ESOC'S SYSTEM FOR INTERPLANETARY ORBIT DETERMINATION

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ABSTRACT

A system for interplanetary orbit determination has been developed at ESOC over the past six years. Today, the system is in place and has been proven to be both reliable and robust by successfully supporting critical operations of ESA's interplanetary spacecraft Rosetta, Mars Express, and SMART-1. To reach this stage a long and challenging way had to be travelled. This paper gives a digest about the journey from the development and testing to the operational use of ESOC's new interplanetary orbit determination system. It presents the capabilities and reflects experiences gained from the performed tests and tracking campaigns.

1. INTRODUCTION

More than half a decade ago ESOC started to set up a system for interplanetary orbit determination. The starting point was the orbit determination program that had been used for Giotto, ESA's first interplanetary mission, in the 1980s. The mathematical formulation of the Orbit Determination Program (ODP) from NASA/JPL served as the main reference document [1,2]. The hurdles, amongst others, that had to be overcome, were the following:

- To define and implement a suitable software design that allows support of various kind of interplanetary spacecraft and mission phases and that gives flexibility in the treatment of uncertain parameters;
- To ensure that the mathematical algorithms and their implementation were correct;
- To cope with tracking data originating from a completely new tracking system that has been installed in most of the ESA network of ground stations and especially for the newly built 35 m deep space antenna (DSA) in New Norcia (NNO);
- To be able to process conventional tracking data, Doppler and range, as well as delta differential one-way range (ΔDOR) acquired at NASA/DSN ground stations.

In order to get absolute confidence that all these novelties were implemented correctly, several tests have been performed in addition to the normal internal quality assurance tests within ESOC. These were done with tremendous support of the Navigation Section at NASA/JPL and included the following:

- Cross verification tests (CVT) against the ODP to ensure the correctness of the mathematical algorithms;
- Tracking campaigns using the ESA spacecraft Ulysses (which is operated from JPL), to validate the capabilities of the DSA and tracking system at NNO and of the orbit determination system when applied to a spacecraft in interplanetary cruise;
- Tracking campaign using Mars Express during interplanetary cruise to ensure the correct processing of ΔDOR tracking data that were essential for the navigation at Mars approach.

Fig. 1 gives an overview of the time frame in which testing and operations have taken place. The highest workload accumulated in 2003.

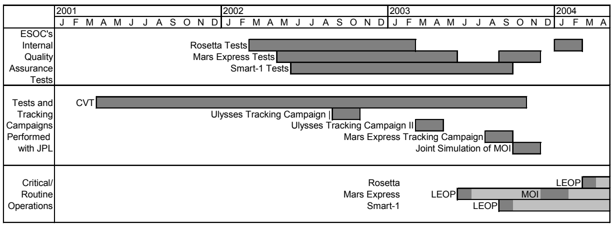

Fig. 1: Timeline of tests and operations

2. SET-UP AND CAPABILITIES

This section summarises the capabilities of the orbit determination system. It does not go deeply into technical details but rather tries to give a comprehensive overview.

2.1 Estimation

The orbit determination software uses as statistical estimation method a square root information filter (SRIF). The more classical and well-known estimation method is weighted least squares and, in fact, the SRIF is mathematically exactly equivalent to weighted least squares but it is numerically superior. The SRIF is based on a so-called epoch-state formulation, i.e. the components of the spacecraft state vector are estimated at a fixed epoch. The spacecraft state can be augmented by other parameters that are treated as "solve-for", i.e. they are updated within the estimation process. Furthermore, a subset of uncertain parameters can be selected that are treated as "consider" parameters. Those are not estimated but contribute to the formal uncertainties in the solve-for parameters. Solve-for and consider parameters can be freely chosen by the user out of a given set and there is principally no limit on the number of these. This user-friendly freedom is allowed by a sophisticated parameter-bookkeeping system, that keeps track of the parameter list throughout the entire orbit determination process.

