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INITIAL NAVIGATION ANALYSIS FOR THE EUROPA MULTIPLE FLYBY MISSION CONCEPT

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Earth and spacecraft-based observations of the Jovian moon Europa have identified it as the most plausible habitat for extraterrestrial life in our solar system. Recently, NASA has formed a Europa Mission Concept to potentially explore this icy world with a sophisticated instrument suite operating from a spacecraft in orbit about Jupiter. Candidate trajectories have been designed that would use the Jovian moons to repeatedly bring the spacecraft near Europa, providing multiple observation opportunities over the mission duration. This paper describes navigation analyses associated with these trajectories that are being assessed for their operational feasibility. The analysis includes determination of the ΔV requirements for the mission concept and notional spacecraft ephemeris knowledge capability.

INTRODUCTION

The Europa Mission Concept, designed to explore the Jovian satellite Europa using multiple flybys, was approved by NASA for Phase-A studies in 2015. Multiple satellite flyby tours have been successfully used to probe the Jovian and Saturnian systems, with the Galileo and Cassini missions, respectively.[1],[2] The Europa Project trajectory would use the multi-body flyby technique to build up a series of globally distributed ground tracks, enabling Europa science, while avoiding the high radiation dose, added mission risk and complexity, and large fuel penalty associated with bringing a spacecraft into an orbit about Europa itself.[3] The spacecraft would carry a payload designed specifically to assess habitability with such a series of flybys. The selected payload includes both *in situ* and remote sensing instruments. A previous article describes a navigation concept that is relatively operationally simple, robust and likely to meet the ephemeris uncertainty needs of the science instruments.[4]

An update to the mission concept study analysis taking into account the developing flight system and using a more recent candidate tour trajectory is presented here. There are some differences between the spacecraft system assumed in the study in Ref 4 and the flight system baseline for the present Europa Mission concept that impact navigation characteristics. The chief difference is that the Europa project is baselined to be solar powered, which from a navigation standpoint, changes expected non-gravitational forces on the spacecraft and also its maneuver execu-

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tion characteristics. A second notable difference is the baselining of a propulsion system comprised of a set of eight redundant 22 N thrusters with significantly different execution performance. The results in this paper document a current snapshot of the expected navigation performance of the Europa Mission Concept during its Jovian Tour phase.

The navigation analysis discussed in this paper supports a reference trajectory composed of a series of elliptical orbits about Jupiter with a perijove near the radial position of Europa, a typical period of 14 days, Europa flyby distances as close as 25 km, and typical flyby velocities of about 4.5 km/s. The navigation concept baselines Earth-based radio tracking of the spacecraft with an 8 hours on 8 hours off cadence, matching expected required data downlink needs. During the typical transfer, three maneuvers would be performed, two deterministic maneuvers that together target the following flyby conditions and one purely statistical maneuver (approach maneuver) at 3 days prior to each flyby to reduce accumulated errors due to dispersion in previous maneuvers, tracking data noise and deficiencies in force models. A deterministic maneuver is one that has a non-zero value in the reference trajectory, and a statistical maneuver, while not present in the reference trajectory, is a placeholder to manage dispersions. Finally, additional tracking after the approach maneuver would be used to improve ephemeris knowledge enabling more demanding instrument pointing. Expected ephemeris delivery knowledge at the flybys and associated ΔV for the tour is presented.

REFERENCE TRAJECTORY DESCRIPTION

The reference trajectory used with this analysis, named 15F10-S22, is associated with a projected launch in 2022 and arrival in the Jovian system in 2025 via a direct interplanetary transfer, as would be achievable with a launch on the NASA Space Launch System (SLS). As in the mission proposal phase, the mission concept currently carries a corresponding secondary reference trajectory, 15F9-A22, associated with a projected launch on an Evolved Expendable Launch Vehicle (EELV) and able to reach the Jovian system in 2028 after an interplanetary phase including Earth and Venus gravity assists. Along with launch year and launch vehicle denotation, the trajectories names also include their development year and a sequence number. Both interplanetary trajectories connect to a generally similar tour, albeit with differences associated with their Sun-Earth relative geometry associated with the projected arrival year. The structure of the tour and the distribution of encounter geometries is similar to an earlier trajectory (14F8-S22) which was detailed in Lam et al.[5] The 15F10-S22 trajectory, shown in Figure 1, includes 42 targeted Europa encounters, 5 targeted Ganymede encounters and 7 targeted Callisto encounters. The targeted Europa flybys are at altitudes ranging from 25 km to 6565 km, with many below 100 km. The majority of the Europa science opportunities would be available during the flybys occurring within 14 day orbits of Jupiter (COT-1, COT-2, COT-3 and COT-4 in Figure 1). Crank-over-the-top (COT) refers to a series of orbits about Jupiter in a phase resonance with a satellite, in this case Europa, resulting in successive close flybys that would each change the inclination of the spacecraft orbit about Jupiter. COT-1 and COT-2 tour phases would produce Europa encounters on the anti-Jovian side of the satellite, while COT-3 and COT-4 produce Europa encounters on the sub-Jovian side of the satellite (Europa is tidally locked with Jupiter). Between COT-2 and COT-3, several equatorial orbits, labeled Petal-Rotation, provide encounter geometry particularly supporting gravity science. The targeted flybys of Ganymede, early in the tour, would be used to assist capture into the Jovian system and reduce the capture orbit period (pump down phase in Figure 1), while the Callisto flybys are used along with Europa flybys to transition the spacecraft orbit from producing Europa closest approach points on the anti-Jovian side of Europa to producing closest approach points on the sub-Jovian side. While the Ganymede and Callisto encounters are not mission concept science targets, they must be equally assessed for navigational feasibility as

