up-to-date on the current capabilities of SPICE and plans for its further evolution. Part of this presentation will include a demonstration of a new visualization tool: Cosmographia. A companion presentation will highlight the WebGeocalc geometry engine.
The research described in this publication was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.
(*) Spacecraft, Planet, Instrument, Camera-matrix, Events

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SMART-UQ: Uncertainty Quantification Toolbox for Generalized Intrusive and Non Intrusive Polynomial Algebra

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The paper is presenting a newly developed modular toolbox named Strathclyde Mechanical and Aerospace Research Toolbox for Uncertainty Quantification (SMART-UQ) that implements a collection of intrusive and non intrusive techniques for polynomial approximation and propagation of uncertainties. Non intrusive methods build the polynomial approximation of the uncertain states through sampling of the uncertain parameters space and interpolation. Intrusive methods redefine operators in the states model and perform the states evaluation according to the newly defined operators.
The main advantage of non intrusive methods is their range of applicability since the model is treated as a black box hence no regularity is required. On the other hand, they suffer from the curse of dimensionality when the number of required sample points increases. Intrusive techniques are able to overcome this limitation since they have lower computational cost than their corresponding non intrusive counterpart. Nevertheless, intrusive methods are harder to implement and cannot treat the model as a black box. Moreover intrusive methods are able to propagate nonlinear regions of uncertainties while non intrusive methods rely on hypercubes sampling.
The most widely known intrusive method for uncertainty propagation in orbital dynamics is Taylor Differential Algebra. The same idea has been generalized to Tchebycheff and Newton polynomial basis because of their fast uniform convergence with relaxed continuity and smoothness requirements. However the SMART-UQ toolbox has been designed in a flexible way to allow further extension of the intrusive and non-intrusive methods to other basis.
The Generalized Intrusive Polynomial Expansion (GIPE) approach, implemented in the toolbox and presented here in the paper, expands the uncertain quantities in a polynomial series in the chosen basis and propagates them through the dynamics using a multivariate polynomial algebra. Hence the operations that usually are performed in the space of real numbers are now performed

in the algebra of polynomials therefore a polynomial representation of the uncertain states is available at each integration step. To improve the computational complexity of the method, arithmetic operations are performed in the monomial basis. Therefore a transformation between the chosen basis and the monomial basis is performed after the expansion of the elementary functions.
Non intrusive methods have been implemented for a set of sampling techniques (Halton, Sobol, Latin Hypercube) for interpolation in the complete polynomial basis as well as on sparse grid for a reduced set of basis.
In the paper the different intrusive and non intrusive techniques integrated in SMART-UQ will be presented together with the architectural design of the toolbox. Test cases on propagation of uncertainties in space dynamics with the corresponding intrusive and non intrusive approaches will be discussed in terms of computational cost and accuracy.

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CelestLab: Spaceflight Dynamics Toolbox for Mission Analysis

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CelestLab is a free and open source spaceflight dynamics Scilab toolbox developed by CNES that is particularly suited to mission analysis. The toolbox contains about 250 functions related to coordinate systems (IERS 2010 conventions), orbit propagation, geometry and events, force models, orbit properties and more. Lots of examples, demos, help pages and tutorials are provided which makes it easy for new users to get started.
The first version was put on ATOMS website (website from which Scilab modules can be downloaded) at the end of 2009 (see http://atoms.scilab.org/toolboxes/celestlab). The module is one of the most popular as the number of downloads for all versions is now over 34000. CelestLab is 100% Scilab language, which makes it easy to install or modify as there is no need for compilers for instance. But there are situations where this approach is limiting. This is the case when open source software exists that we would like to use through Scilab. There are at least 2 main advantages to using existing software: less developing and testing effort is required, and also, the features that are made available are guaranteed consistent with the original versions (considered as standards). That's why CelestLabX (CelestLab extension module) has been created. It contains low level interfaces to available software. The last version of CelestLab/CelestLabX provides utilities related to TLEs (interface to C code for the propagation of Two-Line Elements from: http://celestrak.com/publications/AIAA/2006-6753) and to STELA (Semi-analytic Tool for End of Life Analysis, tool used at CNES and elsewhere for orbit long-term propagation in particular, see https://logiciels.cnes.fr/content/stela?language=en). The presentation will show CelestLab contents in more details and will more particularly focus on recent features. Concrete example of mission analyses based on previous work conducted at CNES will also be shown in order to illustrate some typical uses of CelestLab.

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STAVOR: a mobile application for the space domain

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Modern scientists and engineers use complex software tools to ease and accelerate their work.

In the space domain, the calculi are too heavy to be computed manually, especially optimization algorithms. Software is therefore mandatory for any realistic task. One issue common to most high level technical software is that validating an operational system takes a long time, and during this delay, the available technologies change very quickly. In the aerospace domain we usually found that the available validated applications are not up to date to the latest technologies and therefore, they don't maximize their capabilities, they become complex and they are not user friendly.
We think that standardized validated and operational computing engines should be combined with some, less validated software but using last technologies in order to complete the market needs and get the best of both worlds. These new tools should be very simple, intuitive and interactive to maximize the help to the user; and should be only used in cases where the rapidity prevails to the accuracy of the results: like in education, preliminary analysis or similar cases.
The new technologies that are interesting for this kind of software are: latest 3D visualization technologies, interactive data plotting, touch screens with the corresponding control gestures, mobile platforms, cross-platform solutions. . .
At CS Systèmes d'Information, we decided to create one of these tools as an example, using as much as possible these new technologies. The company is the main developer of an open-source space dynamics library called Orekit. This library has been already used operationally by many actors and proved itself to be very powerful and validated; nevertheless, it demands a minimum of programming skills to be used to its full potential. People in the aerospace domain do not always have this knowledge, nor should they have it. By creating this new tool, the main functionalities of the library should be bypassed to an interactive UI, and therefore, erasing the need of informatics knowledge to use it. With this move we reduce its functionality but we increase largely the target users.
The application itself is a mobile platform (Android), using 3D representations with modern cross-platform languages (WebGL) and controlled by an interactive touch-ready UI. It consists of a simple space mission simulator, and three different results visualization screens: two 3D modules for the attitude and orbit representations, and a 2D cartographic view to represent the information over the Earth surface. The main uses for this application are the education, due to simple, interactive and intuitive visualizations and its portability; but also more industry oriented applications like orbits comparison, preliminary mission analysis, information exchanges between coworkers. . .
This product, called STAVOR, has been conceived as a quick aid application to have in the smartphone or tablet, to coexist with the normally used desktop tools.

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The Pointing Error Engineering Tool (PEET): From Prototype to Release Version

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Recent standardization efforts in Europe have led to the publication of the ECSS Control Performance Standard E-ST-60-10C and the ESA Pointing Error Engineering Handbook ESSB-HB-E-003, which are instrumental in defining a clear pointing error engineering methodology for ESA projects.
The necessity for such methodology gains in importance as current and future ESA missions, especially scientific and laser-communication missions, put more and more stringent pointing performance requirement on the spacecraft systems. As a consequence even small performance changes may not only lead to additional iterations of the design, but potentially result in prominent

mission-critical changes in subsystems and equipment selection. The rapidly growing system complexity for such kind of missions complicates the situation even more.
Hence a systematic and user-friendly software tool that is capable of automating the pointing performance management process based on the standardized methodology and which replaces the conventional manual computation is largely beneficial and necessary. To complement and support dissemination of this pointing error engineering methodology, a software prototype called Pointing Error Engineering Tool (PEET) was developed and released under the ESA Software Community License.
PEET is designed as an extension to MATLAB and completely runs inside the MATLAB environment to exploit its computational features. It provides a graphical user interface implemented in Java suitable for building pointing systems and analysing pointing errors. The core of the PEET software which contains the mathematical algorithms for error computation is implemented using MATLAB classes.
The prototype software released in the end of 2012 (with several updates until 2015) was restricted to the "simplified statistical method" described in the ESA Pointing Error Engineering Handbook, i.e. results are based on the assumption that the central limit theorem is applicable and the total error follows a (nearly) Gaussian distribution. This assumption may lead to significant deviations from the actual result, in case dominant non-Gaussian error contributions exist.
This paper focuses on the improvements in the ongoing development of the prototype to a release version. The key update supersedes the mentioned restriction on Gaussian distributions and implements the so-called "advanced statistical method" described in the ESA Pointing Error Engineering Handbook. In particular, this implies that probability density functions of (arbitrarily correlated) random variable error contributions - rather than only fundamental statistical moments – are taken into account for an accurate level of confidence evaluation of requirements. The benefits of this approach will be highlighted by comparison of the results obtained with both methods.
In addition, a compliant generalization of the "statistical interpretation" concept in the ECSS Control Performance Standard is introduced which allows more flexibility in the definition of requirements and can help to avoid over-conservative budgets. By supporting script-based execution from MATLAB, the release version of PEET also directly supports its integration in a tool-chain, e.g. for optimization runs or parameter trade-off studies.

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Assessing Orbit Determination Requirement with Unscented Transformation: Case Study of a Lunar CubeSat Mission

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The orbit determination (OD) requirement for a lunar CubeSat mission is examined. The motivation for this study is the NASA CubeQuest Challenge, for which the winning 6U CubeSats in the competition are offered a launch on the Exploration Mission (EM) 1 mission as secondary payloads. All such CubeSats will be disposed into a high-energy trajectory that will fly by the moon. Unless high-impulse chemical propulsion system is allowed on the CubeSats, most designs will involve some form of a low-thrust propulsion system to achieve lunar orbit. In order to determine the OD strategy for such a mission, the OD accuracy requirement needs to be understood. One driver for the OD accuracy is its contribution to the delta-V budget and hence the spacecraft's ability to achieve the target lunar orbit. Typically, this type of analysis is done using Monte Carlo simulations, but the large number of cases required to achieve a statistically significant result is often prohibitive.
In this paper, we examine the use of unscented transforms1 to determine the impact of OD accuracy on the delta-V budget. We take a candidate low-thrust propulsion trajectory from EM-1 disposal to lunar orbit and a candidate ground station architecture using one-way Doppler measurements to determine the OD accuracy requirement. This method is not unlike the linear

covariance analysis2, however, its use of sigma points extends its usefulness beyond the linear region, especially for the highly nonlinear problem of the low-thrust transfer to the moon.
Two open-source tools from NASA Goddard Space Flight Center are utilized to perform this analysis: the General Mission Analysis Tool (GMAT)3 is used for the low-thrust maneuver planning and the Orbit Determination Toolbox (ODTBX)4 is used for the orbit determination. GMAT and ODTBX are interfaced with a function that generates the sigma points of the unscented transformation from the orbit estimate covariance matrix obtained from the orbit determination. Another separate function combines the sigma points of the delta-V penalty. The efficacy of the unscented transformation method is demonstrated by comparing the results of this technique with the results from a small-scale Monte Carlo simulation.
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A TLE-based Representation of Precise Orbit Prediction Results

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A TLE-based Representation of Precise Orbit Prediction Results
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Openly accessible TLEs are widely used to predict orbit positions of space objects. The computing efficiency of the analytic SGP4 algorithm is particularly attractive in applications where orbit positions of thousands of objects are needed. On the other hand, the accuracy of the TLE-computed positions is far less satisfactory than that of orbit predictions using more rigorous orbit determination and prediction methods. The delivery of the accurate orbit determination and prediction results is usually in the form of evenly spaced position and velocity, that would need a data file with size of hundreds of KB. This would require large computer storage for hundreds of thousands of space objects, and the use of these orbit data files in the space conjunction analysis would also require large computing storage.
This paper presents an algorithm using TLE/SGP4 to represent the accurate orbit prediction results. First, the accurately predicted positions are used as observations to determine a set of TLE employing the SGP4 algorithm. Then, the differences between the accurate positions and TLE-computed positions are fitted with a series of sinusoid terms. In this way, the accurately predicted position is computed as the sum of the position computed using TLE and the correction computed using the series of sinusoid.
The algorithm is experimented for a prediction time of 30 days for 4 satellites at altitudes of 690km, 820km, 1500km and 5800km. The accurate orbit predictions are obtained after the orbit determination using laser tracking data. 100 simulation runs are performed for each satellite. The maximum errors representing the accurately predicted positions over the 30-day prediction period are about 220m, 85m, 76m and 34m for the four test satellites, respectively. The representation accuracy clearly has dependency on the orbit altitude.
Experiment results show that the predicted orbits over a period as long as 30 days can be represented in the proposed algorithm in accuracy at 200m for satellite at altitude of 700km, and

30m for satellite at altitude of 5840km. The algorithm is purely analytic, so it is computationally very efficient. Also, it uses TLE and a short series of sinusoid, only a data file of 1 KB is needed to represent the predicted orbit. Therefore, the algorithm could find application in the future space conjunction analysis involving hundreds of thousands of debris objects.