The SRIF tries to adjust the solve-for parameters in a way so that it minimises the sum of squares of the weighted residuals, including taking *a priori* information into account. A weighted residual is the difference between the actual measurement and the computed measurement, both divided by the measurement standard deviation. Computed measurements are calculated from the values of the solve-for parameters using mathematical models of the dynamic and measurement process. Since the measurements are a non-linear function of the solve-for parameters, a differential correction method is applied. This requires a linearisation of the system, so that actually the increments to *a priori* values of the solve-for parameters are estimated. An iterative scheme is then applied so that all solve-for parameters are differentially incremented until convergence is achieved.

The SRIF allows incorporating process noise (coloured noise) in the modelling of the dynamics or measurements by using exponentially correlated random variables (ECRVs). These variables are treated as piece-wise constant over time intervals that are small compared with their auto-correlation times and can be solved-for.

2.2 Dynamic Modelling

Starting with an *a priori* spacecraft state and a particular dynamical model with *a priori* values of the dynamical solve-for parameters, the trajectory of the spacecraft is propagated by integrating the equations of motion. The numerical integration method used is a numerical scheme attributed to Nordsieck [3]. Nordsieck's method is a multi-value, variable step size algorithm and is known to be numerically very stable. Due to its multi-value nature it is easy to use in combination with a sophisticated step-size control algorithm. Simultaneously, the variational equations are integrated using the same scheme. They provide the partial derivatives of the spacecraft state with respect to all dynamic parameters that are treated as uncertain, in particular the components of the spacecraft state at epoch.

The dynamical model, i.e. the right hand side of the equations of motion and variational equations, is implemented independently from the integrator. So, it can easily be changed without the necessity to touch the actual integration scheme. In fact, we use the same integration software but different dynamical models for Rosetta, Mars-Express, and SMART-1, each of them being suitably chosen for the particular mission. The pool of dynamical models, that is currently implemented and can be chosen from, is as follows:

- Central gravitational potential acceleration of the Sun, the nine planets, the Earth and Martian moons, and the big three asteroids Ceres, Pallas and Vesta;
- The relativistic perturbative acceleration of principally the same solar system bodies but just the Sun is normally used;
- Acceleration due to solar radiation pressure;
- Acceleration due to the gravitational harmonics of Venus, the Earth, the Moon or Mars;
- Acceleration due to a motor burn, either treated as impulsive or finite duration, and either with respect to the spacecraft body or inertial frame;
- Acceleration due to air drag using a simple generic atmosphere or a special Martian atmosphere (Marsgram 2001)

Each of the dynamical models has one or more pre-defined parameters that can be treated as uncertain.

2.3 Measurement Modelling

The core orbit determination software is preceded by the tracking data pre-processor. It reads the raw data files that are retrieved from the tracking systems of a particular ground station and writes the measurements along with auxiliary information related to them into an observation file that is suitably formatted for the orbit determination process. The software can read either tracking data files originating from ESA's old tracking system, the Multi-Purpose Tracking System (MPTS), or ESA's new tracking system, the Intermediate Frequency Modem System (IFMS). The latter is currently being implemented in most of ESA's ground station network and it is the intention to replace the MPTS entirely in the near future. Furthermore, the pre-processor is capable of processing tracking data acquired at NASA/DSN ground stations.

The orbit determination software can currently process the following measurement types:

- Two-way range from ESA and DSN ground stations;
- Two-way Doppler from ESA and DSN ground stations;
- ΔDOR from DSN ground stations [6].

To do so it requires sophisticated mathematical models that compute these measurements to an accuracy better than the actual measurement accuracy. The general requirement is as follows: The accuracy of the modelled observations should be approximately one order of magnitude better than the accuracy of the actual measurements. What this requirement means in actual numbers for a particular measurement type, assuming X band signals, is summarised in Table 1. These accuracy requirements necessitate an extremely high fidelity in the mathematical models.

Table 1. Measurements Accuracies

	Measurement Accuracy		Model Accuracy	
	IFMS	DSN	IFMS	DSN
2-way range [m]	2-5	1-2	0.1	0.1
2-way Doppler [mm/s]	0.1	0.1	0.01	0.01
ΔDOR [nrad]	-	15	-	1.5

The modelling can roughly be divided into three main parts: the transformation between different time scales, the transformation of the ground station Earth-fixed coordinates into an inertial solar system barycentric system, and the computation of the precision light time. From the precision light time the above-mentioned measurement types are derived. Each part of the modelling is described in more detail in the following.

