FULTT Executive Summary

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1 ACRONYMS

∆DOR	Delta Differential One-way Ranging
DFACS	Draft Free and Attitude Control System
LGA	Low Gain Antenna
LPF	Lisa Path Finder
MGA	Medium Gain Antenna
MNM-OS	Micro-propulsion Normal Mode On Station
MPACS	Micro-Propulsion Attitude Control Subsystem
MPS	Micro-propulsion System
RF	Radiofrequency
SP	Saddle Point
SRIF	Square Root Information Filtering
SRP	Solar Radiation Pressure

2 REFERENCE DOCUMENTS

[1] *Lisa Pathfinder: Consolidated Report for Mission Analysis,* S2-ESC-RP-5001, Issue 4 Revision 5, M. Landgraf, F. Renk, and B. de Vogeleer – 31/03/2014

[2] *Detailed Proposal: Feasibility of ultra-low thrust transfers in L1, L2, Sun, Earth & Moon Systems*, Airbus Defence and Space – 17/05/2016

[3] *JUICE : Navigation Analysis for the Jupiter Tour, HSO-GFA-TN-092*, T. Yamaguchi and A. Boutonnet, ESOC, 14/06/2013

[4] LISA PATHFINDER COLD GAS THRUSTERS CANTING ANALYSIS FOR MPACS & DFACS - EXTENDED MISSION STUDY UPDATE

3 INTRODUCTION

This report provides a summary of the study: Feasibility of Ultra Low Thrust Transfers. The study was performed by Airbus Defence and Space for ESA under **ESA/ESTEC Contract Number 4000118200/16/F/MOS.**

The report contains an overview of the background, the tools used for the study and their application to the main test case, which was a potential extension of the Lisa Pathfinder mission to fly through the Earth Sun saddle point.

Issues addressed include trajectory identification and optimisation, plus the solution of the associated navigation problem.

4 BACKGROUND AND ASSUMPTIONS

4.1 Micropropulsion

4.1.1 Overview

The primary cold gas micropropulsion system on-board LPF is based on a set of six thrusters arranged in a unidirectional configuration – all six thrust vectors have a component in the sun direction, opposing SRP. A nominal and a redundant set of thrusters can be used – the thrust vectors are co-aligned, however the difference in position results in small efficiency differences.

The maximum thrust authority depends on the thrust direction, and is largest along the +z direction, when all six thrusters are firing. The maximum thrust authority also depends strongly on the payload status:

- Payload off / TMs grabbed: approx. 1.5mN (potentially 3mN if TAA limits are modified)
- Payload operating as accelerometer: approx. 100microN

4.1.2 Thrust vector capabilities

Analysis of thrust efficiency in a range of directions is described in [4]. The thrust efficiency (expressed in units of kg of propellant consumed per m/s deltaV) depends on the direction of applied thrust in the spacecraft frame. Referred to inertial space, tilting of the spacecraft can also be used to adjust – within limits – the most efficient thrust direction.

Figure 4-2 shows the maximum achievable force magnitude as a function of the DeltaV demand direction (expressed by the angle theta). Please note that “force magnitude” in the figure is the total magnitude of force achieved, regardless of the direction of such force. For the range of θ for which $\theta=\gamma$ (i.e. approximately -40º< θ<40º as can be seen by subfigure 2 in **Figure 4-2**), the force magnitude is all in the demanded direction. It can be seen that the achieved force vector starts to diverge from the commanded force vector above a given angle of Θ (angle between thrust vector command and spacecraft z axis, e.g. for the redundant set, Θ and γ diverge for angles Θ < - 45°). The angles Θ at which there is a divergence is where the DeltaV is most efficient. **Figure 4-1** gives the definition of the angles γ and Θ.

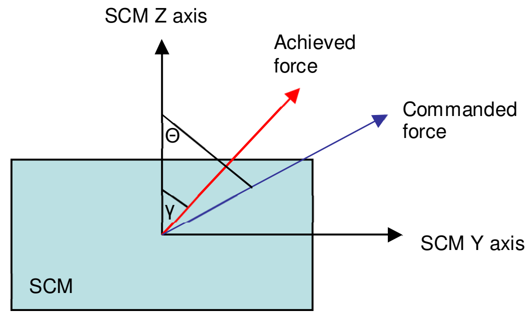

Figure 4-1 Definition of angles γ and Θ

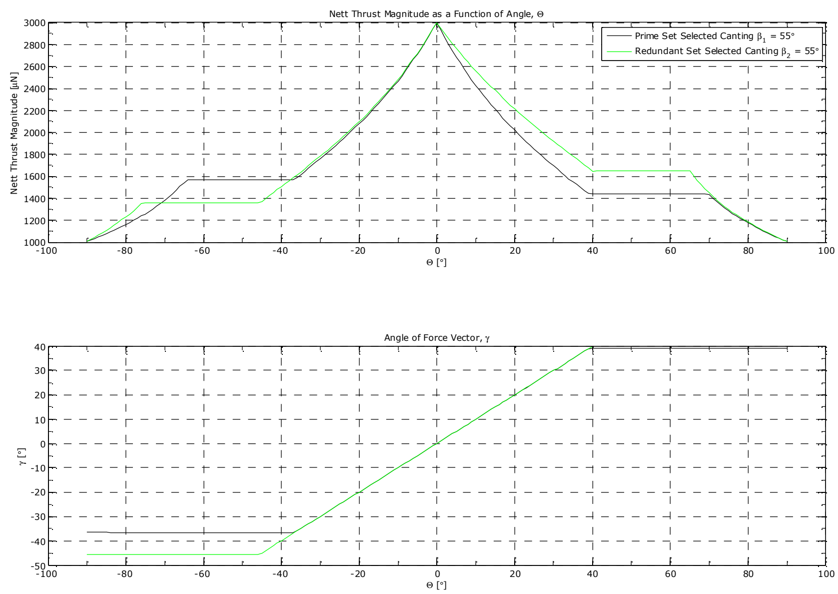

Figure 4-2: Force magnitude as a function of DeltaV demand direction

When operating in Accelerometer mode (ie with free test masses) maximum acceleration and hence thrust is constrained. Net thrust will not exceed 100 microN. This is achieved by commanding lower thrust per thruster. The result is that a flat thrust profile wrt thrust direction angle can be achieved over the range +/- 90 deg. Note that these figures do not take into account the inefficiency from orientation. This effect halves the net thrust at all angles (eg 3000 microN maximum reduces to 1500 microN maximum).

4.1.3 Modes of operation and thrust implications

LPF can operate in several modes which could be considered in the context of this study. These are summarised as follows:

Drag free mode. The payload test masses are free flying in this mode. The spacecraft micropropulsion is used to compensate all external non-gravitational forces so that the only gravity is acting on the free flying test masses. The spacecraft exerts a gravitational influence on the test masses (self gravity) which if unchecked would cause the test masses to impact their housing. To compensate for this effect the spacecraft accelerates with a vector designed to compensate for the self gravity vector, so that the net acceleration on the test masses, excluding external gravity, approaches zero.

Accelerometer mode. The payload test masses are free flying in this model. However the spacecraft is no longer compensating the external non gravitational forces. The test masses are controlled by their suspension system to maintain position relative to the spacecraft. This potentially enables measurement of the non-gravitational acceleration. In this and drag free mode upper limits on spacecraft acceleration must be observed in order to allow controllability of the free flying test masses

Captured mode. The test masses are captured, no longer able to move relative to their housings. In this mode higher acceleration is possible, limited only by the thrusters capability.