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First results of IOTA (In-Orbit Tumbling Analysis)

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Estimating and predicting the tumbling motion of orbital debris is not an easy task. While in orbit an object experiences perturbations due to external forces in the near-Earth environment. This changes the object's orbit and attitude over time. There are observational techniques that are being used to determine the spin rate of orbital debris such as light curve observation, satellite laser ranging and radar measurements. It is difficult to determine the true attitude motion without combining the results from the observations and comparing them with a model.
IOTA is a prototype software developed under ESA's "Debris Attitude Motion Measurements and Modelling" project by Hyperschall Technologie Göttingen (HTG) to address this issue. The software will provide short-, medium-, and long-term propagation of orbit and attitude motion (six degrees-of-freedom), taking into account all the relevant forces and torques acting on satellites and space debris. External influences included are gravitational forces from the Sun-Earth-Moon system, aerodynamic drag, solar radiation pressure, eddy current damping and momentum transfer from micrometeoroid impacts as well as internal influences such as reaction wheel behaviour, tank sloshing, magnetic torquer activity and thurster firing.
Post-processing of the simulated result will enable generation of synthetic measurements of observation. Combining and comparing the observational with the simulated results increases the accuracy of the attitude motion determination and will lead to better understanding of the attitude evolution.
The software is still under development. In this paper the software implementation of the environmental influences and propagation of the state vector will be discussed and preliminary results with of the attitude motion of ENVISAT have been simulated to provide an example of the capabilities of IOTA.

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Debris cloud analytical propagation for a space environmental index

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Environmental indices for spacecraft are meant to rank objects in orbit depending on their effect on the space environment in case of fragmentation. These indices are usually designed to identify potential candidates for active debris removal and several authors have proposed formulations of environmental indices, which take into account different aspect of the debris environment (e.g. background density, spacecraft mass, orbital altitude). The index proposed in this work focusses

on the assessment of the severity of the breakup of a spacecraft estimating the resulting collision probability for operational spacecraft.
First, a grid in semi-major axis and inclination is defined to map the possible initial conditions of the breakup. For each point in the grid, the NASA breakup model is applied to generate the fragment cloud (considering all objects larger than 1 cm). The case of catastrophic collision is considered to model the effect of the spacecraft mass. The spatial density of the cloud is analytically propagated for 25 years by applying the continuity equation to model the effect of atmospheric drag. An analytical approach is also used to compute the collision probability between the fragments in the cloud and target spacecraft crossing it. In this application, the target spacecraft represent the fleet of operational satellites on which the consequences of the breakup are evaluated. For computational reasons, not all operational satellites are individually propagated, but their population is sampled, considering the distribution of cross-sectional area, to define a small number (e.g. 10) of representative objects. The value of the environmental index for a specific initial condition is obtained by summing the resulting cumulative collision probability of all representative objects.
In this way, the variation of the environmental index with semi-major axis and inclination is obtained and stored in the so-called "reference layer", which refers to a fixed value of fragmenting mass. The dependence on the mass is introduced by rescaling the "reference layer" following the power law that, in the NASA breakup model, relates the number of produced fragments and the fragmenting mass. One interesting aspect of the proposed index is that the computational effort is required only for the generation of the "reference layer", which needs to be recomputed only in case of a significant variation in the distribution of active satellites. Given the "reference layer" and the dependence on the mass, the value of the index for any spacecraft can be obtained by a simple interpolation.
Only a few key parameters (i.e. mass, semi-major axis, inclination) are required to characterise the criticality of a mission. The index can be applied to spacecraft already in orbit (as it was done using the data available in DISCOS) and to future missions, to support their mission design and their licensing process. For example, the environmental index can be employed to assess possible waivers to mitigation requirements for spacecraft, considering the estimated severity of their breakup.

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Group Targets Tracking Using GM-PHD Filter Combined With Clustering

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Group targets tracking is a more complex problem of multi-target tracking. A modified Gaussian mixture-probability hypothesis density (GM-PHD) filter combined with clustering method is proposed. In the updated process of the GM-PHD filter, the proposed method introduces the dummy measurements generated by the group centers to improve the tracking performance, rather than partitions the measurements set. After estimating the single target statements, their similarities are computed. Then the estimated targets are clustered to achieve the group tracking. Finally, the track points of the group centers in adjacent time are connected to obtain the entire trajectories of the group targets. Simulations show that the proposed method can effectively track the group targets and performs better than extended target GM-PHD (ET-GM-PHD).

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From Simulation to Reality: Cataloguing of Objects in from Ground-based Optical Observations

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This paper presents the approach for building a catalogue of Earth-orbiting objects from optical observations in surveillance mode. The cataloguing is based on the DEIMOS CORTO tool (CORrelation Tool) which is described in this paper. CORTO has evolved from of a simulated cataloguing system to a cataloguing software suite intended to process data from real observations. Thus, it merges the knowledge derived from simulation experience and the main constraints imposed by real observation activities. The main differences among these two approaches are highlighted in the paper. CORTO processes observations from several sensors in a sequential way. For each incoming observation, it attempts to correlate it to an existing object in the catalogue. If such correlation is possible, an orbit determination process is performed on the incoming measurement in the basis of a-priori state vector and covariance of the object. If such correlation is not possible, an initial orbit determination is carried out, and a new object associated to that measurement is created. CORTO allows a cold start of the cataloguing (i.e, it does not need any external catalogue to start) and thus, can maintain orbital information of object not included in public TLE file. The main results from DEIMOS cataloguing experience are summarised, describing the observation strategy and the measurement distribution considered necessary for achieving a proper cataloguing capability. This summary highlights the main difficulties that can be found, in the correlation activities which impose a several-step approach to correlation in order to avoid miss-correlation of objects. The approach undertaken in the CORTO software is based on a three step process: firstly, a correlation in the basis of comparison of observation with expected visibility periods and rough observation angles is carried out for every object. A second orbit determination compatibility cross-check based on the filtering residuals is performed later. Finally, a procedure for removing false objects (i.e, objects created by spurious measurements), and/or to remove objects which are observed sparsely is performed asynchronously. The system is intended to run in a mostly automated way, but allows an operator to assess the correlations performed automatically by the system, and to correct them if necessary. In addition to this, it is possible to correct errors related to manoeuvring objects. If the operator knows with certainty that a manoeuvre has taken place, information regarding that manoeuvre can be loaded into the catalogue. The operator can also infer when an impulsive manoeuvre has happened with support from the system. Finally, the system allows the user to perform an iterative process in order to estimate the area-to-mass ratios associated to each of the objects in the catalogue. The CORTO cataloguing system is accompanied by a set of auxiliary tools, also described in the paper, which complete the capabilities of the system to ensure the proper cataloguing process. These tools include: CALMA for calibration of observation stations (used to qualify a number of observatories), CORTOEditor, to support operator for operational maintenance of the catalogue, and CHOCO which optionally allows correlating the observed objects with the TLE data. This tool serves to assign the international ID to the CORTO objects, but is not mandatory for successful correlation of objects within CORTO. The catalogue is finally made available through a restricted web system (CAWEB) that supports the monitoring of the catalogue. The paper presents the main results from an observational campaign executed in October 2014 focused on the cataloguing of high altitude objects. The campaign lasted 9 consecutive observing nights, providing more than 200.000 observations from three surveillance and a tracking telescopes located in Spain. Those observations are used to feed-up the CORTO cataloguing system, and have allowed creating a catalogue of objects which are observable from southern Europe. In particular GEO ring longitudes covering Europe are well represented. About 300 objects are systematically observed during several nights, eventually reaching accurate orbits. The achievable accuracy of the observed orbits can reach values around 10-100 meters. Object manoeuvres are also observable. Example cases of observed manoeuvres are reported.

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WebGeocalc: Web Interface to SPICE

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The WebGeocalc tool (WGC) provides a web-based graphical user interface to many of the observation geometry computations available from the "SPICE" system (SPICE=Spacecraft, Planet, Instrument, Camera- matrix, Events; see http://naif.jpl.nasa.gov). It is based on client-server architecture. A user selects a computation and computation parameters using GUI widgets in a standard Web browser. The request is sent to a WGC geometry server that performs the computation and sends results back to the client.

WGC can compute various geometric parameters such as positions, orientations, and surface intercept coordinates for a given time or a series of times; perform geometric event searches for time intervals when a geometric parameter had a particular value or when a geometric condition is met; and perform time conversions. By a simple drag-n-drop action output value or time intervals from one WGC computation can be saved within a WGC session for use in subsequent computations, allowing users to perform cascading geometric event finder searches, with output from one search being used as input for the next one. Saved intervals can also be used to compute geometric parameters over a time window retuned by a search.

WGC offers many useful features such as plotting results and downloading results. It can plot geometric parameters versus time and versus each other, with both kinds of plots providing zoom capability, with ability to return to the previous or initial zoom levels. It presents numeric results in a scrollable table that can be downloaded in MS Excel, CSV, and simple text format.

Currently the NAIF group at JPL operates a WGC server (see http://naif.jpl.nasa.gov/naif/webgeocalc.html) that has access to all data on the NAIF web site. This server is configured to provide easy access to generic SPICE kernels and to SPICE data sets formally ingested in NASA's Planetary Data System. Users can select these data sets by simply picking them from the WGCs kernel set menu. Users can also load into WGC any kernels available on the NAIF server, including operational kernels for many missions, but it has to be done manually, which is more time consuming and requires some knowledge of kernel naming schemas and contents.

WGC can be deployed on any computer that has Java, Apache Tomcat web server, and MySQL Community Server. While NAIF does not plan to make the WGC software available to the general public, it might make the WGC binary war file together with installation and kernel database configuration instructions available to organizations involved in planetary exploration, with significant experience with SPICE and a clear need to manage their own kernel sets used by WGC.

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An open-source, modular software architecture for astrodynamics simulation

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Open-source software plays an increasingly important role in astrodynamics research. In this paper, we describe a new open-source, modular software architecture for astrodynamics simulation. We describe the rationale behind the setup of the software architecture and provide a snapshot of an implementation in C++ using generic programming concepts. The implementation, dubbed "astropnp", includes a few different libraries that provide the basic building blocks to construct astrodynamics simulations. We highlight how this architecture lends itself to "plug-and-play" use and provide insight into a simulation use-case for spacecraft rendezvous.

Our paper describes an attempt towards building an architecture for astrodynamics simulations based on the notion of "micro-services" (Kosar, 2012). The concept of micro-services stems from the idea of building up software using "services" that are small, highly decoupled and focussed on executing small, isolated tasks. Micro-services architectures also feature services that can be

swapped out easily. In the most general sense, these services are orchestrated by language-agnostic Application Program Interfaces (APIs). Implementing an architecture of this nature affords the software a great deal of versatility and modularity.
We survey the use of the micro-services approach towards the implementation of software for, but not limited to, mission analysis, interplanetary trajectory design, rendezvous & docking and formation flying. In addition, we present a case study based on astropnp for spacecraft rendezvous using robust, nonlinear, feedback control. The services for this case study include a generic mathematics library, a generic astrodynamics library and a library with functions for proximity operations in space (in addition to standard libraries for linear algebra, storage etc.). The case study is conducted within the context of a mission concept for Active Debris Removal (ADR), as part of the Stardust network (EU FP7 Marie Curie Initial Training Network).
The software architecture presented here follows on from previous work to establish the TU Delft Astrodynamics Toolbox (Tudat) (Kumar, et al., 2012). Our study includes an overview of existing open-source software tools for astrodynamics, including Tudat, General Mission Analysis Tool (GMAT), Java Astrodynamics Toolkit (JAT), Orekit and PyKEP. Based on this overview, we comment on how these tools might be employed within our modular software architecture. We present a roadmap to improve upon the architecture and generate a tasklist to raise the readiness level of astropnp in anticipation of future wide-spread use.