The time scale used in the ground station is Coordinated Universal Time (UTC). The time scale used to propagate the orbit of a spacecraft and the solar system bodies is Barycentric Dynamical Time (TDB), i.e. TDB is the independent variable in the equations of motion. The difference between TDB and UTC is given by the sum of a constant (32.184 s), the current number of leap seconds (+32 s in 2004), and a number of periodic, relativistic terms. The largest term of the latter has an amplitude of 1.7 ms. However, many more relativistic terms are used in the actual computation to meet the required accuracy. We used in the early stage of the development phase an expression for (TDB – UTC) that was derived in [1]. In this formulation (TDB – UTC) is expressed as a sum of trigonometric functions of time; therefore referred to as "trigonometric formulation". JPL has exchanged the trigonometric formulation in the ODP in the 1980s and nowadays uses a so-called "vector formulation" [2]. There, (TDB – UTC) is a function of position and velocity vectors of various solar system bodies and the Earth-fixed position vector of the ground station. The main advantage of the vector formulation is that it is more accurate compared with the trigonometric formulation. In [2] it is stated that the root-sum-square value of the neglected terms in the vector formulation is 4.2 μs. However, the main disadvantage of the vector formulation is that it is much more time consuming in terms of computation. Therefore we developed a so-called "hybrid formulation" by replacing the three major terms of the trigonometric formulation by its equivalent vector expression. An analysis showed that the maximum difference between the vector and the hybrid formulation is 0.3 μs, i.e. an order of magnitude smaller than the overall accuracy of the vector formulation. (TDB – UTC) influences the measurements in two different ways: directly since it appears as a term in the expression that computes the range, and indirectly since it determines the epochs of the participants, namely the signal reception time at the receiving station, the turn-around time at the spacecraft and the transmission time at the transmitting station. The direct effect on the modelling error of a 2-way range observable using the vector formulation is 0.13 m per AU distance [2]. The above stated maximum difference between the vector and hybrid formulation is equivalent to 0.08 m per AU distance.

The position of the ground station is given with respect to the Earth-fixed reference system. For ESA stations, we use the realisation of the International Terrestrial Reference Frame (ITRF) 2000 [4]. The spacecraft state is given with respect to an inertial reference system. We use the International Celestial Reference Frame (ICRF) aligned with the FK5 star catalogue at J2000.0 [5], which is the same reference system with respect to the JPL DE405 export planetary ephemerides are given, that we use throughout the entire orbit determination software. In order to solve the light-time equation a conversion between the ITRF 2000 and the ICRF FK5/J2000.0 is required. It is well known that interplanetary orbit determination is extremely sensitive to errors in the ground station positions [e.g. 6]. Therefore both the original Earth-fixed coordinates and the transformation between the two systems have to be correct and extremely accurate. To meet the required accuracy for modelling the measurements, the station inertial position needs to be known with an accuracy of a few cm.

The station position with respect to the ITRF 2000 is obtained from GPS data and a local survey. This has been performed by an external company for the DSA in NNO and the coordinates have been supplied to us with an accuracy of better than a cm.

The transformation between the two reference systems consists of a series of translations and rotations. The latter comprises the Earth rotation, nutation and precession as well as polar motion. The implemented model is consistent with the IAU 1980 theory of nutation [5]. We retrieve the required data values for these rotations from the Rapid Service of the International Earth Rotation and Reference Systems Service (IERS) [7] every day. Additionally, we apply translations to the station position, whereby we only apply those translations that are expected to be significantly larger than 1 cm. All of them are listed in the following:

- Plate tectonics (1 – 10 cm/year);
- First order solid Earth tides (about 50 cm);
- Modified Lorentz transformation (about 25 cm).