Due to the LPF thruster configuration, a parasitic Z force of about 10µN is continuously produced. Smaller forces on the other axes (X/Y) are also possible. On this type of mission with very stable Sun pointing attitude, it is likely that these forces could be predicted with good accuracy after initial calibration, thus minimising the impact on navigation. However this force must be considered in trajectory planning

4.2 General requirements

The ultimate aim of the study is to advance the understanding of ultra-low thrust transfers in the L1, L2, Sun, Earth & Moon system. In order to provide some boundaries to this very broad challenge, within the study SOW it is specified that the following constraints and parameter ranges shall be considered:

- Spacecraft (SC) mass range 200-2000kg
- Total thrust authority range 0.1 – 10mN
- Total available deltaV range 1-10m/s
- Possible dependence of thrust authority and efficiency on direction
- Maximum transfer duration < 5 years after libration point departure
- Large amplitude orbits in the L1/L2/Sun, Earth & Moon system
- Accurate navigation within L1, L2, Sun, Earth & Moon system
- Current ground system capabilities

One of the outcomes of the work of this study is the achievable transfer accuracy, as a function of the remaining parameters.

5 NAVIGATION ASPECTS

5.1 Overview

This section summarizes the inputs and outputs respectively required and provided by the Navigation Model which will be used in the Feasibility Analysis to assess navigation performance.

5.2 Inputs

5.2.1 Sensor performance

Two types of “sensors” (in the broad sense) can be used in LLT navigation as considered for LPF: the P/L as an accelerometer, and the ground stations. Their performance will drive the navigation accuracy for obvious reasons. In addition, their availability will have various impacts on the overall GNC behaviour: the accelerometer measurements are generated on-board in almost real time but the navigation is performed on ground, so that if the navigation performance depends on them, it can only be fully achieved much later than the acquisition of measurements, which subsequently delays the ensuing control manoeuvres. This is a particularly important aspect when considering the very limited capability of error recovery of the S/C (limited thrust from the micro-propulsion and ΔV allocation), as excessive delays can lead to irrecoverable states. Ground station navigation is also limited by station visibility but possibly also by antenna pointing given that the attitude of S/C is constrained, in particular in DFACS mode.

5.2.1.1 Accelerometers

Detailed performance data on LPF payload acting as an accelerometer is not available, but based on residual forces in DFACS mode it is expected that it would able to measure non gravitational forces (RP) to an extent better than 5nm/s² accuracy. Indeed in DFACS mode, the P/L permits exact compensation of RP and the only perturbation force which remains is the left gravity with a level of 5 nm/s² (and self-gravity differential one order of magnitude below).

Thus, it could be expected than in other modes than DFACS, the P/L could be used to reduce environment uncertainties, in particular in MNM-OS remove the large uncertainty (10%, 1σ) on the SRP, and perhaps also some of the smaller uncertainty on MPS accuracy.

5.2.1.2 Ground stations

Ground stations are expected to be used for range and Doppler navigation. [2] does not exclude ΔDOR, but given the complexity of this solution it should probably be used as back-up for higher performance. In order to parametrize the navigation models for this "sensor", one must first know its typical errors and measurement frequency. They are given in Table 5-1 from [1].

Parameter	Value	Comments
Range noise 1σ	20m	Assumed white since no correlation given and similarly to JUICE
Range bias 1σ	20m	Bias will be constant for a simulation but can take any value with Gaussian distribution of zero mean and said standard deviation in Monte Carlo runs
Range frequency	1/pass	JUICE was using 1/h, and LEOP 0.08Hz according to [1] so benefits of higher rates could be investigated
Doppler noise 1σ	0.03 mm/s	10 times lower than for JUICE (see [3]), assumed due to much closer distance to the Earth
Doppler bias	0	Also for JUICE
Doppler frequency	1/10 min	Also for JUICE
Station position error (1σ)	1m	Out of equator plane, assumed constant for 1 simulation
	30cm	In equator plane, assumed constant for 1 simulation

Table 5-1: Ground station measurement errors and frequency

Note that these errors are understood as coming from a ground link using the MGA. Availability of MGA is only guaranteed in a given S/C attitude corridor dependant on S/C distance to Earth.

The baseline will consider that the mission can be tracked using LGAs, in an omnidirectional classical configuration, so that ground station link depend only on the visibility of the latter and not on S/C attitude. Visibility is given by the geometry and station location as in **Error! Reference source not found.**, as well as the minimum elevation above the horizon: 10 degrees. Attention will have also to be paid to possible link obstruction by the lunar conjunctions.

Ground stations considered are Cebreros, Kourou, Perth and Maspalomas.

It is assumed that navigation and in particular ground tracking is not used during manoeuvres.

5.3 Outputs

The navigation tool outputs include the following:

- S/C estimated position at each time step in reference frame
- S/C estimated velocity at each time step in reference frame
- S/C position and velocity estimation covariance matrix in reference frame

Note also that the navigation tool will implement an Extended Kalman Filter, slightly different from the standard filtering done on ground, which is usually done through Square Root Information Filter and processes batches of measurements to obtain the estimated state at a specific date. The Kalman filter, sequential, is able to provide estimation at every time step, including specified ones so will cover the same functionality, and also add the important information of the estimation convergence, so that it is possible to evaluate also how long the navigation requires before a certain level of accuracy (necessary for instance for a correction manoeuvre) is reached.

6 LPF PAYLOAD AND ACCELEROMETER USAGE

6.1 INTRODUCTION

In this section it is investigated how the payload on-board LISA Pathfinder could be used in estimating the non-gravitational accelerations on the spacecraft. It is thought that this measurement could be used to improve the navigation efficiency.

It will be assumed that the payload is used in accelerometer mode, rather than in drag-free mode. The two main reasons for doing this are:

(1) Propellant consumption – the propellant used in drag-free mode is about 4 times as much as in accelerometer mode
(2) While LPF can be operated in drag-free mode, most spacecraft can not. In this sense, analysing the accelerometer mode will have much broader applicability, since a highly sensitive accelerometer could in principle be added to any spacecraft.

In addition the performance of the GAP accelerometer is assessed in this context

6.2 Principle

In the LPF accelerometer mode, all TM DOFs are "suspended", ie the TMs are controlled electrostatically to follow the SC, in translational as well as rotational DOFs, and at all frequencies. The aim is to estimate the non-gravitational SC accelerations from the electrostatic forces that have to be applied to the TMs. Provided the residual interaction between SC and TM has been characterised, the external non-gravitational forces acting on the SC can be derived from the applied capacitive actuation:

$$F_{SRP} + F_{Thr} = \left(F_{capact} + F_{SC\to TM}\right)\left(\frac{m_{SC}}{m_{TM}} - 1\right)$$

6.3 Non-gravitational force estimate error budget

In the above equation, all quantities on the right hand side are in principle subject to errors and noise sources, which limit the accuracy and precision with which the non-gravitational forces can be estimated.

6.3.1 Non-gravitational SC Force Uncertainty Budget

Summarising the above, a very coarse estimate of the uncertainty on the non-gravitational force estimate during accelerometer mode can be obtained as follows:

$$\delta(F_{SRP} + F_{Thr}) = \left(\delta F_{capact} + \delta F_{SC\to TM}\right)\left(\frac{m_{SC}}{m_{TM}} - 1\right)$$

Uncertainty Contribution	Systematic	Statistical (5000s integration time)	Comments
Capacitive Actuation	1% of 10^{-8}N = 10^{-10}N	2×10^{-11} N	Systematic uncertainty due to uncertain gain factor
Spacecraft self-gravity	1×10^{-9}N		Could be improved by better measurement on-board, or sufficiently accurate ground tracking.