the targeted Europa encounters. In this paper, an orbit about Jupiter will be referred to as a 'petal', while the duration of trajectory from one targeted close satellite flyby to another is called a 'transfer'.

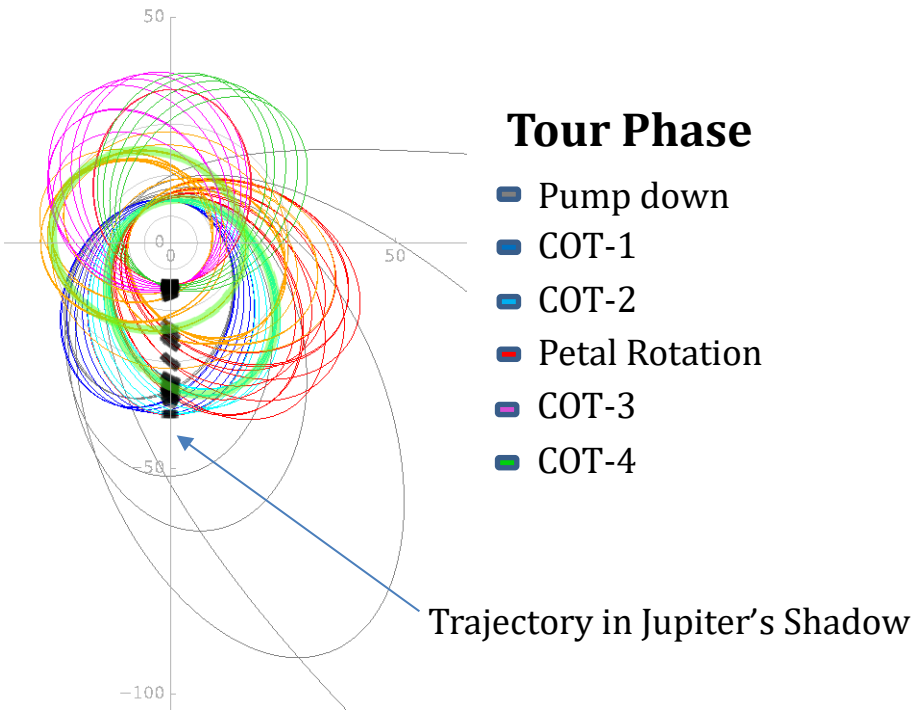

Figure 1. Schematic of the 15F10-S22 tour trajectory showing projected tour subphases viewed from Jupiter's northern hemisphere, centered at Jupiter in its rotating frame with the y-axis pointing to the Sun, z-axis aligned to Jupiter's north pole, and x-axis completing an orthogonal set (credit: Try Lam).

NAVIGATION CONCEPT

The in-flight behavior of a spacecraft will always diverge from a planned reference due to both uncertainties and simplifications in the physical models used, and also unexpected events in flight. A robust navigation concept will assure that a reference trajectory is feasible in the presence of the uncalibrated portion of non-gravitational forces on a spacecraft, uncertainties in the gravity of natural bodies, and dispersions in the execution of maneuvers required to implement the reference trajectory. Feasibility, in this context, refers to the ability of a plan to be likely to meet mission science and safety objectives.