Having the inertial states of the ground station and the spacecraft at their participating epochs (in TDB) the so-called precision light time can be determined. The precision light time describes the time that the radio signal needs to travel between the ground station and the spacecraft. The following effects are taken into account to compute the precision light time:

- The Newtonian light time, i.e. the time light needs to travel on a straight line;
- The reduction of the coordinate velocity below the speed of light due to the gravity field of massive bodies (for the current missions only the Sun, Jupiter, Saturn, and the Earth are considered);
- The bending of the light path due to the gravity field of the Sun;
- The refraction of the signal when passing through the Earth's troposphere and ionosphere;
- The effects due to charged particles in the solar plasma;
- Instrumental delays in the spacecraft or the ground station (e.g. transponder delay, antenna mounting, station electronic delays).

Having the precision light time, the range observable can be modelled easily. Since the reception time at the ground station is recorded in UTC the above-described time transformation (TDB – UTC) evaluated at reception time has to be subtracted from the precision light time. If a two-way range is modelled, (TDB – UTC) evaluated at transmission time has additionally to be added to the precision light time. This is the aforementioned direct effect of the time transformation on the measurements. The two-way Doppler observable is computed by differenced range, i.e. the 2-way range at the start of the count interval is subtracted from the 2-way range at the end of the count interval and the difference is divided by the count time. Usually, we use a count time of 60 seconds. A spacecraft DOR measurement is modelled by differencing the one-way ranges of two different stations and the spacecraft. In a similar way a quasar DOR is computed. In order to build a ΔDOR either two spacecraft DORs around one quasar DOR or two quasar DORs around one spacecraft DOR were taken, so that the surrounding DORs could be interpolated at the epoch of the middle DOR. The difference of the interpolated DOR and the middle DOR subsequently form a ΔDOR.

The modelled measurements are subtracted from the actual measurements to form the residuals. Simultaneously with the modelling of the measurements, the partial derivatives of the measurements with respect to the parameters that are treated as uncertain and directly affect the measurement are calculated. These are used together with the dynamic partial derivatives to form the so-called regression partial derivatives, i.e. the partial derivatives of the measurement with respect to all uncertain parameters. Both, the residuals and the regression partial derivatives are weighted by the measurement standard deviation and serve along with the *a priori* information as input to the SRIF.

This concludes the description of the capabilities of the orbit determination system. It has been developed to this stage over a period of approximately 5 years. But before it could be used operationally it had to be exhaustively tested.

3. TESTING

Several independent tests have been executed to validate the correctness of the new orbit determination system. A series of tests have been performed together with the Test & Validation Office (TVO) within ESOC's Flight Dynamics System. The purpose of the TVO in general is to increase the reliability of Flight Dynamics products by performing independent checks of the results. Independent means in particular that the TVO software is coded without reference to any operational software. As such, extensive tests of individual modules of the orbit determination software, tests of the orbit determination system as a whole, and tests of the orbit determination system as part of the Flight Dynamics System have been performed. It is not the intention of this paper to describe these kinds of tests in any further detail. More focus should be given to tests auxiliary to the normal quality assurance procedures within ESOC. These were the cross-verification tests with JPL's ODP and the tracking campaigns with Ulysses and Mars Express.

3.1 Cross-Verification Tests with JPL

The main purpose of these tests was to validate ESOC's interplanetary orbit determination system against the ODP of JPL. Tests took the form of both Agencies making computer program runs using identical input data, then comparing the results and assessing the differences. Scenarios have been chosen that are representative for Rosetta and Mars Express. Since the ODP is mature and has been successfully applied for many interplanetary missions, JPL results have been treated as the reference against which the correctness of ESOC results have been compared. A phased approach has been followed for the tests, i.e. each module of the orbit determination system has been tested sequentially. In the following the sequence of tests is described and the final test results are presented. More results and technical details of the tests can be found in [8]. Some of the tests have brought to light software bugs, modelling errors, or a lack of sufficiently high fidelity modelling. These are not described here since their descriptions would be too detailed. However, it should be stressed that the cross-verification tests were extremely useful to identify and to remove these kinds of deficiencies in the new ESOC system.