With the above estimates, the total uncertainty on the external non-gravitational forces is estimated as

$\delta(F_{SRP} + F_{Thr}) \approx (1\times10^{-10} + 1\times10^{-9})(212.5 - 1) = 2.3\times10^{-7}N$

As can be seen, the uncertainty is dominated by the knowledge of self-gravity – if this could be reduced to the level of 10^{-10}N, the overall uncertainty would be reduced the uncertainty by approximately a factor of 5.

6.4 GAP accelerometer

Unlike the LPF payload GAP is designed from the outset as absolute accelerometer.

Accelerometer performance and noise sources better characterised
Much more modest in terms of mass & power budgets
Almost 'COTS' – long line of ONERA accelerometers heritage

Published error budget:

	Contributor	Allocation	Explanation
Accelero-meter	E1	1 pm/s^2	From noise **Erreur ! Source du renvoi introuvable.**
	E2	2 pm/s^2	With temperature measurement at 0.3 mK and bias thermal sensitivity at 6.5 $nm/s^2/K$, calibrated
	E3	3 pm/s^2	Considering scale factor known at 10^{-3}
	E4:	0 pm/s^2	The quadratic factor will be less than 100 s^2/m
Both	E5:	3 pm/s^2	Considering a global misalignment of 1 mrad
Spacecraft	E6:	1 pm/s^2	Considering accelerometer at 0.1 m of S/C COM, angular rate stability of 1 µrad/s and pointing of 100" over 10 000 s (giving angular acceleration of $\Delta\theta/T^2 = 5\ 10^{-12}\ rad/s^2$)
	E7:	90 pm/s^2	Requirement on S/C self-gravity for [0-0.1] mHz
	E8:	2 pm/s^2	Requirement for [0-0.1] mHz
	Margin	43 pm/s^2	
	Total	100 pm/s^2	Quadratic sum, valid for [0-0.1] mHz

Table 6-1 GAP error budget

The budget is dominated by uncertainty of SC self-gravity, but ≈ 5x better than LPF

7 TRANSFER DESIGN TOOL

7.1 Introduction

This note describes the Software Design of the Transfer Design Tool as used in the FULTT study. The tool is multi-functional consisting of a trajectory propagator, search facility to identify trajectories for multi-saddle point crossings and an optimisation facility to refine those trajectories identified as suitable candidates in the searches.

7.2 Modes of Operation

7.2.1 Propagation mode

In this mode the tool will propagate the spacecraft trajectory, starting from a defined initial Cartesian state, mass and epoch. The duration of the propagation is specified. For the Saddle Point (SP) crossing problem, durations of typically 2 to 4 years are considered..

A sequence of manoeuvres using the specified thrust can be applied.

7.2.2 Search mode

In this mode multiple propagations using the above propagation mode are performed, all with the same starting and modelling assumptions but differ wrt the manoeuvres.

For each of the three possible manoeuvres increments in start time and DeltaV vector (and hence manoeuvre duration and direction) can be applied.

7.2.3 Multiple Shooting conversion mode

Having obtained a suitable manoeuvre sequence via the above described Search Mode the problem is now converted to one of optimisation. This is done via an intermediate process of converting the Single shooting propagation (ie only one initial value and a single propagation)) into a Multiple shooting propagation (ie many initial values each followed by a propagation).

7.2.4 Multiple Shooting propagation mode

After generation of a sequence of multiple shooting segments a multiple shooting propagation can be performed. The number of segments and initial values are read and the corresponding sequence of propagations is performed.

7.2.5 Optimisation mode

In this mode the previously obtained multi-segment trajectory can be optimised. The objective is to maximise mass at the end of the simulation (ie minimise propellant usage) whilst observing specified constraints. Additional constraints enforced by the optimisation process are the inter-segment errors (ie final value n – initial value n+1) = 0. Optimisable control parameters are the initial values of each segment, the segment durations and for the thrusting segments, the thrust direction (as an azimuth and elevation angle).

8 TRANSFER FEASIBILITY TOOL

8.1 Introduction

This note describes the use of the Transfer Feasibility Tool as used in the FULTT study. The tool is multi-functional consisting of a trajectory propagator and a multiple shooting propagator, together with options for Monte-Carlo simulation.

8.2 Modes of Operation

8.2.1 Propagation mode

In this mode the tool will propagate the spacecraft trajectory, starting from a defined initial Cartesian state, mass and epoch. The duration of the propagation is specified. For the Saddle Point (SP) crossing problem, durations of typically 2 to 4 years are considered.

A sequence of manoeuvres using the specified thrust can be applied

8.2.2 Multiple Shooting propagation mode

After generation of a sequence of multiple shooting segments a multiple shooting propagation can be performed. The number of segments and initial values are read and the corresponding sequence of propagations is performed.

8.2.3 Closed Loop Guidance and Navigation mode

In this mode the previously obtained multi-segment trajectory can be simulated using a closed loop Guidance and Navigation functionality.

The top level mode of operation is the same as the multiple shooting mode in that a datafile is read defining initial values for a sequence of segments, together with manoeuvre data for the segments.

In this mode the individual segments are simulated by application of Guidance and Navigation functions, the so called ‘closed loop simulation’. The target reference trajectory for the Guidance function is the nominal propagation based on the initial segment and manoeuvre data (if a thrusting segment). This reference trajectory is propagated in parallel with the closed loop simulation.

The only explicit guidance target is to follow the reference trajectory. By using a pre-generated reference trajectory that fulfils the mission constraints, such as saddle point crossing, then ensures that these constraints are met by the closed loop system within the accuracy and controllability achievable.

Errors are introduced into the trajectory via

- Implementation error: Thrust error
- Environmental error: SRP
- Navigation error: estimated position and velocity

The guidance function can be run in a selected mode:

- Corrections using manoeuvre pair and fixed interval between start and end of manoeuvres, run in continuous loop with first manoeuvre of current pair replacing second manoeuvre of previous pair
- Corrections using manoeuvre pair and fixed interval between manoeuvres, run in continuous loop with first manoeuvre of current pair replacing second manoeuvre of previous pair
- Corrections using manoeuvre pair and fixed interval between manoeuvres, run in threshold mode such that manoeuvre pair only executed if specified position error threshold exceeded

A biasing SRP force can be applied to the trajectory. This forces guidance correction manoeuvres to be applied in a Sunward direction which if of sufficient magnitude, when added to the error correction manoeuvres, can force the net manoeuvre direction to lie in the Sunward hemisphere, ie consistent with potential spacecraft constraints. The SRP bias would be consistent with a planned underestimation in the SRP used in the trajectory design.

8.2.4 Feasibility check mode

In this mode a pre-generated solution file (ie multiple shooting segment data file) is propagated to perform an estimate of feasibility of the transfer, given real world constraints on the spacecraft. Aspects assessed are:

- Closest approach to Earth and Moon: Gravity gradient
- Total Deterministic Manoeuvre DeltaV/propellant mass
- Transfer timeline
- Deterministic Manoeuvre directions wrt Sun direction

8.2.5 Monte-Carlo simulations

The program also implements a Monte-Carlo facility. This allows repeated simulations to be performed via a top level loop around the Multiple Shooting/Closed loop Guidance and Navigation function.

Errors are generated in:

- Measured Position
- Measured Velocity
- Achieved Thrust
- Solar Radiation Pressure

These are implemented via defined means and standard deviations with normal distributions. The Position and Velocity error covariance terms are either specified as constants throughout the trajectory or as time dependent variables from pre-generated data using the stand-alone Navigation tool.

9 CASE STUDY – LISA PATHFINDER

9.1 Introduction

This case study is focused on a mission extension to an LPF like spacecraft. The aim of the extension is to transfer the spacecraft from an existing Lagrange Point Orbit (LPO) to achieve passages through the Sun-Earth gravitational equilibrium points.