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Uniform Trajectory Locators (UTLs) – an open API for trajectory discovery and utilisation

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With the dramatic increase in rideshare opportunities for CubeSats and other small spacecraft, landers and rovers, platforms offering to fly large numbers of virtual payloads, and projects to manufacture spacecraft-on-demand on orbit by the thousand, discovery and planning of trajectories for actual and potential space objects is becoming a bottleneck. Multiple organisations have, or are planning, projects that require the launch and calculation of trajectories for hundreds to hundreds of thousands of space systems to be sourced and managed over the next decade.
To address this issue, we propose Uniform Trajectory Locators (UTLs), an open specification for a web based representational state transfer (RESTful) application programming interface (API) permitting the manual and automated discovery of existing, upcoming or potential trajectories for use by new and existing, virtual and physical, space systems and payloads.
UTLs permit the distributed decentralized publishing of trajectories of space objects, whether launch opportunities, existing spacecraft in orbit, or potential opportunities for deployment in low earth orbit, in interplanetary space or on planetary bodies. As well as publishing existing opportunities, UTLs can be used to request trajectory solutions from domain specialists and automated tools, permitting production and operations schedulers responsible for sourcing and managing the launch and operation of constellations of satellites and exploration tools to autonomously search for optimal solutions for missions.
UTLs build on existing web standards such as Uniform Resource Locators (URLs) and eXtensible Markup Language (XML), and return results in an object orientated Trajectory Markup Language (TML). UTLs are designed to permit different levels of truth and detail to be returned by their publishers to consumers by tools ranging in complexity from static text files to sophisticated algorithmic back ends. The level of detail and information provided about the same trajectory by different stakeholders (e.g. launch providers, brokers, potential consumers, space situational awareness providers, etc.) can be substantially different for commercial, political, technical or temporal reasons. The same trajectory can be published by multiple independent distributed providers with a security and transaction model allowing consumer by consumer level information disclosure and automated electronic trading of access to trajectories. UTLs can be used to provide lightweight wrappers around existing tools to provide platform independent interfaces to existing databases of launch opportunities, orbital parameters and flight dynamics tools.

A variety of demonstration proof of concept UTL tools and data sources under development with academic and commercial space systems providers will be presented, with a focus on sub-kilogram scale platforms supporting the mass exploration of space for science, education and commerce.

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Rugged: an open-source sensor-to-terrain mapping tool

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The Sentinel-2 mission is a component of the Copernicus program. It consists of two spacecrafts each carrying a high resolution multispectral imager (13 bands) devoted to environmental, security, and agricultural applications. One of the key feature of Sentinel-2 is the huge amount of imagery data that will be produced, as each spacecraft produces 1.7TB of raw data daily. All data are transformed on ground at the Image Processing Facility to create the various products levels.
One element of this processing chain is the Rugged library: an open-source library built on top of the open-source Orekit space flight dynamics library. Rugged is used to compute very quickly and accurately the mapping between ground points and on-board pixels, taking into account a Digital Elevation Model. Direct location computation is used to identify which ground point is seen by a specified sensor pixel. Inverse location computation is used to identify which sensor pixel will see a specified ground point. These methods are the basic elements for complete processing algorithms. They are computationally intensive due to the very large number of pixels to manage (12 detectors, 13 bands, global coverage of land surfaces).
The geo-location methods are at the boundary between image processing and flight dynamics as they handle accurate geometrical models. Rugged has therefore been designed as an intermediate level library, and it relies on Orekit to compute the global geometry (spacecraft orbit and attitude, Earth precession nutation and proper rotation including all IERS corrections, Earth mean ellipsoidal shape). The Rugged library adds on top of this the Digital Elevation Model intersection computation.
The presentation will describe the overall Rugged technical architecture, the issues that were faced and how they were solved in order to achieve a high performance level while not compromising accuracy. It will also present the open-source strategy and the governance model of the library. The library is already operationally used, some perspectives for it will be discussed.

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A Comparative Study of Programming Languages for Next-Generation Astrodynamics Systems

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Due to the computationally intensive nature of astrodynamics tasks, astrodynamicists have relied on compiled programming languages such as Fortran for the development of astrodynamics software. Interpreted languages such as Python on the other hand offer higher flexibility and development speed thereby increasing the productivity of the programmer. While interpreted languages are generally slower than compiled languages recent developments such as JIT (just in time) compilers or transpilers have been able to close this speed gap significantly. Another important factor for the usefulness of a programming language is its wider ecosystem which consists

of the available open-source packages and development tools such as integrated development environments or debuggers.
The aim of this study is to identify the most promising programming language for developing next-generation astrodynamics systems and tools. This target language shall offer an acceptable compromise between numerical performance and programmer productivity and possess a mature and sustainable ecosystem.
The study compares three compiled languages and three interpreted languages which were selected based on their popularity within the scientific programming community and technical merit. The three compiled candidate languages are Fortran2008, C++14, and Java 8. Python 3.5, Matlab 2015b, and Julia 0.4 were selected as the interpreted candidate languages. All six languages are assessed and compared to each other based on their features, ease-of-use, and ecosystem. Additionally idiomatic solutions to classical astrodynamics problems are developed in all candidate languages and compared based on their performance and simplicity.

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poliastro: : An Astrodynamics library written in Python with Fortran performance

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Python is a fast-growing language both for astronomic applications[[1]] and for educational purposes[[2]], but it is often critizised for its suboptimal performance and lack of type enforcement. In this paper we present **poliastro**, a pure Python library for Astrodynamics that overcomes these obstacles and serves as a proof of concept of Python strengths and its suitability to model complex systems and implement fast algorithms.
poliastro features core Astrodynamics algorithms (such as resolution of the Kepler and Lambert problems) written in pure Python and compiled using numba, a modern just-in-time Python-to-LLVM compiler. As a result, preliminary benchmarks suggest a performance increase close to the reference Fortran implementation, with negligible impact in the legibility of the Python source. We analyse the effects of these tools, along with the introduction of new ahead-of-time compilers for numerical Python and optional type declarations, in the interpreted and dynamic nature of the language.
poliastro relies on well-tested, community-backed libraries for low level astronomical tasks, such as astropy[[3]] and jplephem. We comment the positive outcomes of the new open development strategies[[4]] and the permissive, commercial-friendly licences omnipresent in the scientific Python ecosystem.
While recent approaches involve writing Python programs which are translated on the fly to lower level code, traditional Python libraries for scientific computing have succeeded because they leverage computing power to compiled languages. We briefly present tools to build wrappers to Fortran, C/C++, MATLAB and Java, which can be also useful for validation and verification, reusability of legacy code and other purposes.

Rendezvous & Docking (I) / 32

Experimental evaluation of Model Predictive and Inverse Dynamics Control for spacecraft proximity and docking maneuvers

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An experimental campaign has been conducted to evaluate the performance of close proximity and docking maneuvers of two different guidance and control algorithms based on Model Predictive Control and on Inverse Dynamics Control. The metrics of performance includes fuel usage, time to target, computational burden and constrain handling.
The experiments have been conducted on two ˜10 kg Spacecraft Simulators that float via air-pads over a 4-by-4 meter polished granite monolith surface recreating a reduced gravity and a quasi-friction-less motion in two translational and one rotational degrees-of-freedom (planar motion). By using eight cold-gas thrusters and a reaction wheel, the Spacecraft Simulators are capable of autonomous motion over the floating surface. An onboard tank of compressed air (propellant), a power system and on-board computer give full autonomy to the Spacecraft Simulators. All the required processing (sensor readings, communications, navigation, guidance and control, and actuator commanding) is handled on-board in real-time. The experimental set-up will be described in detail. The navigation problem has been considered solved with the Spacecraft Simulators sensing their position and attitude by using an overhead optical positioning system (VICON) augmented by an on-board Fiber Optics Gyroscope. The accuracy of the Spacecraft Simulator position and attitude knowledge (as well as the knowledge of the target state) can be artificially deteriorated to simulate real sensor limitations and constraints (e.g precision and field-of-view of a RADAR system).
With the Model Predictive Control framework, a cost function (e.g fuel consumption) subject to system dynamics and constraints (e.g. maximum available control actuation level and obstacle avoidance) is minimized over a discretized time period with a finite prediction horizon. By solving the optimization problem, a control input for each discrete time sequence is generated.
In the Inverse Dynamics Control approach the trajectory of the spacecraft is simplified to a known function depending on a set of parameters (e.g. a polynomial). The desired initial and final conditions are then imposed on that trajectory and the parameters that are left unset are then optimized to meet other constrains whilst minimizing a cost function (e.g. fuel consumption). The control input to follow the prescribed trajectory is then generated from the prescribed trajectory and applied to the system.
Both guidance and control approaches eventually reduce to two different non-linear optimization problems that need to be solved periodically in order to provide the control inputs. In this study, the open-source Interior Point OPTimizer (IPOPT) software package has been used.
The set of test scenarios that have been conducted are designed to represent a wide set of rendezvous and proximity operations scenarios (unconstrained and constrained, cooperative and uncooperative docking and proximity operations with or without obstacle avoidance). The goal has been to define a set of tests that can be used to benchmark different guidance algorithms so that a meaningful comparison of different approaches can be made.

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GNC Techniques for Proximity Manoeuvring with Uncooperative Space Objects

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Active debris removal and satellite servicing are some of the current hot spots in space research: plenty of engineering challenges, they deal with fully or partially uncooperative orbiting objects to be approached and captured autonomously by another space vehicle. The Active Debris Removal (ADR) topic focuses on trading-off, designing and making operational mechanisms placed on board an active chaser that can rendezvous with and grapple an inert and tumbling target, to eventually change its dynamics transferring it to a disposal orbit. On the other hand, satellite servicing (SS) deals with refuelling and/or maintenance of active spacecraft, and therefore supposed to be partially cooperating, to be approached and docked by the active chaser to carry out the needed operations when connected. To perform these tasks, different techniques are currently being

proposed in literature, starting from the robotic arm to grasp the target to tethered nets/tentacles to wrap it. From dynamics point view, these technologies differ for the flexibility involved in different elements and connections. A general-purpose system design should effectively intervene on objects different in configuration, materials and possibly in dimensions.
ADR and SS tasks define new challenges for Guidance, Navigation and Control (GNC): these missions cannot be tele-operated and ground-controlled due to communications delays, intermittence, and limited bandwidth between the ground and the chaser. Therefore, there is substantial interest in performing these operations autonomously: the research work, here presented, moves in that direction and have the main objectives of • developing reliable and validated dynamics models, to drive ADR and SS systems design and support GNC implementation, including the flexibility modelling and contact dynamics of capture mechanisms and coupled stacks configurations; • implementing GNC laws adapted to perform the involved operations, from approaching to removal\servicing, to demonstrate mission feasibility and increase the level of autonomy; • validating dynamics models and control laws through experimental activities, including microgravity campaigns and hardware-in-the-loop testing.
A multi-body dynamics simulation tool was developed in house at Politecnico di Milano – Department of Aerospace Science and Technologies, fully integrated in Matlab/Simulink and suited to design guidance and control laws: it provides a fast and accurate simulation environment to describe multiple bodies' six degrees of freedom dynamics, possibly linked by different flexible/rigid connections and including flexible appendages, propellant sloshing and a detailed environmental model to account for all the relevant perturbations, especially at low altitudes. An upgrade of the abovementioned simulator was also implemented to describe the deployment and wrapping dynamics of flexible nets around targets, with the inclusion of collision detection and contact dynamics algorithms. A parabolic flight campaign was successfully performed to validate both flexible dynamics and contact dynamics models: the net 3D trajectory was reconstructed using stereovision (an ad hoc Matlab software was implemented to this end).
In the paper, the abovementioned multibody dynamics tools, their validation process and preliminary simulation output are presented, for both rigid and flexible techniques."

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JOSCAR/JDRAGON: Tools for Maneuver Strategy Computation Developed in Java and using PATRIUS

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JOSCAR/JDRAGON are both new tools of maneuvers strategies computation, developed internally in CNES (Centre National d'Etudes Spatiales, French Space Agency) at the Orbital Maneuvers Office (DCT/SB/MO).
Thus, thanks to the decision, some years ago, to use Java technology, existing Fortran tools and libraries are going to be rewritten specially in order to use the powerful CNES PATRIUS library. In that frame, JOSCAR/JDRAGON tools have been rewritten in Java even if they are always based on the same basic principles of the initial OSCAR/DRAGON Fortran versions which were intensively used for the Automated Transfer Vehicle (ATV) and still today, for the operational design of the LEOP, phasing and rendezvous scenarios for GALILEO missions. This paper will describe the methods implemented as well as the software functionalities, pointing out the differences between JAVA versus FORTRAN version, the first one taking advantage of some PATRIUS new functionalities as well as almost 20 years usage feedback.
Thus, JDRAGON is able of computing a near-optimal mission plan, using initial conditions for target and chaser spacecraft's, an amount of maneuvers to be optimized respecting some constraints of application and rendezvous conditions. It is based on a robust and fast method, which requires calling a numerical propagator iteratively. For this purpose, JPSIMU has been also developed based on its PSIMU predecessor which is the heart of numerous CNES flight

dynamics tools (as in ATV-CC or GALILEO FDS ones). At a higher level, JOSCAR, which uses JDRAGON as a kernel, allows to perform End-to-End Monte-Carlo simulations, necessary to test robustness of the computed strategies for mission analysis purpose.
Above the new design of these tools, thanks to the Java object approach, their validation is also a big challenge. Thematic validations have been performed with not easy comparisons with the Fortran version, given the differences in their corresponding flight dynamics libraries. Special efforts have been put into performance, looking for optimal tools settings, aiming at having both fast computations and satisfactory accurate results. Concerning the quality of the code, Eclipse environment analysis tools have been used in order to be compliant with CNES coding standard rules.
At last, in order to deal with the considerable input/output data generated, a Graphical User Interface has been also developed using GENIUS (a higher level CNES JAVA toolkit based on Swing) which allows using these tools in a more friendly way on many different Operating systems from Windows to Linux.