Trajectory Propagation

Three different test scenarios were developed:

- 90 days heliocentric cruise with 5 different force models. The position difference at the end of the arc between ESOC and JPL were ca. 1m in all 5 tests.
- 20 days centred on (Rosetta) Mars swing-by with 3 different force models. The position differences at the end of the arc were ca. 44 m in all three tests. In this highly dynamic scenario, the end state is very sensitive to the state at pericentre, where the differences were about 0.1 m in position and 0.05 mm/s in velocity.
- 2 days Mars-centric orbit with 3 different force models. The position differences at the end of the arc were less than 2.2 mm.

Modelling of 2-Way Range and Doppler

Actual measurements of the Nozomi and Stardust spacecraft acquired at DSN stations have been used to define the test scenarios. Both spacecraft were in heliocentric cruise at 1.72 and 1.24 AU geocentric distance respectively. Many sources of differences concerning the operational set-ups between ESOC and JPL have been identified, most important the usage of different realisations of the ITRF, different sources of the Earth Orientation Parameters, and different formulations in the expression of (TDB – UTC) (see section 2). For the Nozomi measurement these have been removed and the Agencies' set-ups have been artificially aligned. By doing so differences of 13.5 mm and 0.0005 mm/s in 2-way range and Doppler respectively have been achieved. For the Stardust case, the Agencies have used their normal operational set-ups and Table 2 summarises the results. The differences are well within the bounds defined by the required accuracy as given in Table 1. Some auxiliary tests related to the modelling of measurements were performed, namely a multi-pass and station modelling of 2-way range and Doppler, modelling of tropospheric and ionospheric effects, as well as modelling of measurements received at DSN stations with remote electronics (some antennas at Goldstone). The results were all compatible with the differences given in Table 2.

Table 2: Summary of observation modelling

Parameter	Difference (ESOC – JPL)
(TDB–UTC) at reception and transmission time	-90 ns
Precision round-trip light time (TDB)	0.17 ns
ICRF station position at reception time	34.5 mm
ICRF spacecraft position at turn-around time	5.3 mm
ICRF station position at transmission time	35.4 mm
2-way range measurement	55.8 mm
2-way Doppler measurement	0.003mm/s

Modelling of ΔDOR

Rosetta in heliocentric cruise was taken as scenario for modelling a ΔDOR measurement. DSN stations at Goldstone and Canberra were assumed to take artificial measurements of quasar and spacecraft DORs. The differences in the DOR measurements were large (~10 nrad) mainly due to the different formulations of the time transformation (TDB – UTC) and the usage of different realisations of the ITRF. However, these cancel out in the end when a ΔDOR is formed and the final differences in the ΔDOR were 1.2 nrad. This is well within the accuracy requirements as given in Table 1.

Partial Derivatives

These tests have been split into two parts: tests of the dynamical partial derivatives and tests of the regression partial derivatives. For the former, the dynamic scenarios from the trajectory propagation tests were taken and the partial derivatives with respect to selected dynamic parameters – most important the spacecraft state at epoch – were compared at the end of the arc. Similarly, for the latter, a range and a Doppler measurement were modelled at the end of the arc and the partial derivatives with respect to selected measurement parameters – again most important the spacecraft state at epoch – were compared. The relative differences between ESOC and JPL results were less than 10^{-4}. Since the partial derivatives do not have as strict accuracy requirements as the measurement modelling itself, these differences were fully acceptable.

Covariance Mapping and Target Plane Parameters

After an orbit determination the spacecraft state together with its uncertainties are mapped to a future epoch. When targeting to a solar system body is performed the future epoch chosen is the time of closest approach and the miss distance and its uncertainties are expressed in the B-plane. To ensure the correctness of this part of the software a 60-days arc resembling Mars Express' approach to Mars with four different dynamic models was chosen for the test. The relative differences between ESOC and JPL results were less than 10^{-4} in the B-plane quantities. This was also fully acceptable.