9.2 The LPF transfer problem

The LPF orbit is a large Lissajous orbit with significant out of plane amplitude. Large out of plane amplitudes complicates the transfer to the Saddle Point because it lies in the ecliptic. The higher inclination implies that larger manoeuvres could be needed to help to modify the plane and present more opportunities for Saddle Point crossing.

The global range of transfers can be explored by application of small escape manoeuvres in the Lissajous orbit. A preliminary mapping of trajectory characteristics can be achieved via this process.

A search based on the LPF orbit shows that many classes of transfer exist. Many of these transfers include close or distant Lunar flybys. Distant flybys are attractive as they offer the potential to achieve significant trajectory modification without the difficulties of navigating through a deep gravity well.

9.3 Grid search strategy

The first set of 2D grid searches was performed covering a range of manoeuvre start times and DVs. Also a range of initial orbits corresponding to alternative start epochs were considered.

An example of the initial orbit is the LPF LPO with initial state given by the .oem file. The trajectory is propagated forwards for nominally 2.5 years – allows sufficient time for multiple SP crossings to evolve whilst limiting operational overheads.

Searches have been performed with DV specified in inertial or rotating frames. Searches have been repeated with impulsive and finite thrust. Finite thrust either 1000 microN or 100 microN.- corresponding to captured or free flying payload test masses.

For a 1 m/s DV implemented with 1000 microN or 100 microN takes approximately 5 or 50 days respectively

- Range of manoeuvre start times considered is up to 230 days (> 1 rev in the LPO)
- Range of DVs considered is typically 0-1 m/s.

The outputs of the search are the closest approaches to the Saddle Point and also closest Lunar approach. Multiple minima in SP distance occur over the period of the transfer.

9.4 Optimised Transfers for case study

9.4.1 Overview

Using preliminary trajectories obtained by the grid search process, selected cases have been further investigated via the application of the Transfer Design Tool to achieve zero miss distance (wrt the SP) and minimise or constrain DV requirements.

These solutions require at least one and in general two Saddle Point crossings. The objective is to find transfers efficient in propellant usage. Key characteristics of manoeuvres, ie manoeuvre directions, are considered. The influence of solar radiation pressure has also been considered.

9.4.2 Example case

The initial trajectory estimate in this case has two SP close approached at approximately 1000 and 7000km. Closest approach to the Moon is over 100000 km so Lunar influence is low. This is based on the .oem solution of 24th September 2016.

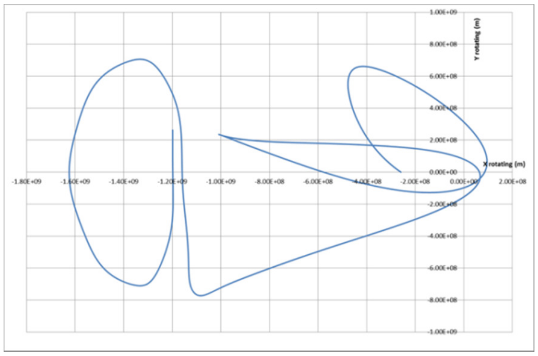

Figure 9-1 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 1 to SP1

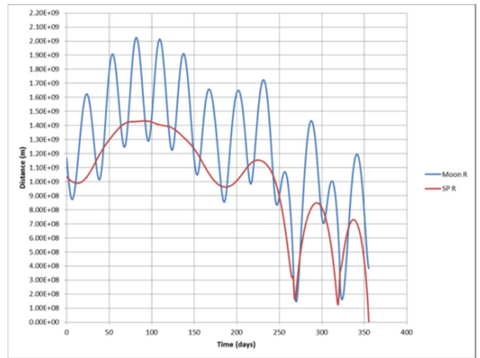

Figure 9-2 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 1 to SP1 distance evolution

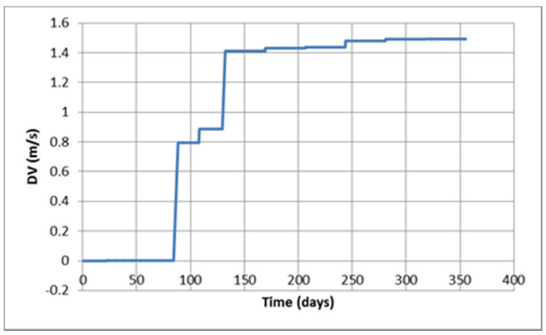

Figure 9-3 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 1 to SP1 DV evolution

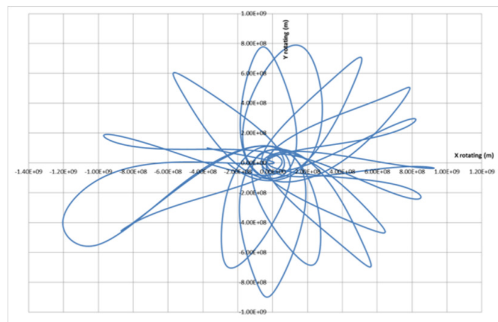

Figure 9-4 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 2 SP1 to SP2

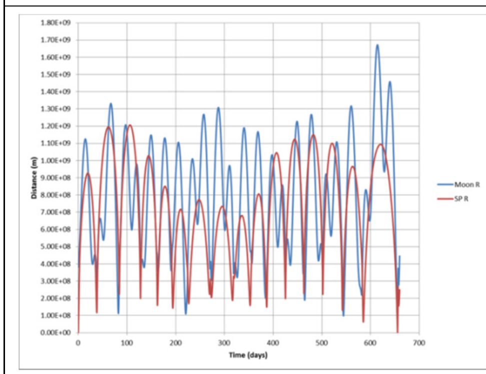

Figure 9-5 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 2 SP1 to SP2 – distance evolution

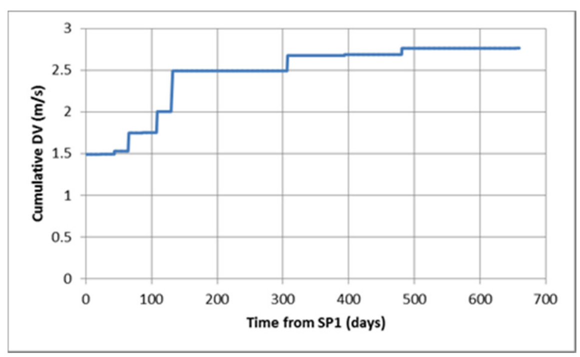

Figure 9-6 Optimised case 1 approach 2 transfer using 1000 microN thrust manoeuvres: Initial condition September 2016 . Part 2 SP1 to SP2 – DV evolution

Alternative approaches were used in obtaining the solution:

- Starting from the two SP crossing preliminary solution find the exact 2 SP crossing solution

- Start from a partial solution reaching the first SP crossing and find the exact solution. Then perform additional grid searches to find an efficient second SP crossing case. Optimise the resulting transfer from the first SP crossing to reach the second crossing

The second approach results in lower DV of less than 3 m/s. Approach 2 is baselined. In approach 2 the first solution is optimised to SP1 crossing. The second solution starts from SP1 and is optimised to SP2 crossing. No close Earth approaches occur (except after SP2): ***Total DV = 2.8 m/s***

Further solutions were obtained to investigate the effects of lower thrust and Solar Radiation pressure on the transfer DeltaV.

10 TRANSFERS INCLUDING GUIDANCE AND NAVIGATION

The effects of navigation and errors in Thrust and SRP assumptions must be considered in order to obtain the total DV budget and hence feasibility of the transfer under consideration. A guidance algorithm is used to derive thrust commands and close the simulation loop, to restore the trajectory to the nominal reference case (ie the case without any error). Different transfer types will be affected in different ways by the inclusion of these errors. Selected cases derived with the transfer design tool were assessed.