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An open-source simulator for spacecraft robotic arm dynamic modeling and control.

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A wide range of space missions require a robotic arm (e.g. satellite servicing, active debris removal and berthing). The kinematics and dynamics of space manipulators are highly non-linear and differ considerably from their terrestrial counterparts. The base-spacecraft is not anchored to the ground and thus it is free to react to the manipulator's motion, making the modeling and control of space-based manipulators a complex task. The base-manipulator interaction is stronger when the mass and inertia of the manipulator and of the base-spacecraft are comparable. Therefore, the difference between space and terrestrial manipulators tends to become more acute on manipulators mounted on small spacecraft.
There is a wealth of literature tackling the subject of spacecraft manipulators modeling and control but each research group has typically developed its own code in order to simulate and validate control approaches. In an attempt to help speed the process and make space manipulators a more accessible research topic an open-source kinematics, dynamics and control simulator for space based robotic arms has been developed.
The simulator is written in MATLAB/ Simulink and it can be used for standalone MATLAB scripts and Simulink models. The Simulink models can subsequently be used for automatic code-generation and compiled to run in real-time on the selected target hardware. This open-source code is thus suitable from prototyping work all the way to hardware implementation.
The six degrees-of-freedom simulator is capable of computing the homogeneous coordinate transformation matrices, the velocity Jacobians, the combined inertia matrices and the velocity terms (using the Lagrangian approach) as well as the joint and base reaction using the Newton-Euler approach. Several control manipulator and base-spacecraft control approaches have also already been implemented.
The architecture and usage of the simulator will be presented as well as some examples that demonstrate how the simulator performs under some basic manipulator control applications. Finally an implementation example, on a real-time embedded hardware on an experimental test-bed, will be provided.

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GNC simulation tool for active debris removal with a robot arm

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Studies have shown that the population of large objects has become a problem in Low Earth Orbit (LEO). The danger of collisions is higher than ever before and important objects are at high risk of major damage. One proposed solution for this problem is Active Debris Removal (ADR) using a robotic arm mounted on a spacecraft. A gripper or another adequate tool installed on the robot arm could capture almost every debris part (target) that endangers other satellites in the orbit. Once a connection (docking) is established, the chaser spacecraft can be used to safely deorbit the target by transferring it to a disposal orbit. To analyze this approach, a GNC simulation tool for Active Debris Removal with a robot arm was developed. A realistic benchmark scenario based on the capturing of the inactive Envisat satellite was chosen for a simulation study. The scenario considers uncertainties in the mechanical parameters and measurement noise. It focuses on the most critical phases of an ADR mission. In particular, the rendezvous phase, the capture phase as well as the GNC controlled de-orbiting phase.
The tool is used to design and simulate the GNC algorithms, satellite dynamics, kinematics and environment as well as the robot arm control. The developed control system consists of multiple parts at different hierarchical levels. A trajectory planning module coordinates different controllers of chaser satellite and robotic arm and provides reference trajectories for a successful docking of the two satellites. An optimization based approach trajectory was designed using an inverse satellite model of the chaser and the robotic arm. The results are used as the feed-forward control part of the two degree of freedom control approach. The trajectory design plays an important role to avoid a collision between the rotating passive target and the chaser satellite. Due to the rapid rotation of the target satellite, a simultaneous combined control of the chaser satellite together with the mounted robot arm is necessary, and considering actuator limits is crucial. The feed-back controller synthesis for the thrusters, reaction wheels and robot joint control was implemented using multi-case and multi-objective optimization with parameterizable models generated by the newly developed tool. This results in a robust control setup that is able to handle mechanical uncertainties and sensor noise.
The simulation tool can also be used to visualize the ADR scenario, based on CAD models of the satellite and the robot arm. The object-oriented design allows the change of modular components and parameters of the simulation. Simulation results show that the suggested ADR benchmark scenario can be successfully completed using the developed control algorithms. The object-oriented GNC simulation tool for active debris removal with a robot arm allows the comprehensive design and analysis of an ADR scenario. It can be used to develop and verify the required GNC of the satellite and the control of the robot arm.

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A Sequential Method to Compute Multiobjective Optimal Low-Thrust Earth Orbit Transfers

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A sequential algorithm for optimizing low-thrust Earth orbit transfers in terms of the propellant consumed is proposed. The dynamical model includes the effect of the Earth shadow and the J2 effect of the gravitational potential. The algorithm is based on two steps of growing complexity. In the first step, a near-optimal solution is obtained using simplified dynamics. In the second step, a hybrid approach embedded in a direct collocation scheme is used to consider the optimal coast arcs out of the Earth shadow. This novel approach is a continuation of a previous work in which the minimum time problem was solved. The ultimate goal of the work is to develop a robust and flexible tool that addresses the multi-objective design of low thrust transfers in Earth

orbit. Hence, the user will not only obtain just point solutions, but will be able to explore the set of Pareto optimal trajectories for these two objectives.

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Low Thrust Trajectory Optimization for Autonomous Asteroid Rendezvous Missions

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The demand for deep space missions and the desire to investigate small space objects like asteroids and comets is increasing constantly. Such exploration missions open up the possibility to gain further scientific knowledge about the origin of our solar system as well as to find and eventually mine rare earth elements that are narrow on Earth or even to discover novel resources. To realize such missions, trajectory planning and optimization are of utmost importance. When flying further into outer space, the spatial distances become enormously huge while fuel is very limited and signal runtimes are increasing rapidly. These conditions impose an unmanned and autonomously working spacecraft. The mathematical field of optimization and optimal control provides the foundation for autonomous decisions and facilitates more safety and minimal resource consume. Solutions may additionally be transferred to other, earth-bound applications like e.g. deep sea navigation and autonomous driving with minor additional expenses. The aspects investigated in the present paper focus on the specific challenges of guidance and control regarding the cruise and approach phase of a spacecraft starting in a parking orbit around the Sun and reaching for an asteroid in the main belt. The underlying optimal control problems are solved using so called direct methods also known as transcription techniques. Those transform an infinite-dimensional optimal control problem (OCP) into a finite-dimensional non-linear optimization problem (NLP) via discretization methods. The resulting high dimensional non-linear optimization problems can be solved efficiently by special methods like sequential quadratic programming (SQP) or interior point methods (IP). For solving the problems introduced in this paper the NLP solver WORHP, which stands for 'We Optimize Really Huge Problems', is used, a software routine combining SQP at an outer level and IP to solve underlying quadratic sub problems. Within this paper the transcription is performed using the robust method of full discretization. The trajectory optimization and optimal control problems are modeled and solved using low thrust electric propulsion on the one hand and chemical propulsion on the other hand for comparison. The movement of the spacecraft is described through ordinary differential equations (ODE) considering the gravitational influences of the Sun and the planets Mars, Jupiter and Saturn as well as the different thrust commands. Competitive mission aims like short flight times and low energy consumption can be provided with a weighting factor within the optimization process. The varying challenges of the two propulsion types are analyzed and comparative solutions and results introduced. Several mission trajectories are compared, aiming at different destination asteroids and optimizing with different weighting factors for energy cost and flight time duration in order to investigate the different possibilities of an asteroid rendezvous mission. The results show the huge gain of trajectory optimization as input for on-board autonomous decision making during deep space missions as well as the great increase in possibilities for flight maneuvers by providing solutions for changing and contradictory mission objectives. Furthermore, trajectory optimization can be used to analyze the potentials of different propulsion systems beforehand.

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About Combining Tisserand Graph Gravity-Assist Sequencing with Low-Thrust Trajectory Optimization

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Gravity-assist maneuvers have the potential to be mission enablers, due to “"free energy"” they provide. The efficiency of low-thrust propulsion is further one means of improving mission payload mass. Combining both for a given mission possibly improves overall mission performance, which makes it desirable to investigate low-thrust gravity-assist missions.
For means of investigating a broad range of mission options, the System Analysis Space Segment group of DLR is working on methods of combining the optimization of low-thrust trajectories and gravity-assist sequences with the help of the Tisserand Criterion and shape-based trajectory models. The hurdles faced by violations of Tisserand Criterion premises are shortly discussed and the repercussions these have on planning a gravity-assist sequence for a low-thrust mission. A metholodology, based on benchmarking the results with non-gravity-assist trajectories is presented in this paper, grounded on a loop combining the optimization of the trajectory and the selection of the next gravity-assist partner. Furthermore it is shown how the solution space can be reduced with the help of constraints originating in the maximum possible Delta-V gain and the gravity-assist partner pool.”

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Enhancement of DLR/GSOC FDS for Low Thrust Orbit Transfer and Control

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On behalf of future missions with electric propulsion controlled by DLR’s German Space Operations Center (DLR/GSOC) the present operational multi-mission Flight Dynamics System (FDS) is enhanced to support the preparation and operation of corresponding satellite missions.
For designing an easily extendable framework, low-thrust scenarios including orbit raising to GEO, GEO station keeping as well as LEO missions were considered. Each low-thrust phase is modelled by a thrust profile comprising non-equidistant thrust vector and thrust level. In a first step, we consider the thrust level to be constant over a thrust phase.
Based on this design several multi-mission FD software modules are enhanced to handle low thrust phases, e.g. orbit prediction, orbit determination including thrust level estimation, and generation of orbit related products.
The low-thrust transfer trajectories are optimized by means of the software package ASTOS/GESOP 1. This package was configured within the present FDS with astrodynamics models used at DLR/GSOC to ensure maximum compliance between Mission Planning/Analysis results and operational realization. Operational and technical constraints, e.g. thrust interruption during eclipses, are implemented, too.
Throughout the paper, we will demonstrate the extended FDS capabilities by means of a GEO positioning reference mission. We found a remarkable consistency between resulting ephemerides by the optimizer in comparison to the FDS, which shows the correct processing of thrust profiles within the FDS. For the future the need for extension to variable thrust levels will be analyzed.
1 GESOP© by Astos Solutions (2015), Version 7

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Low thrust orbit transfer optimiser for a Spacecraft Simulator (ESPSS -EcosimPro® European Space Propulsion System Simulation)

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The paper describes the general strategy for electric propulsion orbit transfers, a open source optimiser for orbit transfer based on averaging techniques and the integration of such trajectory optimiser into EcosimPro® ESPSS (European Space Propulsion System Simulation). EcosimPro® is a Physical Simulation Modelling tool that is an object-oriented visual simulation tool capable of solving various kinds of dynamic systems represented by a set of equations, differential equations and discrete events. It can be used to study both transients and steady states. The object oriented tool, with the propulsion libraries ESPSS from ESA for example, allows the user to draw (and to design at the same time) the propulsion system with components of that specific library with tanks, lines, orifices, thrusters, tees. The user enhances the design with components from the thermal library (heaters, thermal conductance, radiators), from the control library (analogue/digital devices), from the electrical library, etc. The use of the new feature included into ESPSS, the satellite library is particularly interesting for orbital manoeuvres because the satellite library includes the flight dynamic (orbit and attitude) capabilities for a full spacecraft including orbital, attitude perturbations and power concerns during Sun's eclipse phases. In order to simulate realistic missions, an optimiser for orbit transfer has been integrated thanks to the design of few new components for interfacing the optimiser and the existing library. Hence the simulations take into account the interactions between the AOCS and the optimal thrust direction wanted to perform the orbit transfer and real strategies for power management during eclipses. Full satellite missions, for example continuous electric orbit transfer from GTO and from super-GTO to GEO are presented and particular behaviours highlighted.