3.2 Tracking Campaigns

Three tracking campaigns have been performed, two of them with Ulysses and one of them with Mars Express. The Ulysses tracking campaigns took place in September 2002 and in March 2003. The Mars Express tracking campaign was performed in August 2003. The latter two campaigns were mainly dedicated to check ESOC's ability to retrieve and process DSN ΔDOR measurements, first in general (Ulysses) and second with the spacecraft at hand (Mars Express). In this paper, focus is given to the first Ulysses tracking campaign since this was the first semi-operational end-to-end deep-space test of not only the new orbit determination system but also of the new DSA in NNO. The overall campaign objectives and results are summarised. A more detailed description can be found in [9].

The Ulysses tracking campaign in 2002 consisted of four tracking passes made from NNO with S-band up- and X-band downlink. The Ulysses geocentric distance at that time was 4.8 AU. The campaign had four main objectives:

1. Validate the capability of NNO to track a deep space spacecraft;
2. Validate the capability of NNO to perform ranging and Doppler measurements in a deep-space scenario;
3. Validate the end-to-end provisions and processing by Flight Dynamics of the tracking data;
4. Validate the capability of NNO to perform telemetry function in a deep space scenario.

Objective 3 was the most important one as far as the orbit determination system was concerned. To fulfil it, the routinely acquired range and Doppler tracking data of Ulysses, as tracked from the DSN, were delivered to ESOC over a period of almost a month. These data were processed by the orbit determination system and an orbit solution using just the DSN data was generated. The same exercise was done at JPL by using the ODP. A tremendous effort was taken to ensure that both Agencies used consistent orbit determination set-ups in order to allow a sensible comparison. The results were carefully compared and the differences were all acceptable and to a level that was expected from the cross-verification tests. To give an indication, the spacecraft position at epoch was different by about 5 km, the spacecraft velocity at epoch by about 1 mm/s. In a certain sense, this was a final cross-verification test since a full orbit determination solution was compared.

During this period, NNO tracked Ulysses over four passes. The tracking data were delivered to the Flight Dynamics System and processed by the orbit determination software. In order to get an estimate of the noise and bias that had to be expected, the NNO data were "passed through" the above-mentioned DSN data only solution, i.e. the NNO measurements were modelled by using the values of the solve-for parameters obtained from the DSN data only solution. The residuals were computed but no new solution generated. The resulting Doppler residuals are shown in Fig. 2. The lightly shaded symbols represent post-fit residuals of DSN data. They exhibit a pattern that one expects of a good deep-space solution: the residuals are scattered around zero mean with a standard deviation of the expected measurement noise. The darkly shaded points represent the pre-fit residuals of the NNO data.

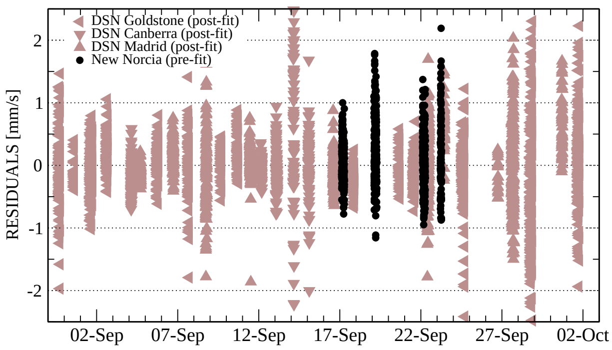

Fig. 2: Pass-through of NNO Doppler residuals on a DSN data only solution.

At the end of the tracking campaign, after more orbit solution comparisons between ESOC and JPL, confidence was achieved that the orbit determination system works correctly and yields reliable results. This was a big relief for the team after 4 years of development and we anticipated the first real operational application with delight.

4. CONCLUSIONS AND OUTLOOK

This paper gave a comprehensive overview about the capabilities of ESOC's new system for interplanetary orbit determination. It furthermore detailed the efforts for testing the system before it became operational. The system has been used extensively for spacecraft operations for more than one year now and two references can be given that detail the operational application of the system for Mars Express [10] and SMART-1 [11], both issued in these Proceedings. However, the development phase is still not over. Future enhancements are necessary to support ESA's current and future interplanetary missions. Planned updates in the near future are to add antenna angles and optical camera images as measurement types. Furthermore, the new formulation of the theory on Earth precession and nutation [7] is going to be implemented in order to stay compatible with IERS products.

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