10.1 Navigation analysis

Analysis of selected transfers was performed with the Navigation Tool to generate covariance and mean data files for use with the Transfer Feasibility Tool. NavFilter covariances are consistent with antennas/GS data. LPF's MGA is the default (28deg full beamwidth). Ground stations used are: CEB (preferred) / PER / KOU / MAS. The option was used for hybridisation of accelerometer measurements.

Parameter	Value		
Range noise 1σ	20 m (MGA) 20 m (LGA)	Station position error (1σ)	1.0 m (out)
Range bias 1σ	20 m		0.3 m (in)
Range frequency	1/1 h	Minimum elevation	10 deg
Doppler noise 1σ	0.03 mm/s (MGA) 0.10 mm/s (LGA)	ECRV ΔSRP	6.7 nm/s² bias
Doppler bias	0	ECRV MPS noise	0.4 nm/s² / 1 d
Doppler frequency	1/10 min		0.4 nm/s² / 100 d
AMU noise	500pm/s² / 2 h		
AMU scale factor 3σ	1000ppm		

Table 10-1 Navigation analysis key assumptions

10.1.1 Navigation analysis for selected cases

The following plots in Figure 10-1 show evolution of the standard deviations and mean values in the state vector from the navigation analysis for the example case in the previous section.

Further results were obtained with the assumption of improved navigation performance assuming accelerometer hybridisation.

10.2 Stochastic error from navigation analysis

Position and velocity errors are generated for selected transfers using the navigation tool. The evolution of the covariance matrix and mean values for a given reference trajectory are obtained in a pre-processing phase and then used by the Functional Navigation tool in the closed loop analysis.

The guidance algorithm frequency is nominally lower than the convergence period observed for the navigation after a manoeuvre/update. This means that with infrequent guidance manoeuvres the converged navigation covariances and means can be used for error generation in the closed loop.

Special cases exist where more frequent guidance manoeuvres are needed, close to pericentre (either Earth of Moon pericentre). Low frequency operation in this region can lead to exponential error growth that becomes irrecoverable with the limited thrust. For example guidance manoeuvres at circa 1 day before and after pericentre could be typical (ie guidance manoeuvre intervals between circa 1 and 4 days). The navigation solution will not in general be converged over such shorter periods. Consequently a pre-processing analysis is performed to obtain an error multiplication factor (>1) as a function of time from the last guidance manoeuvre. The factor returns to 1 after a period of several days.

Stochastically generated errors in position and velocity are obtained consistent with the pre-derived covariance matrix and mean values. In addition Thrust implementation and SRP errors are stochastically generated with a defined 3 standard deviation value. A Monte Carlo simulation is performed with nominally 1000 individual simulations. Probability density functions for DeltaV are obtained and trajectory evolution wrt reference monitored.

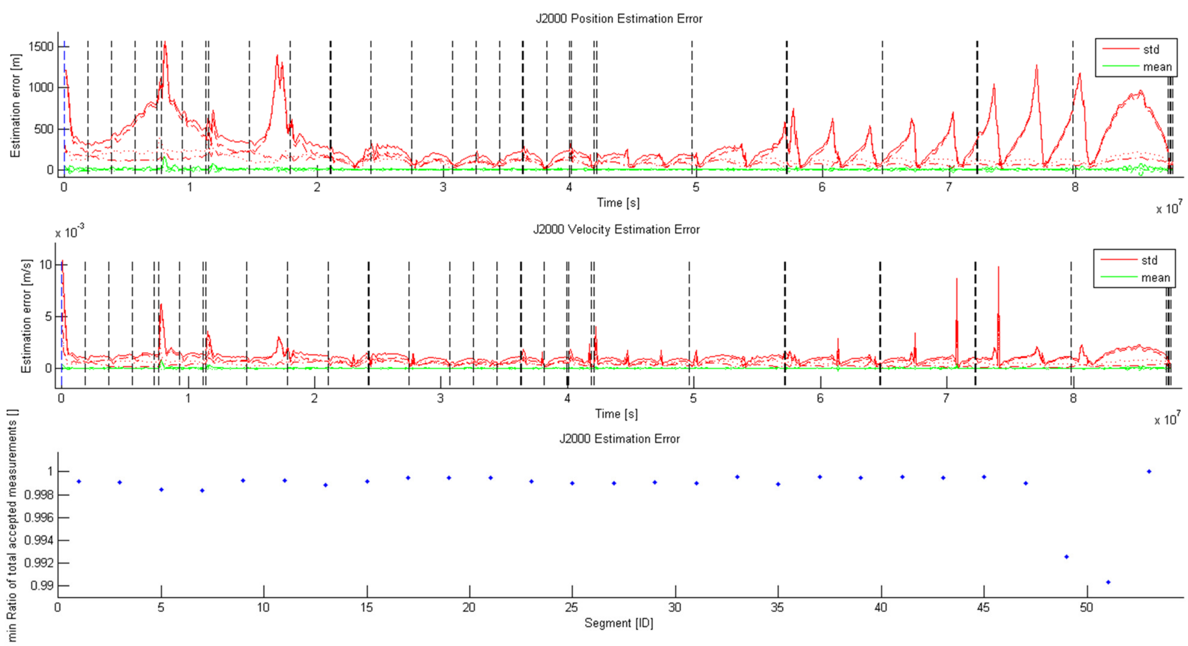

Figure 10-1 Deviations and mean for Example case 1 under nominal navigation assumptions

10.3 Example case: with multiple SP crossings

The behaviour of the closed loop system for this test case (solution in section 9.4.2) has been assessed with a range of alternative parameters controlling the guidance system behaviour. The objective is to assess stability of the solution (ie non-divergence from the reference trajectory) and the end to end DeltaV required. The trajectory is assessed in two sections which have different characteristics: LPO to SP1 crossing, SP1 to SP2 crossing

1. The LPO to SP1 section is predominantly far from Earth in LPO or very high HEO with high perigee (> 100000km). Closest Lunar encounter is > 100000km.

2. The SP1 to SP2 section is in a sequence of lower period HEOs with perigees as low as 30000km. Closest Lunar encounter is > 100000km.

10.3.1 Trajectory Section 1 analysis

10.3.1.1 Effect of trajectory biasing and correction manoeuvre direction

Analysis is performed with a long guidance time constant (time between correction manoeuvres) of 14days. As the trajectory lies generally far from Earth rates of divergence from the reference trajectory are relatively low. Firstly no biasing is applied. The cumulative DV is: 0.25

A plot is generated of number of manoeuvres with a component in the anti-Sun direction. The nominal number of manoeuvres for this section and guidance period is 18. As anticipated the manoeuvre directions show wide variability with typically half having an anti-Sun component.

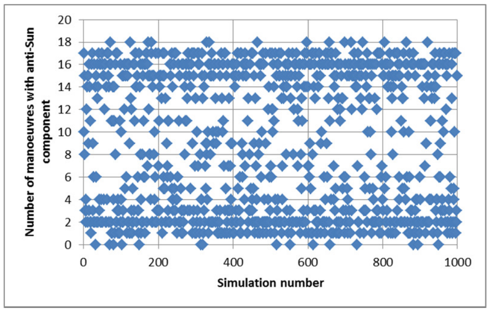

Figure 10-2 Number of manoeuvres with anti-Sun component vs simulation number for case 1 section 1 unbiased trajectory 14d correction interval.