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CAMELOT - Computational-Analytical Multi-fidelity Low-thrust Optimisation Toolbox

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In this work a toolbox for the fast preliminary design and optimisation of low-thrust trajectories is presented. The toolbox, called CAMELOT (Computational-Analytical Multi-fidelity Low-thrust Optimisation Toolbox), solves highly complex combinatorial problems to plan multi-target missions characterised by long spirals including different perturbations. CAMELOT implements a novel multi-fidelity approach combining analytical, adaptive surrogate modelling and accurate computational estimations of the mission cost. Decisions are then made by using two optimisation engines included in the toolbox.
The main elements of CAMELOT are: • Fast Analytical Boundary-value Low-thrust Estimation (FABLE) • Multi-Population Adaptive Inflationary Differential Evolution Algorithm with Adaptive Local Restart (MP-AIDEA) • Automatic Incremental Decision Making And Planning (AIDMAP)
FABLE provides accurate cost estimations of orbital transfers using a multi-fidelity analytical and semi-analytical approach. FABLE implements an analytical propagator that includes perturbations due to the J2 zonal harmonic, drag, solar radiation pressure and low-thrust propulsion. The effect of shadow regions is also included. Different control parameterization can be implemented to analytically compute optimal transfers. In order to reduce the computation burden, FABLE can generate surrogate models of the transfers' cost to allow fast evaluation of complex trajectories.
MP-AIDEA is a single objective global optimiser based on the hybridisation of evolutionary computation and mathematical programming. The optimiser has been designed to automatically adapt its input parameters to the specific problem under consideration, in order to avoid tedious manual tuning of the algorithm.
AIDMAP is an incremental decision making algorithm that allows for planning & scheduling of complex tasks. AIDMAP incrementally builds a decision tree from a database of elementary building blocks. The resulting graph is then evaluated using a set of deterministic or probabilistic heuristics. The deterministic heuristics in AIDMAP are derived from classical Branch & Cut algorithms while the probabilistic heuristics are bio-inspired and mimic the evolution of the slime mould Physarum Polycephalum, a simple organism endowed by nature with a powerful problem-solving heuristic.

CAMELOT has been applied to a variety of applications from the design of interplanetary trajectories to the optimal deorbiting of space debris, from the deployment of constellations to on-orbit servicing.
The paper will present two key applications. One is a multi-fly-by interplanetary mission to the inner part of the solar system to visit the fourteen known Atira asteroids, and search for new ones. CAMELOT was used to generate a globally optimal sequence of asteroids, departure and arrival dates, that allows visiting the maximum number of Atira asteroids in a given time and maximises the chance to discover new ones. The other is a solution to the problem of deorbiting multiple non-cooperative objects from the LEO region. CAMELOT was used to identify the sequence of targets that maximize the number of removed satellites, while minimizing the propellant consumption, using two different strategies: multi-target delivery of de-orbiting kits to perform a controlled re-entry, low-thrust fetch and deorbit with a single towing spacecraft. To speed up the computation, a surrogate model for the ΔV required to realise all possible low-thrust transfers between different targets was used.

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Aerodynamic categorization of spacecraft in low Earth orbits

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Spacecraft re-entering the Earth atmosphere in an uncontrolled manner may get stabilised by restoring aerodynamic torques, if they have an appropriate shape and mass distribution. While the aerodynamic force (mainly drag) is usually a second-order effect compared to the gravitational acceleration by the Earth at altitudes above 150 km, sometimes the aerodynamic torques can already compete with the Earth gravitation gradient-induced torques at altitudes around 250 km and below. Therefore it is of interest to have an understanding of how to compute the aerodynamic torques in this altitude regime.
The usual approach to compute the aerodynamic coefficients at high altitudes is to construct a surface model of the spacecraft, where the surface is either modelled with plane face elements or discretized into small triangular or quadrangular "panels". The aerodynamic coefficients are then computed for each surface element and summed up with an Integral or Monte-Carlo method, utilizing that the flow around the spacecraft can be considered as free-molecular.
While the Monte-Carlo method can give quite accurate coefficients for a given configuration, it does not allow to parametrize the object of interest. This is different to analytical solutions, where the geometric dimensions and mass distribution appear as explicit parameters, and where the influence of configuration changes on the results are direct. On the other hand, the possibility to get analytical solutions is limited to convex geometric shapes. This can be extended to concave shapes by using some kind of shadowing algorithms, but in this case the additional effort needed to examine the shadowed areas can foil the advantages of the analytical approach compared to the numerical analysis.
In any case, the combination of different methods can give an added value. For basic geometries analytical solutions are known. Comparing these solutions with Monte-Carlo results can serve as a calibration method for the Monte-Carlo method's statistical uncertainties. For more complex geometries the Monte-Carlo method can give a measure of the effect of shadowing and multiple reflections, which cannot be considered exactly, or not at all, in analytical solutions or integral methods.
Due to the special form of the free-molecular gas-surface interaction and its momentum transfer some simplifications are possible especially for the typical high-speed conditions in orbit, which can be used to extend the validity of the analytical solutions or at least extend their approximate range of validity. This will be used in the paper for a simplified categorization of spacecraft without extensive multiple internal reflections w.r.t. to their aerodynamic characteristics.

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Optimal Lunar Landing

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"This paper discusses the problem of lunar landing guidance. It is focused on the powered descent portion that leads to a lunar touchdown. The paper starts by discussing the dynamical models and environment models used in the analysis. This includes the spherical harmonic gravity model of the moon and the moon-fixed dynamical models that include the accelerations induced by the moon's rotation. It then discusses two successful lunar landers, the U.S. Surveyor and Apollo Lunar Module, and their guidance systems. This is followed by a literature survey on optimal lunar landing algorithms.

The next sections discuss lunar transfer and give results from a linear tangent landing algorithm. This is a two-dimensional problem with a flat lunar surface. This is applied to a two-dimensional spherical moon problem by rotating the thrust vector as the descent progresses. The terminal algorithm is based on a bang-bang controller and is designed to insure a vertical descent. The next section replicates these results with MATLAB's fmincon and then applies fmincon to the full three-dimensional problem. For both problems in fmincon the descent is discretized into N segments but the time of each step is also a decision variable. In the three-dimensional problem both thrust angle and throttle are decision variables. This allows an easy computation of fuel-time optimal landings. The two-dimensional result matches the analytical solution exactly.

The last section applies the YALMIP package to the 3D problem. Convex solvers and branch and bound solvers are employed and the results compared. Different constraints are applied and their effect on the trajectories are demonstrated."

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PETbox: Flight Qualified Tools for Atmospheric Flight

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The Planetary Entry Toolbox (PETbox) is a set of multiple modules developed by DEIMOS Space S.L.U. to support Mission Engineering and Flight Mechanics in the area of Atmospheric Flight.

PETbox has been intensively and successfully used in multiple ESA projects, EU projects and private initiatives covering a very wide range of vehicles (launchers, lifting bodies, capsules, UAVs, winged bodies, hypersonic transport vehicles, space debris…) in multiple environments (Earth, Mars, Titan) and in multiple flight phases (launch, coasting, entry, descent, landing, sustained flight). The set of modules that composes PETbox allows a critical range of multiple analyses, a full "Mission Engineering process" that supports engineers at different levels, from Pre-Phase A studies to Post Flight Analyses. The core module of PETbox is endosim (endoatmospheric simulator) which is the simulation framework used by the Atmospheric Flight Competence Center (AFCC) of DEIMOS Space. The toolbox is live and continuously evolving according to the improvements and modifications implemented daily and that currently integrates more than 50 years of engineering work of the AFCC team of DEIMOS Space.

The applications range is wide, covering vehicle design (shape design, configuration design, system specifications, MDO…), aerothermodynamics (computations, inspection, analyses, support to databases refinements…), flying qualities (trim, stability, controllability, GNC specifications, …), trajectories (modeling, end to end simulation and optimization, analyses, flight predictions…),

guidance (design, prototypes, functional validation, . . .), sizing conditions (performance and margins verification, specifications for system and subsystems, correlations analyses, . . .), safety aspects (nominal and off-nominal footprints, survivability and risk analysis of debris, separation analyses. . .), visibility aspects (with fixed or mobile ground stations, with GPS, between spacecrafts. . .), post flight analyses (trajectory reconstruction, data fusion, analyses, . . .), etc.
Practical example of use and key applications in multiple projects will be presented with special emphasis on the use of PETbox in the current ExoMars program (2016 and 2018 missions) and in the recent Intermediate eXperimental Vehicle (IXV) that successfully flew on February 11th, 2015. IXV has represented a unique opportunity to increase the TRL level not only of re-entry technologies but it also marked a key milestone in the overall validation of the design methodology and tools implemented in the areas of Mission Analysis and Flight Mechanics; it confirmed the robustness of the approach and the maturity of PETbox which is now Flight Qualified and ready for future challenges in the European re-entry technology roadmap.

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Real-Time Atmospheric Entry Trajectory Generation Using Parametric Sensitivities

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The real-time generation of optimal trajectories and controls for nonlinear systems is a technology of interest to many applications. But the online solution of an optimal control problem (OCP) is often not computationally feasible on embedded systems. We present a method to generate a near-optimal control sequence and the corresponding state trajectory based on the parametric sensitivity analysis (PSA) of nonlinear programs (NLPs) which does not require performing the classical gradient based NLP process online and hence reduces the computational load. The OCP is transcribed into a parametric NLP which is solved offline for a nominal set of parameters. Additionally the parametric sensitivities of the optimal solution with respect to different types of perturbations are computed at discrete points along the nominal trajectory. The sensitivities are used online in a Taylor expansion of the nominal solution and an iterative feasibility and optimality restoration to compute a new near-optimal control sequence and trajectory from the disturbed state to the terminal set without resolving a disturbed instance of the original NLP. This process is repeated successively in the neighborhood of the nominal trajectory. The proposed method is demonstrated for the guided, hypersonic entry of a small capsule into the Martian atmosphere. The PSA algorithm is used as feed forward command and trajectory generation to provide the input for a drag-energy tracking controller.

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IRENA - International Re-Entry demoNstrator Action

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IRENA is an action performed by an international consortium aiming at defining two technology demonstrator projects to validate advanced entry/re-entry technologies for further space exploration missions. In addition, IRENA shall create the ground for the implementation of these two projects in an international framework, such as promotion to ISECG, or next step in Horizon 2020 program. Technically speaking the consortium, which consists of European major industry, CNES, DLR, JAXA and NASA, has identified the EDL technologies requiring increase of the TRL level in near future, and identified possible flight and ground demonstration missions to

cover the technological gap. In order to satisfy the needs of all involved parties, the selected demonstrator concepts focus on deployable decelerators, aerocapture and TPS ground validation aspects. Series of dedicated workshops have allowed to bring the demonstrator concepts to a completion of phase 0 development stage, and also define their implementation plan which is work still in progress. The presentation/paper will provide an overview of the IRENA project, workflow and the identified technology demonstrators proposed for future implementation. This action has received funding from the European Union's Horizon 2020 research and innovation programme, under grant agreement No 640277, within the strategic objective COMPET-2014. It has started in January 2015, and is planned to finish in April 2016.

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IXV GNC verification from inspection to flight demonstration

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The Intermediate eXperimental Vehicle (IXV) is an ESA re-entry lifting body demonstrator built to verify in-flight the performance of critical re-entry technologies. The IXV was launched on February the 11th, 2015, aboard Europe's Vega launcher. The IXV´s flight and successful recovery represents a major step forward with respect to previous European re-entry experience with the Atmospheric Re-entry Demonstrator (ARD), flown in October 1998. The increased in-flight manoeuvrability achieved from the lifting body solution permitted the verification of technologies over a wider re-entry corridor. Among other objectives, which included the characterization of the re-entry environment through a variety of sensors, special attention was paid to GNC aspects, including the guidance algorithms for the unique lifting body, the use of the inertial measurement unit measurements with GPS updates for navigation, and the flight control by means of aerodynamic flaps and reaction control thrusters. From a wider perspective, the development chain for the GNC starts from the shape conception, which implements the control authority needed during orbit and atmospheric flight, up to the production of the flight software which implements the GNC design. In IXV, the design and verification of the GNC has followed an ECSS based approach in which analysis (ex: Monte Carlo simulations) and test (processor and hardware in the loop) have been the main elements of validation to deliver a Qualified product and to verify the GNC for the last set of parameters before flight. The successful flight of IXV has constituted the final verification of the GNC and hence a significant milestone for Europe: the ARD flight demonstrated the GNC for a capsule and the flight of IXV has verified the GNC for a lifting body using active flaps. The flight constitutes not only the final validation of the GNC but also a valuable source for verification and tuning of methods and tools. Several steps in the postflight analysis are foreseen which incrementally will exploit the flight performance using different tools and techniques. An initial postflight analysis has been conducted using several inspection, simulation and reconstruction techniques. This paper describes on one side the overall design, verification and missionisation process for IXV with emphasis in the tools and techniques that have been applied, which include either engineering tools to formal analysis tools like the Functional Engineering Simulator (FES) or the Real Time Test _Bench (RTTB). On the other, the techniques used to derive the initial in-flight verification of the GNC will be presented as well as the main results and conclusions.

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Low Thrust Trajectory Design and Optimization: Case Study of a Lunar CubeSat Mission

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The NASA CubeQuest Challenge offers a launch on the 2018 Exploration Mission 1 (EM1) to competing teams who can demonstrate a high probability of success for a 6U CubeSat design. The selected competitors will have their CubeSat disposed into a high-energy lunar flyby trajectory soon after EM1 launch. If the CubeSat remains ballistic, the lunar flyby will eject it from the Earth-Moon system, reaching a distance of 4 million km in 2 months. The trajectory design goal of the Challenge is a thrust profile that will allow the CubeSat to enter a stable lunar orbit (with prescribed orbit size bounds), and maintain that orbit for 28 days, all within 1 year of disposal. Due to EM1 safety considerations, high-thrust chemical propulsion is not allowed, which eliminates the possibility of direct lunar orbit injection during the lunar flyby. This limits competitor teams to using low-thrust propulsion and designing trajectories that take advantage of the natural Earth-Moon-Sun system dynamics.