During the notional tour phase of the Europa Mission Concept, the close satellite encounters would be both of prime interest for science instruments and most stressing in terms of trajectory safety. Given those characteristics, the concept that has been developed for tour navigation is to return the spacecraft to a reference trajectory at the time of targeted flybys, allowing the spacecraft position to deviate from the reference between targeted encounters. Since each tour petal contains one or two deterministic maneuvers in the time period between the early encounter and apojove, the key implementation would be to place a purely statistical maneuver, called the approach maneuver, targeting the upcoming encounter at three days prior to the encounter. The placement at three days before an encounter allows time to determine the spacecraft position in

the presence of dispersions from the earlier maneuvers and correct accumulated orbit determination (OD) errors, while close enough to the upcoming encounter to limit error growth from dispersions in the approach maneuver itself. The desire to limit engineering activity near the encounter period, would constrain the plausible placement of the approach maneuver on one side, while the observability of the dispersions from the earlier maneuvers in the petal using Earth-based tracking constrains the location on the other. While this strategy is similar to that employed in the Cassini tour of Saturn, the timelines would be shorter, making it more challenging to implement.[2]

Away from encounters, it would still be imperative to maintain ephemeris knowledge to the level required to acquire spacecraft signal from Earth at any time, support science performed away from encounters, and design planned maneuvers. The mission current baseline to achieve this goal is ground-based OD using Earth-based radiometric Doppler and range tracking data. In the present mission concept, Doppler measurements would be readily available with a likely cadence of eight hours on and eight hours off as a byproduct of telecommunications activity for science and health data playback, making it a convenient data type. Range data could be collected at the same cadence, but with impacts to telemetry return that are to be assessed. As pointing the telecommunications antenna to Earth while simultaneously providing science instrument point could be infeasible during the satellite closest approaches, the navigation baseline assumes no tracking during periods within 12 hours of encounters. Finally, in the present flight system design, there is no engineering instrumentation to support collection of optical (or other) target-relative navigation data. While this imposes practical limitations on determining target-relative location of a spacecraft, the selected instrument suite was chosen with the expectation that science objectives could be met with the capability of a ground-based navigation system.

NAVIGATION CONCEPT ANALYSIS

Overview

The navigation analysis includes two components, used in concert. The first uses an OD covariance study to predict ephemeris uncertainties for varied tracking assumptions and modeling uncertainty levels for a reference trajectory. The second uses a Monte Carlo method to compute expected dispersions in deterministic and statistical maneuvers in the reference trajectory given a modeled maneuver execution behavior and expected ephemeris knowledge along the reference trajectory.[6] The statistic of interest for use in fuel mass allocation is $\Delta V99$, the value for which 99% of the dispersed trajectories use less ΔV. The Monte Carlo results provide the mission $\Delta V99$, individual maneuver $\Delta V99$, and a maneuver covariance that may be used in the OD covariance study. Typically, the injection of the maneuver covariance statistics into the OD covariance study initially results in larger ephemeris uncertainties, and the process must be iterated until both ephemeris errors and maneuver statistics converge. Both the Monte Carlo-based maneuver analysis and the OD covariance study presented here were performed using the Mission Design and Operations Navigation Toolkit Environment (MONTE) software developed at JPL.

Orbit Determination Analysis

The function of the OD analysis is to predict the expected uncertainty in the spacecraft ephemeris as a function of time, given errors in dynamical models, tracking data schedule, and tracking data errors. We employ a covariance study in which the state partial derivatives of dynamic model parameters and measurement data are calculated about their nominal values and used to predict the impact of errors in those parameters and data on the ephemeris. The nominal values of the model parameters are matched closely with those used in producing the reference, and they match exactly those used in the maneuver analysis discussed below. The two main results of the analysis are a time history of expected ephemeris knowledge to be used as an input to the maneuver $\Delta V99$

study, and uncertainties in ephemeris mapped forward from times that solutions may be available to times of interest for instrument observations. In general, observing scenarios are sensitive to both delivery, the expected error compared to a reference trajectory, and knowledge, the expected error compared to the last OD ephemeris available at the time of an observation.

The parameter error assumptions used in this analysis are listed in Table 1. In addition to parameters of modeled forces, an extra time-varying acceleration error term is included to account for the impact of un-modeled sources of small or unpredictable error such as solar radiation pressure, uncalibrated attitude control thrusting or momentum wheel desaturations and higher order gravity fields of the satellites and Jupiter. The OD process simultaneously estimates the ephemeris of the spacecraft, the system natural body ephemerides, some orientation and gravity parameters, and maneuvers. The contribution of Earth platform parameters such as DSN station locations, troposphere and ionosphere and Earth orientation (which are applicable when Earth-based tracking is used) are included in the filter as consider parameters at standard values.