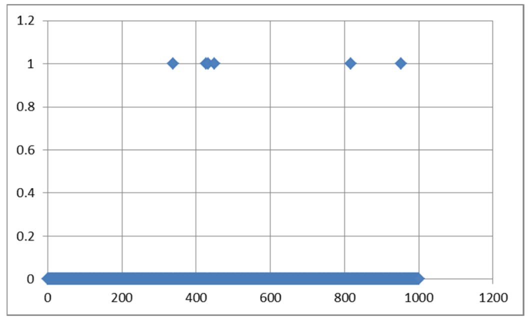

Figure 10-3 Number of manoeuvres with anti-Sun component vs simulation number for case 1 section 1 biased trajectory 14d correction interval - SRP A/M bias of 0.004

Trajectory design biasing can be applied to require a sunward acceleration to be applied over the trajectory in addition to the nominal deterministic manoeuvres. This is simulated as an additional SRP force equivalent to the design bias.

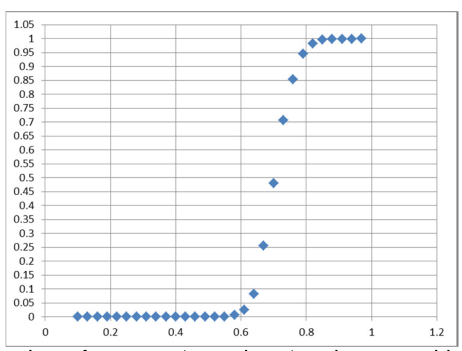

Figure 10-4 Probability density functions for case 1 section 1 trajectory with 14d corrections and SRP A/M bias of 0.004

Therefore for this class of transfer, providing an anti-Sun bias to the trajectory requiring rectification by the guidance system eliminates all manoeuvres with an anti-Sun component. The penalty is a DeltaV increase from typically 0.3 to 0.8 m/s (at 99.5% cumulative probability).

10.3.1.2 Effect of guidance correction interval

Reduction in the correction interval is examined. A 7 day case is used with nominally 86 correction manoeuvres. A trajectory bias is retained to ensure compatible correction manoeuvre directions.

Therefore the effect of higher frequency corrections is to increase the total correction DV to approx. 0.9 m/s. Also a greater number of manoeuvres with an anti-Sunward component are present, typically 2 being required.

10.3.1.3 Effect of accelerometer

Navigation accuracy can potentially be improved by addition of accelerometer data, using a generic high accuracy accelerometer model. In this example the effect of adding the accelerometer only has a marginal reduction in the total correction DeltaV, which also includes effects also from Thrust implementation and SRP environmental errors and trajectory biasing.

10.3.2 Trajectory Section 2 analysis

10.3.2.1 Effect of trajectory biasing and correction manoeuvre direction

Analysis is performed with an intermediate guidance time constant (time between correction manoeuvres) of 7days. As the trajectory now lies closer to Earth the rates of divergence from the reference trajectory are increased.

Firstly no biasing is applied. The cumulative DV is 1.25 m/s. The nominal number of manoeuvres for this section and guidance period is 86. As anticipated the manoeuvre directions show wide variability with typically half having an anti-Sun component.

Therefore for this class of transfer, providing an anti-Sun bias to the trajectory requiring rectification by the guidance system significantly reduces the number of manoeuvres with an anti-Sun component. Increasing the bias further reduces the number but a significant number remain. The penalty is a DeltaV increase from typically 1.25 to >2.1 m/s (at 99.5% cumulative probability).

Bias	DV at 99.5%
0	1.25
0.003	2.1
0.005	3.5

Table 10-2 Effect of bias on DV for transfer section 2

10.3.2.2 Closed loop trajectory characteristics

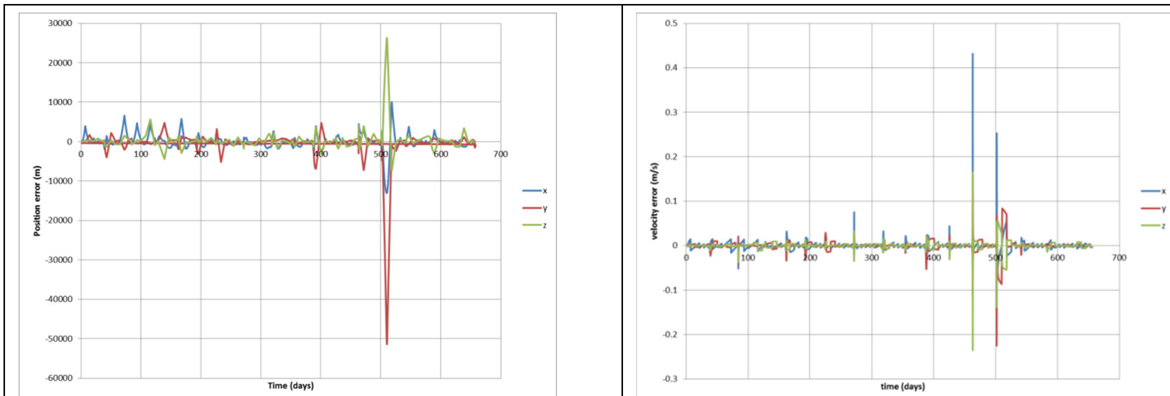

Figure 10-5 Position and velocity errors for case 1 section 2 biased trajectory 7d correction interval - SRP A/M bias of 0.003

The closed loop trajectory derived from guidance commands can be compared with the reference trajectory. The comparative position and velocity profiles are seen in the preceding plots. The largest excursions in the states occur after the lowest perigee passages (as low as 32000 km). Errors are amplified after the passage requiring greater subsequent correction. The extent of the excursion is related to the time after perigee passage at which the correction manoeuvre occurs.

10.3.2.3 Effect of correction period

Correction periods influence the DeltaV required. As correction intervals are increased the spacecraft can reach a situation where the trajectory divergence cannot be corrected with the available thrust.

With an 8 day period and SRP bias of 0.003 the correction at 99.5% is 1.85 m/s. For section 2 increases beyond approximately 8 days lead to divergence. Shorter periods increase the DV as assessed for the 4d case. The nominal number of manoeuvres is 156

Period	DV at 99.5%
8	1.85
7	2.1
4	3.1

Table 10-3 99.5% DV for a range of correction intervals, case 1 section 2, SRP A/M bias 0.003

10.3.2.4 Eliminating manoeuvres with anti-Sun component

The possibility to implement dog-leg manoeuvre pairs to eliminate anti-Sun manoeuvers. This can be demonstrated in Keplerian HEO, assuming operations away from post pericentre passages

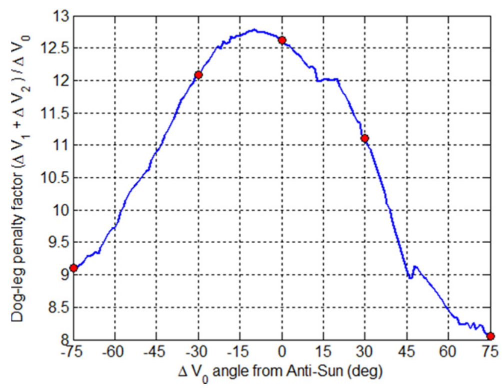

Figure 10-6 Nominal manoeuvre amplification to achieve manoeuvre pair with no anti-sun component

Whilst the amplification factor is high the number of manoeuvres affected is relatively small (a few percent of the total number). However the total effect is not insignificant in the total budget.