In this paper, we examine the incremental and iterative process of designing and optimizing a low-thrust trajectory that achieves and maintains lunar orbit within the prescribed 1 year. We start by exploring the post-disposal dynamics of the Earth-Moon-Sun system, and determine the general transfer trajectory types that remain in the vicinity of the Earth-Moon system. Next, we design a transfer trajectory using multiple unconstrained impulsive maneuvers, and look for minimum Delta-V solutions where the individual Delta-V magnitudes would be attainable in realistic time with thrust levels on the order of 1 mN. Finally, we convert the impulsive maneuvers to finite-burn maneuvers and re-optimize to maintain a continuous trajectory while minimizing total thrust duration. In doing so, we incorporate the need to perform periodic tracking and orbit determination as a constraint on the maximum continuous thrust time, and discuss the impact of this constraint on the optimal trajectory.

This is a particularly difficult trajectory design problem due to a highly nonlinear dynamical system (Earth-Moon-Sun) and a milli-newton thrust propulsion system. We discuss the use of Earth-Sun L1 dynamics to transition from an Earth-departing trajectory to a lunar arrival trajectory. We use a multiple-shooting method to match these trajectories, and show how this greatly aids in optimization convergence. We also discuss the benefits of using a stable lunar distant retrograde orbit (DROs) as an intermediate target before spiraling the CubeSat down into its final low lunar orbit. Finally, we address the design of the lunar spiral-down phase and final lunar orbit itself, including orbit stability and visibility from ground stations.

We use the NASA GSFC General Mission Analysis Tool (GMAT), combined with the VF13ad optimizer, for trajectory design and optimization. GMAT allows us to set up the trajectory design problem using impulsive propulsion, then easily transfer that solution to finite-burn propulsion. Additionally, the ease with which the VF13ad optimizer can be used from GMAT allows us to transition from a feasible to an optimal solution with minimal changes to the problem setup.

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Low-Thrust Transfers from Distant Retrograde Orbits to L2 Halo Orbits in the Earth-Moon System

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This paper presents a study of transfers between distant retrograde orbits (DRO's) and L2 halo orbits in the Earth-Moon system that could be flown by a spacecraft with solar electric propulsion (SEP). Pseudo-spectral optimal control is used to optimize these highly non-linear transfers. Similar types of transfers that have been studied in the literature include: from Earth orbit to Moon orbit using low-thrust1, from Earth orbit to libration point orbits using low-thrust1, from Earth to DRO using impulsive maneuvers1, from Earth to DRO using low-thrust1, and from L1 halo orbit to L2 halo orbit. Transfers between DRO's and halo orbits using low-thrust propulsion have not been studied previously.

This paper takes advantage of modern advancements in computer hardware and optimization software to perform a study of the trajectories that could be flown by a low-thrust, SEP spacecraft

from a DRO to a halo orbit about L2. This includes fleshing out several families of transfers that exist, identifying the minimum thrust-to-weight ratio required to fly on each family, and providing an understanding of the time-of-flight vs. propellant-mass. Topputo[5] showed that many distinct families of ballistic transfers exist between the Earth and Moon in a four-body model, and others have demonstrated that such variations exist for other types of transfers in Earth-Moon space[6], [7]. By exploring the families of transfers that exist between DRO's and L2 halo orbits, this paper provides deeper insights into the trade space available. The circular restricted 3-body problem (CRTBP) is used throughout the paper so that the results are autonomous and simpler to understand.

DRO's are a type of orbit that have received increased attention in the past few years because of the unique characteristics they exhibit. DRO's are a type of repeating orbit that exists only in the 3-body problem[8]. When viewed in a reference frame that rotates with the orbits of the primary and secondary bodies, a DRO is retrograde about the secondary body, at a relatively high altitude such that the orbit is significantly perturbed by both the primary and secondary bodies. DRO's are unique in that they sit between two-body orbits and libration point orbits in terms of stability. These orbits are often dynamically stable, though it has been shown that perturbations in a high-fidelity model of the solar system may cause a spacecraft to depart an otherwise stable DRO[9]. Parker, Bezrouk, and Davis demonstrated several trajectories that transfer from Earth to a DRO, requiring no maneuvers and remaining on the DRO for thousands of years3.

Mission concepts that have examined DRO's include the proposed NASA/JPL ARM (Asteroid Redirect Mission)[10] and the Orion/MoonRise concept[11], [12]. Both of these mission concepts would benefit from the capability to transfer between a DRO and a halo orbit about L2. Ongoing research by Davis and Parker is finding that impulsive transfers between those orbits do exist, but they are costly on the order of 150 m/s and require transfer times on the order of weeks to months. Spacecraft with SEP have the potential to greatly reduce the propellant mass required to make such transfers, without much increase in time of flight.

The open source, pseudo-spectral optimal control package PSOPT[13] is used to optimize transfers in the CRTBP. A variety of families of solutions are discovered by seeding different initial guesses, and by then using the continuation method to discover similar transfers. An example transfer optimized using PSOPT is presented in Fig. 1, below. The new software package Maverick[14], developed at CU Boulder, is also used to compare solutions.

[*See attachment for Figure*]. An example transfer viewed in the Earth-Moon synodic reference frame. Dynamics are the CRTBP. The Moon is plotted to scale on the x-axis at (1-μ), and the Earth would appear at (-μ). This transfer has a thrust-to-mass ratio of 1.3E-4 N/kg and a transfer time of 47 days. Thrust is nearly always on for this example. 1.75% of the initial mass is used as propellant. Thrust vectors are plotted only in the bottom-right plot.

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Optimization of low thrust multi-revolution orbital transfers using the method of dual numbers

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The possibility of using the method of dual numbers in automatic differentiation for solving optimization problems of the low-thrust multi-revolution orbital transfers is considered. Traditionally the motion equations for the spacecraft with electric propulsion for multi-revolution orbital transfers are written in osculating elements or their modifications which exclude the special features in the right-hand sides of differential equations. These right-hand sides of the equations become especially complicated when different perturbations influencing the spacecraft movement are taken into account. Within the formalism of the Pontryagin maximum principle the right-hand sides of the optimal motion equations for the adjoints equations are quite complicated which results in some difficulties in solving optimization problems. Therefore the use of dual numbers method in numerical differentiation of optimal Hamiltonian for calculating the right-hand sides of the optimal motion equations of the spacecraft is effective. Another aspect of using the dual numbers method for numerical differentiation is to calculate the sensitivity matrix when solving boundary value problem corresponding to the optimal control problem. In this case, using dual numbers method allows obtaining the accurate sensitivity matrix. When using the continuation method for solving boundary value problem it helps to improve the convergence and to significantly reduce the number of steps for the external integration of Cauchy problem. The numerical results for optimal multi-revolution orbital transfer from the arbitrary initial orbit into the geostationary orbit are presented.

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Low-thrust trajectory design with fully analytic three-dimensional spirals

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We present a preliminary low-thrust mission design technique based on the family of generalized logarithmic spirals. This family is an exact and fully analytic solution to a specific tangential thrust profile, for which even the time of flight can be solved in closed-form. The method includes a control parameter that can be adjusted at convenience and the motion is three-dimensional. Advantages of the method are its speed, intuitive use, and the fact that it can reproduce trajectories generated with more sophisticated methods. Thanks to the conservation of two integrals of motion the versatility of the algorithm is improved significantly by introducing coast arcs, decomposing the trajectory in intermediate nodes, and the use of gravity assist maneuvers. Combining these

techniques yields a flexible method for which all the computational load reduces to solving two equations with two unknowns. Examples of interplanetary orbit transfers are presented.

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Advanced Electric Orbit-Raising Optimization and Analysis with LOTOS 2

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Telecommunication satellites located in the Geostationary Equatorial Orbit (GEO) are typically not directly placed there by the launch vehicle. The satellites are often injected in a Geostationary Transfer Orbit (GTO) and then transferred to the GEO using their own onboard propulsion system. State of the art for the GTO to GEO transfer is still the chemical propulsion. Just recently few satellites transferred or are transferring to GEO using Electric Propulsion (EP), since it is very attractive to exploit their high specific impulse reducing the propellant mass of the orbit transfer. Since the total spacecraft mass is reduced this yields launch vehicle cost reductions. Further, Electric Orbit-Raising (EOR) is now available for most telecommunication satellite platforms or at least under development. But electric orbit-raising requires much more complex maneuver sequences than what is needed for pure chemical transfers. Since EP provides only small thrust magnitudes in comparison to chemical propulsion, the transfer lasts many months. A careful planning of the spacecraft attitude maneuvers is required in advance to fulfill this mission. In recent years, many software tools have been developed for the preliminary assessment of low-thrust orbit transfers. Unfortunately, most tools lack both maturity and accuracy necessary to fully exploit the capabilities of electric orbit-raising. For example, during the transfer any crossing of the GEO ring poses a certain collision risk with high value assets. Thus, the precomputed transfer trajectory has to avoid crossings of the GEO belt. Further, ground station visibility might be considered for transfer planning as well as limitations and constraints related to different spacecraft subsystems, such as eclipse handling, power generation, storage and consumption, or EP firing limitations in general. Other possible limitations are related to the attitude of the spacecraft or consider environmental aspects like the radiation dose. Using Non-Linear Programming (NLP) to optimize the attitude profile in combination with detailed modelling of complex mission constraints and limitations of the spacecraft model is essential, especially under consideration of tight accuracy and fidelity requirements for achieving optimality in sense of propellant consumption and transfer duration. Besides optimization, many aspects have to be analyzed in more details. It encompasses subsystem issues for example of the Attitude and Orbit Control System (AOCS) as well as station visibilities. In the newest version of the low-thrust optimization and analysis tool LOTOS (Low-thrust Orbit transfer Trajectory Optimization Software) all aforementioned features are available. But the tool is not only limited to electric orbit-raisings; it also supports hybrid transfers where chemical maneuvers are followed by the low-thrust transfer. Another feature of the software is the support of spacecraft operations. This mode identifies the spacecraft location on a pre-computed reference trajectory and uses its attitude profile for re-optimization. Such a processing is required due to deviations of the real-flown spacecraft trajectory from the nominal one. A full overview of the software capabilities and features will be given in this paper, such as hybrid transfers, 6 degrees of freedom attitude control and verification of trajectories. Highlights of the Graphical User Interface (GUI) will be presented as well. It includes automatic user defined analyses and customizable reports to support mission analysis engineers and to relieve them from repeating tasks. For example, the reports provide full access to all data of the user defined scenario and may include tables and plots.

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Many-Revolution Low-Thrust Orbit Transfer Computation using Equinoctial Q-Law Including J2 and Eclipse Effects

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Mission designers addressing the computation of low-thrust many-revolution transfers need versatile and reliable tools for solving the problem with efficient computational times. This paper proposes a Lyapunov feedback control method, Q-law by Petropoulos with algorithm modifications to accommodate for the singularities in the original equations and to include the most relevant perturbations, such as the J2 perturbation and the effect of coasting during eclipse periods. The optimization of the control-law parameters via a multi-objective evolutionary algorithm (NSGA-II) improves the results significantly and permits to easily compute the minimum time transfer and a well-spread Pareto front, trading transfer time versus propellant.

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Dynamic Test Facilities as Ultimate Ground Validation Step for Space Robotics and GNC Systems

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Ground testing of space technologies, and in particular space robotics and Guidance Navigation and Control (GNC) ones, is crucial in order to de-risk space missions that heavily rely on them. In this context, this paper describes the role of the platform-art© robotic facility located at GMV premises in Tres Cantos (Madrid, Spain) as last on-ground validation step within the development and validation of a whole range of different space technologies.
The platform-art© hardware in-the-loop facility can simulate the dynamics of any satellite system: the paper will present how the facility has been used in a number of activities related to a different space scenarios such as Lunar descent and landing, on-orbit servicing (including contact), active debris removal (e.g. ANDROID mission) and Rendez-Vous (e.g. Mars Sample Return capture mission). The modularity of the facility will also be described, which allows for fast re-adaptation of the setup for different scenarios, also enabled by the different mock-ups that are available at platform-art© (with different scale factors and fabrication precision depending on the specific test needs. Lessons learnt and graphical results from different ESA projects that have used the platform-art© hardware in-the-loop facility.
It will also be presented, complementary to the dynamic test facility, an integrated Design, Development, Verification and Validation environment covering from Model-in-the-Loop (e.g. Matlab/Simulink algo-rithms) till real-time Processor-in-the-Loop (with autocoded or hand-made GNC produced SW) and later on extended through the use of dynamic test facilities with real dynamic and air-to-air HW-in-the-Loop (sensors, robotic manipulators, ...) stimulation.