Alternative possible mitigation strategy is to implement differential gravity profile in the core algorithm to avoid guidance ‘over-corrections’ which lead to the anti-Sun corrections becoming necessary. This has the potential to avoid additional DV penalties

10.3.2.5 Effect of accelerometer

Navigation accuracy can potentially be improved by addition of accelerometer data, using a generic high accuracy accelerometer model. In this example the effect of adding the accelerometer results in a correction DeltaV reduction from approximately 2.1 to 1.8 m/s

10.4 Example case: Transfer to SP1 with closer Lunar flyby

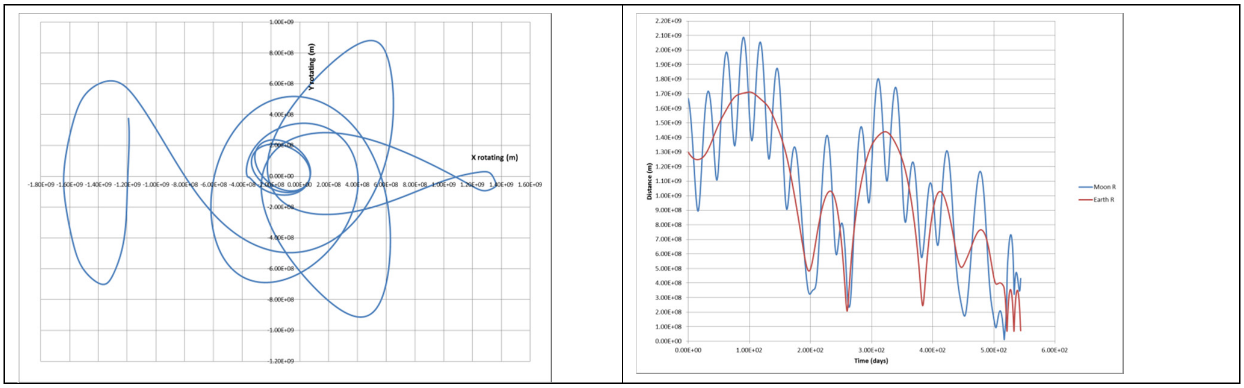

Figure 10-7 Trajectory in rotating frame and Earth and Moon ranges for case 4

The behaviour of the closed loop system for this test case has been assessed with a range of alternative parameters controlling the guidance system behaviour for the example of a trajectory having a medium distance Lunar flyby. The objective is again to assess stability of the solution (ie non-divergence from the reference trajectory) and the end to end DeltaV required. The trajectory characteristics are reproduced in the following plots

The Lunar flyby (at approx 520 days) is at a range of approximately 11000km.

This trajectory has two distinct phases:

- distant from Earth and Moon (until approx 500 days)
- close Lunar flyby and lower Earth perigees: 500-540 days

The nominal guidance interval adopted was therefore split:

- 7 day correction interval for times < 416 days (where a small deterministic manoeuvre is made)
- 1.5-2.5 day correction interval for times > 416 days

Trajectory biasing is again applied to reduce the number of manoeuvres with an anti-Sun component. The baseline case is therefore the following: Manoeuvre intervals: 7day and 1.9 day - SRP area/mass bias: 0.006. Nominally 98 manoeuvres are required

The number of manoeuvers with an anti-Sun component after biasing is a number between typically 4 and 8. The 99.5% correction DeltaV is 2.45 m/s.

The error in velocity shows a peak close to the Lunar flyby. The relatively high frequency correction (2d) in this region ensures that in general the trajectory error is low.

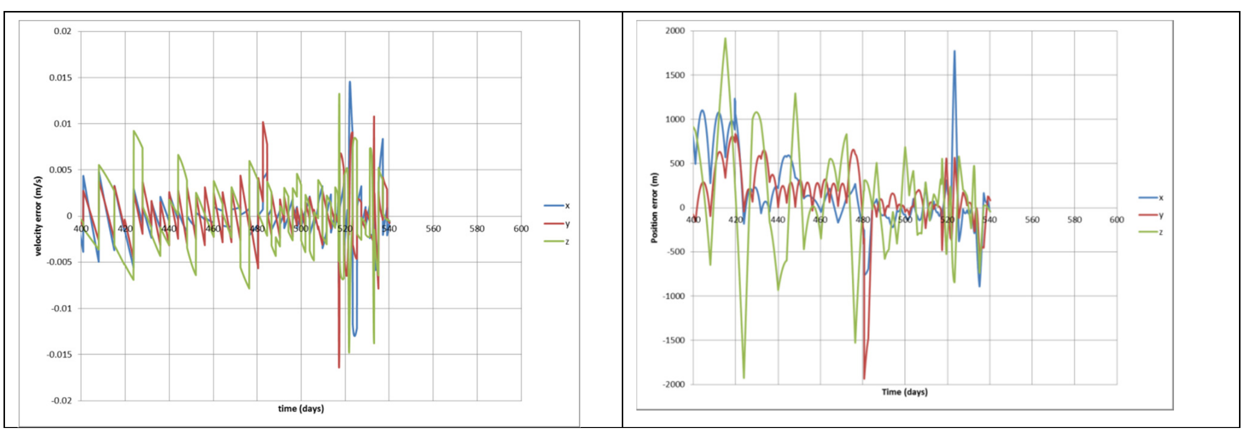

Figure 10-8 Trajectory error evolution for case 5 trajectory with 7d/1.9d corrections and SRP A/M bias of 0.006

10.4.1 Influence of correction manoeuvre thrust

The effect of thrust on the correction was examined, as shown in the following table. The table indicates that the DV is weakly dependent on Thrust in the range considered.

Thrust	DV at 99.5%
1000	2,45
500	2.65
250	2.65

Table 10-4 DeltaV dependence on correction manoeuvre thrust for case 5 trajectory with 7d/1.9d corrections and SRP A/M bias of 0.006

11 NAVIGATION REQUIREMENTS

Based on the observations for the cases analysed above some recommendations on how to handle the navigation for the low thrust transfer can be made. These recommendations are phrased as requirements (or goals) in the classical ESA formalism.

11.1 Ground station accuracy

The required navigation performance to ensure successful completion of the transfer and regular trajectory control manoeuvres is in the order of several kilometres, which is therefore in accordance with the above performance figures, even using only one station and the worst one, as well as worst case hypotheses on the environment errors. The convergence is fast enough that this performance is reached within a few days which is also sufficient to ensure trajectory control manoeuvres.

Ground station performance, as it is, is sufficient even in presence of large environment errors. The latter being the driving parameter, a requirement would anyway have to be formulated to reduce them before addressing the ground station measurement performance but even such requirement is not needed.

This means that from navigation point of view low, **the ultra low thrust transfer to saddle point is feasible with current ESTRACK capability** without requiring any technology modification or additional complexity like the use of ΔDOR, and this is a major outcome of the feasibility study.

11.2 Ground station use

REQ-NAV-GSU-01: Estimation convergence and ground station availability

The S/C station Doppler and Range measurements shall be available during at least 3 days (goal 5 days) with no interruption other than station non-visibility before any commanded manoeuvre.

Rationale: this is the observed convergence time of the navigation below errors of the order of the few km.. In practice it may be possible to have slight interruptions more than just for visibility, to be finely analysed on a case by case basis. Long interruptions away from manoeuvres do not seem to matter much for performance (ex: interruption of almost one week) which could help relax ground operations. However, lack of control manoeuvre for an excessively long timespan, especially near the endpoint, may cause an irrecoverable trajectory so the maximal duration of ground operation interruption ought to be consolidated considering the whole GNC and not just the navigation.