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Design and parameter identification by laboratory experiments of a prototype modular robotic arm for orbiting spacecraft applications

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This paper describes the design and the parameter identification procedure of a modular spacecraft robotic arm that combined with appropriate ancillary equipment (base-spacecraft, target and end-effector) provides an experimental set-up where control approaches and whole mission scenarios (e.g. servicing and debris removal) can be validated and demonstrated.

The originality of the here studied prototype robotic arm consists in its modularity. Each manipulator link contains its own power system, communications, harmonic drive motor with controller, torque sensor and a computing platform. The structure of the links has been manufactured using additive manufacturing allowing to quickly and inexpensively generate links of different lengths (with different mass and inertia properties). The manipulator links are modular and easy to re-arrange to meet the requirements of a particular experiments (number of links and length of these links).
A spacecraft equipped with a robotic arm is required to fulfill a wide range of missions (e.g servicing and debris removal) and few spacecraft with robotic arms have already been successfully flown. The dynamics of a space manipulator are substantially different from its terrestrial counterparts as the base-spacecraft is free to react to the manipulator motion. When the base-spacecraft mass and inertia are comparable to the manipulator's ones (i.e. as it is the case for small base-spacecraft) the base-spacecraft reaction can be significant and can not be safely omitted when modeling or treated as a small disturbance during control.
A substantial amount of theoretical and simulation work has been previously conducted by many researchers to tackle the operation and control of a space manipulator and its base-spacecraft. The difficulty to recreate the conditions of a space manipulator mounted on a small spacecraft on the ground limits the availability of validation experiments on control approaches and dynamical modeling.
A prototype modular robotic manipulator consisting of three links with three rotational degrees-of-freedom has been designed and integrated. Each link has a length of ~40 cm and has a mass of ~2 kg. This robotic manipulator is mounted onboard a ~10 kg Spacecraft Simulator that floats via air-pads over of a 4-by-4 meter granite monolith recreating in two dimensions the reduced gravity and quasi-friction-less environment of space. As it operates on a granite table the movement of both the manipulator and the base-spacecraft are restricted to two translational and one rotational degrees-of-freedom (planar movement). The base-spacecraft is equipped with eight cold-gas thruster and a reaction wheel and thus is able to control its position and attitude.
As the manipulator can be re-arranged in a different number of configurations, a method to quickly identify the manipulator parameters (mainly mass, inertia and the Denavit-Hartenberg parameters) will be presented. Finally, a few basic manipulator and base-spacecraft control capabilities of the experimental set-up are demonstrated.

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Rapid Deployment of Design Environment for EUCLID AOCS design

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Euclid is a cosmology mission dedicated to study the geometry and the nature of the Dark Universe with unprecedented accuracy. Euclid will observe a 15000 deg^2 wide area of the sky from the Lagrange point L2 of the Sun-Earth system. The scientific goals of the mission result in very demanding performances for the AOCS subsystem; as an example, the observations' Relative Pointing Error shall be kept within 75 mas over a time scale of 700 sec. The Euclid S/C is procured by ESA and supplied by TAS-I; SENER is the prime contractor of the AOCS sub-system, for which the work is executed in partnership with ADS-NL. Amongst other responsibilities, SENER is responsible for the design, implementation and verification of GNC/AOCS algorithms; in the frame of the activities for the Phase B2 of the study, a set of tools and methodologies were developed and employed for the design tasks undertaken. The core of the design effort resided in the preparation and set-up of the Euclid Design Simulator (EDS). Such simulator is based on the internal tool SENERIC, whose model library was used as core or starting point for the DKE,

actuators and sensors models required for Euclid. This allowed a fast deployment of environment for completing AOCS SRR in 1 month and AOCS PDR in 7 additional months. Along with the units and dynamics simulation, a simulation management architecture was developed. The main user for the EDS is the AOCS engineer, focusing on design tasks. As a consequence, the EDS simulation management infrastructure was developed taking into account the following usage guidelines: fast data preparation and processing, ability of storing and retrieving simulated cases, ability of running different models with the same source data set, ability of running Monte Carlo simulations, all without requiring lengthy and complex case set-up. The product of such effort is a simulator which allows launching test cases from MATLAB Environment initialization (leveraging on native types for data definition), as well as flexibly generating and retrieving XML databases equivalent to the MATLAB native data types. The EDS currently makes use of Simulink libraries and model referencing, and further efforts are envisaged towards the auto-coding of the AOCS/GNC algorithms being designed through the EDS. It is being used for developing models and specifications aiming at an ideally seamless integration within the formal verification environment, in first step via FES, and later for the RTS production to be applied in the AOCS SCOE. EDS is applied during the whole process as the flexible reference design tool.

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ROSPA, cross validation of the platform-art and ORBIT test facilities for contact dynamic scenario setup and study

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The set-up and use of ground validation testing facilities from the early phases of the missions can provide a very valuable feedback to the equipment and technologies being developed. The validation activities of the own ground testing facilities are key to the usability and confidence of the results obtained from them. GMV's platform-art dynamic test bench has already been validated (for navigation purposes based on optical cameras) with flight data coming from PRISMA mission through the PRISMA-HARVD experiment. In addition, platform-art dynamic test bench is currently being extended thanks to ESA loan of several new devices, including a new high-span KUKA robotic arm, which will extend the functionality of the current test bench. On the other hand, ESA has established an air-bearing facility known as ORBIT (Orbit Robotics Bench for Integrated Technology). The facility located within ESTEC's Automation and Robotic laboratories provides several air-bearing platforms which can move frictionless on a 45 m^2 flat floor.

The in-space Robotic Servicing Physical Assessment (ROSPA) is a study with the purpose of recreating and studying the dynamics during/after contact between target and chaser in a rendezvous and capture mission. The data of the experiments run in the platform-art (GMV) and in ORBIT (ESA) will be used for a cross-validation of the facilities.

Two different scenarios have been setup in both facilities: simple contact and gripping scenario. In the simple contact scenario the Mitsubishi PA10 robotic arm approaches the specifically designed mock-up mounted on the air-bearing (or on the KUKA robotic arm, in platform-art) and touches it through a compliance device and a load cell to measure the contact forces and torques. In the gripping scenario the compliance device is replaced by a gripping device, which after an open loop trajectory attempts the gripping of a Launch Adapter Ring (LAR) mock-up.

In the scope of the activity also another functionality of the platform-art facility is exploited: the space-like environment simulation. This functionality has been developed and demonstrated in the frames of previous collaborations between ESA and GMV, for projects such as NEOGNC (Interplanetary mission, MarcoPolo-R) and ANDROID (Active Debris removal mission). The output of this activity is a data base of representative images taken during an open loop sequence of the robotic arm approach and the gripping with the LAR of a mock-up of the TANGO spacecraft (the 2nd spacecraft of the Swedish PRISMA mission). The representativeness of the images in such a close range scenario is meant in terms of illumination conditions and disturbances recreated in laboratory like fuel on lens, micro pieces of MLI floating and thruster plume in the Camera FoV.

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Differential Algebra Space Toolbox for Nonlinear Uncertainty Propagation in Space Dynamics

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The problem of uncertainty propagation represents a crucial issue in spaceflight dynamics since all practical systems - from vehicle navigation to orbit determination or target tracking - involve nonlinearities of one kind or another. One topic of recent interest concerns for instance, the space surveillance and the accurate propagation of uncertainties on the initial conditions of resident space objects in order to identify and track them. Another relevant application consists of computing landing dispersion both for reentry missions and for estimating the casualty area of space debris. In addition, within the space mission design process, uncertainty propagation is a fundamental tool to assess the fulfillment of mission requirements and constraints, to evaluate mission performances, to perform sensitivity analyses, and to verify the robustness of guidance and control laws. However, most spaceflight mechanics problems involve nonlinearities. This observation finds proof even in basic applications. For instance, many problems in celestial mechanics require conversions between different coordinate systems (e.g. the conversion from polar to Cartesian coordinates that form the foundation for the observation models of many sensors). Such transformations are typically nonlinear and therefore entail the problem of applying nonlinear transformations to the estimated statistics.
Although in the recent past, the power of computing hardware has been growing exponentially, giving rise to many new possibilities in the field of numerical mathematics, the vast majority of numerical algorithms developed to tackle the problem of uncertainty propagation, are still largely based on the same pointwise or linear algebra (matrix) techniques developed 50-100 years ago. In particular, two main families of techniques exist: the first one is represented by standard linear methods, where a linear approximation is introduced and used to perform uncertainty analyses. Despite the advantages in terms of computational time, they are typically characterized by a low level of accuracy, since the linear approximation holds only in a restricted region of the problem domain. The second one is represented by standard Monte Carlo techniques, which are able to provide very accurate results, but require large computational times and memory burden. In light of the above, the scientific community has started focusing on the development of new tools, able to improve the approximation of standard linear methods available in the literature or to reduce the computational time required by standard Monte-Carlo simulations.
Differential algebra (DA) perfectly suits with these requirements, providing a method to easily extend the linearization methods and allowing the implementation of efficient arbitrary order methods. The resulting technique performs much faster and yields more accurate results in many mathematical, physical as well as engineering applications. Dinamica, with the support of ESA, has recently completed the implementation of the Differential Algebra Space Toolbox (DAST), a DA-based software tool for the efficient, nonlinear propagation of uncertainties in space dynamics. DAST is divided in three main layers: the Differential Algebra Computational Engine (DACE), the Software Framework (SF), and the Uncertainty Propagation Tool (UPT). DACE provides the user with the tools to perform all the basic DA operations by replacing the operations between single numbers by suitably chosen operations on polynomials. The same sequence of operations coded for floating point numbers can thusly be evaluated using this new meaning of each operator

almost without changes to the code. SF is built on DACE and comes with a set of more advanced features (e.g. vectors and matrices of DA, propagation schemes), specific astrodynamic routines (DA Kepler solver and DA Lambert solver) and ready-to-use dynamical models. Finally, UPT allows the user to propagate uncertainties through different techniques (from classical Monte Carlo to range estimation using polynomial bounders or classical linear covariance propagation) and perform statistical analyses.
Although the general mathematical foundation suggests potential applications of the same approach to various fields such as biology, automotive engineering, and finance, this work focuses on the application of DAST to a set of different test cases in the field of astrodynamics and space engineering, with the aim to illustrate the potentials with respect to classical approaches. Thanks to the use of DA techniques, indeed, DAST has been shown to be orders of magnitude more efficient than traditional methods for uncertainty propagation.

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Tool for Real-time Pradiction of IXV Trajectory in the Mission Control Center

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IXV (Intermediate eXperimental Vehicle) re-entry vehicle has successfully flown a fully autonomous mission from the launch until it's splashdown in the eastern Pacific Ocean on 11th February 2015, and has been regarded as a major milestone in European re-entry technology roadmap. The vehicle has been designed to fly autonomously without the commanding from ground and without a need for a continuous downlink. It's trajectory and attitude however had to be closely monitored in MCC for operational and safety purposes including when out of the visibility window. For this purpose a dedicated too has been developed - TPVT (Trajectory Propagation and Visualization Tool).
The tool updated the current state and trajectory predictions based on the IXV vital-layer telemetry data whenever it was in the direct visibility of any of the ground stations, and also taking into account latest atmospheric measurements by the sounding balloons. In order to represent the vehicle's behavior as close as possible, the same GNC algorithms have been implemented in the propagation algorithm. For sake of propagation velocity the utilized models had to be kept simple in order to allow regular update of the trajectory every couple of seconds whenever fresh telemetry data was available. The propagated trajectory was displayed for the MCC operators for monitoring purposes, and the current state vector transmitted to the naval ground station in specific antenna pointing data format in order to facilitate the acquisition of the signal on the naval station. At the end of the mission the TPVT had an essential role in the vehicle's localization by providing an updated expected splashdown position, where it was found by the recovery ship's crew just 20 minutes later.
This paper gives and overview of various features of the tool, its interfaces with the MCC, visualization features, and also an assessment of its propagation performance during the real mission.

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Asteroid proximity GNC assessment through High-fidelity Asteroid Deflection Evaluation Software (HADES)

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This work presents the general architecture and capabilities of the HADES software developed at Deimos Space SLU. Detailed models about the close proximity environment about Near Earth asteroids and the involved operations are often required during preliminary assessment of mission requirements especially under the presence of uncertainties. It is vital to assess the compliance to the mission requirements in terms of safety and illumination. These play an important role in the selection of control techniques and operational orbits. The developed software deals with the high-fidelity modelling of spacecraft operations at irregular shape asteroids. The first version of HADES which includes the main GNC functionalities has been developed and tested. The spacecraft dynamics considers all the possible perturbations, i.e. third body effect from the Sun, the SRP and irregular gravity field of the rotating asteroid. The software uses both spherical harmonics and actual asteroid's shapes. In the first case the coefficients can be given from actual data or they are calculated on the size of user-defined ellipsoid; in the second case the gravity field is reconstructed from the asteroid tetrahedral mesh. The software can handle any operational orbit, with particular care paid to inertial and body fixed hovering. One important aspect when designing proximity operations is to evaluate how the different control techniques and on-board instruments affect the performance of the system. Different control techniques based on both continuous and discrete methods have been considered and implemented. The manoeuvre execution itself can be affected by errors in the magnitude and in direction. The spacecraft orbit determination is performed through a performance models or by on-board measurements, a navigation camera and a LIDAR, which are processed by an Unscented H-infinity Filter (UHF). The latter was selected for its ability to deal with unfiltered biases and non-Gaussian distribution of the measurements. The visibility and illumination condition are considered for the image processing, with the measurements affected by the attitude and pointing errors. HADES can employ different levels of accuracy between the assumed environment knowledge and the model used in the controller and in the UHF. For instance the gravity could be modelled from the shape, but the correction manoeuvre and the trajectory estimate could be calculated by a limited number of spherical harmonics. Also the shape model could be known with a certain error. HADES comes with a Monte Carlo (MC) module which allows drawing more noticeable statistical parameters, such as the control budget, accuracy of the estimation and control systems, or the occurrence of failures when the controller cannot maintain the orbit. Different examples applied to the case of the asteroid Dydimos will be shown for inertial and body-fixed hovering, with some MC analyses for the station keeping of the Asteroid Impact Mission (AIM) for different levels of system and dynamics uncertainties. Finally the ongoing activity on the asteroid deflection by low push methods, i.e. laser ablation, ion-beam shepherd and gravity tractor will be shown. The present tool was developed within the European Commission funded Stardust project under the Marie Curie scheme.

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Dynamical analysis of rendezvous and docking with very large space infrastructures in non-Keplerian orbits

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The idea of building a space station in the vicinity of the Moon as a gateway for future human exploration of the Solar System is being investigated since many years, and recently the attention of scientific community has become even more intense. The natural location for a space system of this kind is about one of the Earth-Moon libration points, in particular EML1 or EML2. Therefore, the entire analysis has to take into account the non-Keplerian dynamics that regulates the orbital motion in these environments.

At the current level of study, the final configuration of the entire system is still to be defined. However, it is already clear that in order to assemble the structure several rendezvous and docking activities will be carried out, many of which to be completely automated. In addition, numerous proximity manoeuvres will be held along all the nominal lifetime of the space infrastructure. In

non-Keplerian orbits the motion is completely different from the one in LEO, since the effects of three-body problem are not negligible, and various problems still deserve particular attention despite the knowledge about autonomous approaching operations in space, acquired with the experience of the International Space Station.
In this paper, the dynamics of very large space structures in non-Keplerian orbits is analysed, taking into account the flexibility of the system and the coupling effects between the modes of the structure and those related with the orbital motion. The results are then exploited to have a sensitivity analysis about the different families of non-Keplerian orbits as a function of the possible configurations of the space station with respect to the numerous rendezvous and docking manoeuvres to be facilitated: the dynamical stability of the system can be evaluated and assessed with respect to safe rendezvous and docking manoeuvres design between robotic vehicles and orbiting infrastructure.
A Multi-Body approach is here preferred, but the flexibility of the system is included both with a lumped parameters and with a distributed parameters technique. The results obtained with the two different approaches are compared, analysing the precision of the results and the computational time that is required to perform the computations. The configuration and the parameters of the large space structures are fully parametrized and the model is maintained as generic as possible, in a way to delineate a global scenario of the mission. However, the developed model can be tuned and updated according to the information that will be available in the future, when the system will be defined with a higher level of precision.
The results are critically presented with respect to the proximity manoeuvring complexity and required resource budgets for some reference scenarios.

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Chattering-free Sliding Mode Control for Propellantless Rendez-vous using Differential Drag

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Owing to the well-established correlation between spacecraft mass and mission's cost, there is great interest in fuel-optimal relative maneuvers between two or more satellites in the literature. In this context, the exploitation of natural perturbations is an attractive means to reduce or even remove fuel consumption, and, hence, propellantless maneuvers using solar radiation pressure, geomagnetic field, Coulomb forces, and atmospheric drag were proposed. Amongst them, the idea to use differential drag as control force for relative motion is particularly attractive to enhance the maneuverability of small satellites in low-Earth orbit, so that ongoing and forthcoming missions envisage this technique to achieve propellantless rendez-vous, cluster keeping, or constellation deployment, e.g., QARMAN, SAMSON, and Flock, respectively.
Because of several assumptions and modeling limitations, however, severe uncertainties affect satellite drag estimation. In order to effectively compensate for these uncertainties, successful attempts that include linear quadratic regulators, nonlinear adaptive control, model predictive control, and sliding mode control (SMC) were applied. SMC is widely adopted to cope with uncertainties due to its high robustness, however, it has two major drawbacks when applied to real-life problems:

- It usually causes **chattering**, i.e., high-frequency oscillations in the control force. When attitude control is used to tune differential drag, chattering can degrade the control performance and even jeopardize the maneuver if attitude actuators are saturated, e.g., reaction wheels.
- Prior knowledge of the **uncertainty bounds** is required. This is in general difficult to accurately estimate especially for time-varying drag forces.

In this paper a new chattering-free sliding mode controller is developed to perform an optimal rendez-vous of two satellites in low-Earth orbit, exploiting differential drag as control force. The proposed controller is designed to successfully compensate for uncertainty effects and/or unmodeled dynamics associated with air drag and to account for practical limitations such as input saturation. **Continuity of the control functions embedded in the new controller guarantees no chattering.** It is also shown that **an accurate estimation of the uncertainty bounds is not necessary.** Specifically, selecting conservative bounds leads to smaller errors, while the required control effort is not very sensitive to these bounds. Numerical examples are used to validate the robustness and accuracy of the new chattering-free sliding mode controller.

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GNC design and validation for rendezvous, detumbling, and de-orbiting of ENVISAT using clamping mechanism

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This paper describes the development and validation of a GNC for active debris removal of the ENVISAT S/C, using clamping mechanism. The design of the GNC is focused on the phases where the choice of clamp mechanism is of particular relevance to the ADR execution. These mission phases include the pre-capture, close range rendezvous with uncooperative target, the post-capture detumbling and stabilization of the composite, and the composite de-orbiting. The derived GNC solutions are based on robust MIMO control, which is especially suited to the considered mission scenario. Namely, the GNC is designed considering S/C dynamics with multiple-bodies attached by clamping mechanisms, with flexible appendages and sloshing effects, and with uncertainties in the mass, centre-of-mass and inertia (MCI) parameters, among others. The results of the GNC design activities are presented, namely in what respects to: the multi-body modelling of the chaser and composite S/C, the LFT modelling of the plant dynamics for GNC synthesis, the GNC trade-offs, the resulting GNC architecture and modes, and, finally, the methodology and results adopted for the Guidance and Control subsystems design. An advanced verification and validation framework for GNC was also developed, and is presented therein, that includes analytical validation of GNC robustness, stemming from the mu -synthesis framework. In addition, numerical validation is performed in a high-fidelity simulator of S/C dynamics, including multi-body dynamics, flexible modes, sloshing, MCI perturbation, environmental disturbances, among others. The validation results are obtained for the nominal scenario, as well as for alternative definitions of ENVISAT rotational motions and of relative sensor technology. Finally, Monte-Carlo campaign is adopted in complement to the analytical validation, allowing for the consolidated derivation of a set of requirements and recommendations for future missions, especially those within the Cleanspace initiative.

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ATHENA: Astrodynamics Toolbox for High-fidelity Error-propagation and Navigation

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This work presents the general architecture and capabilities of the ATHENA software package. ATHENA is a toolbox for uncertainty propagation, gudance, navigation and control of single and multiple spacecraft with distributed architecture. ATHENA implements advanced state estimation and filtering techniques and a high fidelity dynamical model. ATHENA allows for the analysis of the coupled orbital and attitude dynamics during close proximity operations and docking with cooperative and non cooperative targets.
The toolbox implements high fidelity models for both asteroids and Earth orbiting objects. The dynamic model includes third body effects, solar radiation pressure and the irregular gravity field of Earth, Moon and asteroids. The toolbox includes spherical harmonics gravity models, tetrahedral models from radar observations, and finite volume models. In the first case, the harmonic coefficients are from actual data or are calculated from the inertia matrix of a user-defined ellipsoid; in the last case, the gravity field is reconstructed from point masses. Given an initial distribution of point masses of arbitrary value contained within the original shape, their position and mass is optimised in order to match the original centre of mass and moments of inertia.
ATHENA allows for the design and analysis of multi-sensor fault tollerant autonomous navigation systems with sensor fusion across multiple platforms with resource sharing. The paper will present an example of autonomous navigation of a formation of spacecraft flying in the proximity of an asteroid. Multiple measurements collected by on-board cameras, attitude sensors and LIDAR are data fused to estimate the state of each spacecraft with respect to the asteroid. Inter-spacecraft links are used to combine measurements and improve resilience and accuracy. It will be shown that different combinations of measurements can be constructed to improve the navigation performance. The overall effect is a system more robust in the presence of failures.
ATHENA includes models of binary systems and the coupled orbital and rotational motion of the two bodies around their centre of mass. The paper will present and example of navigation and control of a single spacecraft placed at a stable libration point.
The toolbox can simulate close proximity operations and autonomous rendezvous and docking (R\&D) with non-coopeative targets on elliptical orbits. In particular, ATHENA can generate optimal scheduled plans and control profiles to rendezvous with multiple targets or to dock multiple spacecraft with a single target. This capability will be shown within an orbit servicing scenario. In this demonstration scenario, the target spacecraft is formed by a set of active payload modules (APM) connected to the spacecraft bus via a standardized interface. ATHENA generates a plan and corresponding guidance for a swarm of small satellites that collaborative (as a colony) re-configure the target spacecraft by removing and replacing APMs.

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GNCDE as DD&VV environment for ADR missions GNC

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GNCDE is an integrated GNC development and verification environment, developed by GMV in the frame of an ESA-GMV co-funded activity. It contains templates of four different adaptable scenarios (Rendezvous and Docking, 3-axis stabilization, Formation Flying, Launchers), a complete set of libraries for sensors, actuators, DKE and GNC blocks, and a set of tools to ease the design and analysis of the mission (e.g. guidance trajectories design, control and estimation synthesis, covariance analysis, Monte Carlo campaign, Statistical analysis, Autocoding, 3D visualization through direct connection with tools like Celestia, etc.). GNCDE has been already successfully used to design the GNC of different rendezvous missions (such as Advanced Re-entry Vehicle and Mars Sample Return Orbiter) and it is the current development environment for the formation flying software of PROBA-3 phase CDE.
This paper will focus on the utilization of GNCDE for assessing GNC concepts of two different ADR scenarios, both aimed at the post-life disposal of ENVISAT:
1) Design, development, verification and validation of the GNC for RDV and de-orbiting phases of E-Deorbit mission, currently in phase B1. E-Deorbit is so far the most advanced ESA activity

with the objective of de-orbiting ENVISAT. It is unique in its operational complexity and requires a high reliable and strongly validated GNC design.
2) Quick preliminary feasibility evaluation from GNC point of view of PRIDE vehicle used as active debris removal spacecraft. PRIDE is the ESA program aimed at developing a reusable robotic spacecraft with different in-orbit servicing capabilities, among which the possibility to serve as an ADR vehicle.
The high flexibility of GNCDE has permitted to adapt very quickly the rendezvous and docking template (originally used for an ATV-ISS docking scenario) to the two different ADR scenarios, parametrizing it opportunely to include configuration, initial orbital and attitude data of both ENVISAT and the chaser spacecraft. Sensors and actuators parameters have been also modified to take into account the typical accuracies and errors in the two cases. The rendezvous trajectories have been tailored to these scenarios and the GNC laws adapted to their specific needs.
In the case of E-Deorbit, the work to be done has a long schedule aiming at a fully validated GNC and the design is still on-going. The paper will present the process which is being followed for GNC DD&VV of this specific scenario, how this process is supported by the GNCDE environment and the available preliminary results. In the case of PRIDE scenario, the study preliminary indicates that the vehicle could be suitable for an ENVISAT ADR mission. Using the link between GNCDE and Celestia, a video showing the capture phase, including synchronization between PRIDE and ENVISAT, has been also set up.