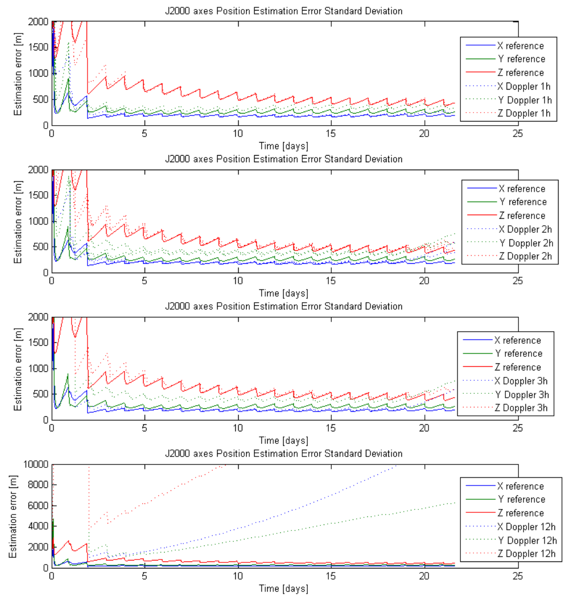

Figure 11-1: Doppler measurement frequency sensitivity analysis

REQ-NAV-GSU-02: Measurement density

For phases where the navigation is active (required before a manoeuvre and station in visibility) the following measurements shall be made available:

- Range: at least 1 per pass (i.e. at least 1 per 24h)
- Doppler: at least 1 per hour

Rationale: The range is essential but its frequency has low impact : minimal frequency of one per 12h is equivalent to one per 24h given the visibility duration is less than 8h every 24h. Same analysis was carried out for Doppler on Figure 11-1, justifying the 1 hour figure. For 2h, the performance start degrading noticeably and above 12h the estimation becomes divergent. Doppler frequency also impacts convergence time.

REQ-NAV-GSU-03: Navigation initialization

The navigation shall be initialized with conditions ensuring no excessive divergence of state before the first measurement is available, using one of the two following solutions:

- Initialisation at the time of the first measurement
- Initialisation with conditions which integrated from initial date to the time of the first measurement lead to dispersion significantly lower than 1,000,000 km.

Rationale: Initializing the navigation with poor conditions can lead to cases too challenging for the navigation robustness, as was evidence through the study. This requirement may therefore be useful to mention.

REQ-NAV-GSU-04: Earth fly-by GNC

For each Earth fly-by:

- A targeting control manoeuvre shall be carried out at the latest at C/A-3.5days
- A clean-up manoeuvre shall be carried out at the latest at C/A+3.5days
- RF measurements shall be maintained during the fly-by

Rationale: A 7-day (roughly) interval for corrections was found by the Guidance and Control team (using the navigation model) to be the upper limit for proper transfer completion: above this duration, the trajectory can become irrecoverable (assuming 1mN thrust). This requirement is important for feasibility and therefore *mentioned here even if it is not taken from pure navigation considerations. As both the fly-by targeting and clean up are important, they are positioned symmetrically in first approximation around the flyby C/A.*

The JUICE studies evidenced that very good observables can be gathered during the gravity assist itself, hence the use of a steerable MGA to maintain the link for Jovian moons flybys. The same logically applies to ULTT, and navigation during C/A allows a clean-up shorter after C/A and also more accurate.

REQ-NAV-GSU-05: Moon fly-by GNC

For each Moon fly-by:

- A targeting control manoeuvre shall be carried out at C/A-1day.
- A clean-up manoeuvre shall be carried out at C/A+1day.
- RF measurements shall be maintained during the fly-by

Rationale: Same as above, taken from the Guidance and Control team analysis on moon fly by, evidencing a maximum interval to ensure control at around 2 days.

REQ-NAV-GSU-06: Station choice for transfer

The transfer navigation, excluding terminal saddle point delivery, shall be done from the Cebreros station.

Rationale: The performance appears sufficient for one station, even Cebreros which does not always have the best performance. It is therefore preferred for purely logistical reason, as performance does not justify superseding the logistical constraint.

REQ-NAV-GSU-07: Station choice for terminal Saddle Point delivery

The terminal navigation should be done using the Cebreros, Kourou, and Perth stations.

Rationale: Goal to ensure the best possible performance and maximal science return. Use of a fourth station is not really worth the cost, all the more so since Maspalomas and Cebreros are close geometrically and should not be used together. Between the Kourou+Perth+Maspalomas and Kourou+Perth+Cebreros solutions, the latter is preferred, as it makes sense to keep Cebreros which monitors all the previous transfer.

11.3 Use of on board assets

REQ-NAV-UOBA-01: Antenna use

The navigation should be supported by MGA whenever possible. When MGA is not available, the navigation shall be supported using the LGA.

Rationale: Self-explanatory. The MGA gives slightly better performance and the better SNR will be easier to process on ground. However the LGA can be sufficient when the MGA is not available (which necessarily occurs with a non-steerable MGA) and no stronger requirement on other antennas or steering capability is needed. This also means the ULTT would be feasible from this point of view with the current LPF design.

REQ-NAV-UOBA-02: SRP calibration

SRP errors shall be calibrated out as much as possible. SRP should be calibrated to a residual error below 1%.

Rationale: The requirement is self-explanatory. The goal is taken as feasible (less than 0.1% actually on SRP modelling studies for GNSS satellites) and sufficient to remove most of the effect on the estimation

REQ-NAV-UOBA-03: Accelerometer use

If the SRP calibration expectation is not better than accelerometer measurements for the mission profile, the SRP should be calibrated using accelerometer measurements at the beginning of each new navigation arc.

Rationale: This implies to use only a punctual update of the SRP estimate by the accelerometer rather than continuous measurements and should be sufficient since the SRP varies little over the horizon of a few days (an update could also be made 3-5 days before a manoeuvre for optimal estimation). This is meant to take advantage of the accelerometer without excessive operational burden. This is a goal to improve the estimation and not mandatory as performance can be met even with large SRP/environment errors.

Overall there is no strong or driving requirement on the navigation, which suggests that at this level completion of an ULTT is perfectly feasible.

12 SUMMARY

A set of tools have been developed to enable assessment of the Feasibility of Ultra Low Thrust Transfers in regions of multi-body gravity field influence. Aspects include:

- Trajectory design/optimisation
- Error analysis and stochastic manoeuvre budgets

Navigation analyses have been performed to understand the solution sensitivity to key input assumptions.

A case study was performed to understand the performance that could be achieved. The case study has focussed on the LPF mission potentially extended from its initial LPO to achieve multiple SP crossings. Key issues to be addressed were:

- Developing transfers with low deterministic DeltaV
- Developing transfers with low stochastic DeltaV
- Establishing the feasibility of performing corrections when flybys and/or lower pericentres are present.
- Obtaining sufficient accuracy in the critical mission pahses: Saddle Point crossings for LPF

Low deterministic DeltaV transfers enabling multiple SP crossings were found. DeltaVs of 1-3 m/s resulted, depending on transfer duration. Longer transfers allow lower DeltaV. A first SP crossing within one year is feasible with subsequent crossings at 1.5-2 year intervals.

Navigation analysis and the inclusion of a closed loop guidance and navigation model in simulations has shown that the stochastically generated correction DeltaV (at 99.5%)is typically 2-3 m/s. Higher DeltaVs are needed when the trajectory flys closer to the Moon or Earth.

If manoeuvre direction constraints are present (eg no anti-Sun component is allowed) then trajectory biasing is required, which also increases the total stochastic DeltaV. The above figure includes such biasing. The absence of any such constraint would allow DeltaV reduction of up to half.

A set of requirements have been derived regarding space and ground segment systems needed to implement such transfers

13 DOCUCUMENT CHANGE DETAILS

ISSUE	CHANGE AUTHORITY	CLASS	RELEVANT INFORMATION/INSTRUCTIONS
1.0			Initial Issue

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