Table 1. Tour Baseline Orbit Determination Covariance Assumptions

Tracking Data Schedule	Every other 8 hour pass
Doppler data weight	0.1 mm/s for 60 s sampling for Sun-Earth-Probe (SEP) angles higher than 15°; 1 mm/s between 7.5° and 15°; 5 mm/s below 7.5°; no data used below 3°
Range data weight	3 m for SEP angles higher than 7.5°; no data used below 7.5°.
Tracking exclusion near encounters	No tracking data collected from 12 hours before to 12 hours after the flyby
Un-modeled accelerations	4.5×10^{-6} mm/s^2 per axis estimated stochastically as white noise in 8-hour batches
Maneuver design data cutoff	2 days prior to maneuver

For convenience, the OD analysis is performed in arcs starting some time before the projected approach maneuver for a targeted encounter and ending after the following targeted encounter. Simulated tracking data during the arc is used to perform orbit determination for the design of the maneuvers that are targeted to the second encounter. The covariance of the planetary and natural satellite parameters from an arc are used to initialize the ensuing arc to simulate the effect of improved knowledge of the bodies over time. For this study, the covariance of the satellite ephemerides is scaled to produce an error of 10 km for Callisto at the start of the tour. At that value, the satellite ephemeris error dominates the encounter delivery error, but the satellite ephemeris uncertainties reduce to below other error sources after 2 encounters of a given body.

In order to understand the impact of OD errors on the design of an upcoming maneuver, it is useful to inspect the time evolution of the state error mapped to the maneuver target. Most of the tour maneuvers are targeted to the B-plane of the upcoming encounter (see Appendix 1 for a definition of B-plane geometry). Figure 2 shows the time evolution of the OD error for a typical tour transfer. The error associated with the execution of a maneuver is only included in the mapped error after the maneuver epoch, so state error improvement from additional tracking is not masked by the sometimes large downstream error contribution of each maneuver. This mapped view is useful for determining a favorable data cutoff time (DCO) for the OD employed in a maneuver design.

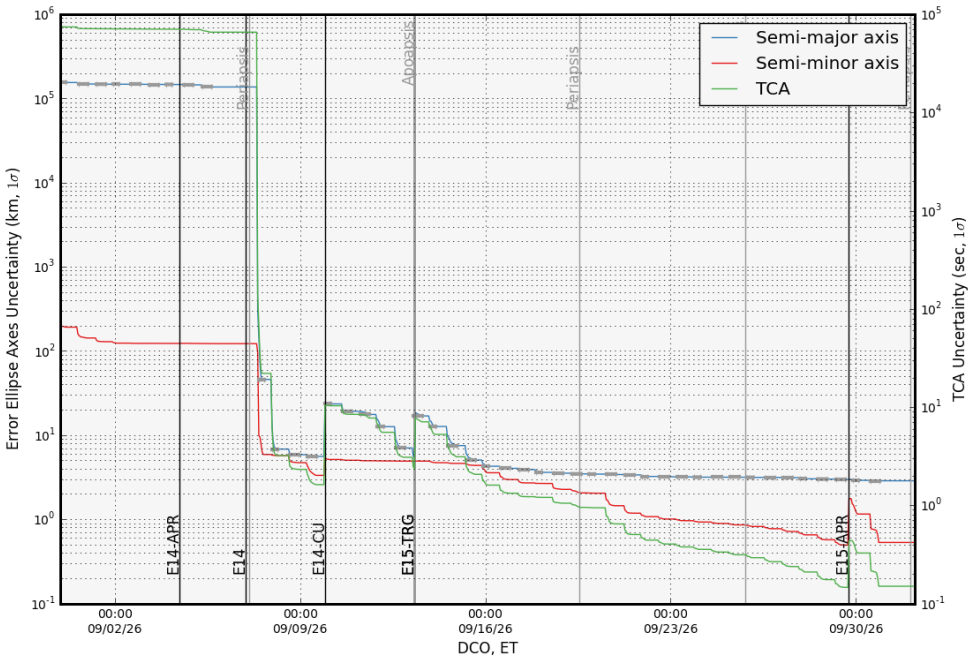

Figure 2. The time evolution of spacecraft state error mapped to the targeted encounter B-plane vs data cutoff time. The semi-major component of error is plotted in blue, semi-minor in red, and error in closest approach time is shown in green (with its y-axis on the right).

The spacecraft state uncertainty at any selected time can be mapped to times and coordinate frames of interest to determine ephemeris delivery and knowledge errors of use to compare navigation capability margin against instrument observation requirements. It is expected that the most stringent observation requirements for both delivery and knowledge during the tour phase will be at the time of the targeted satellite encounters. Since many encounters are at altitudes of 100 km and lower (several at 25 km), even modest ephemeris errors may be a challenge for observations with remote sensing instruments. Figure 3 compares the component of ephemeris knowledge in the body-altitude direction at the time of encounter for all 42 Europa encounters for three cases of interest. The first two cases show the the OD knowledge at the DCO for the approach maneuver alone and with the inclusion of execution from the approach maneuver. The last case shown is OD knowledge with the inclusion of a day of tracking after the execution of the approach maneuver. It can be seen that for the majority of the encounters, the driving delivery error is the execution dispersion of the approach maneuver. For science observations that can benefit from ephemeris knowledge alone, using some tracking after the approach maneuver is beneficial.

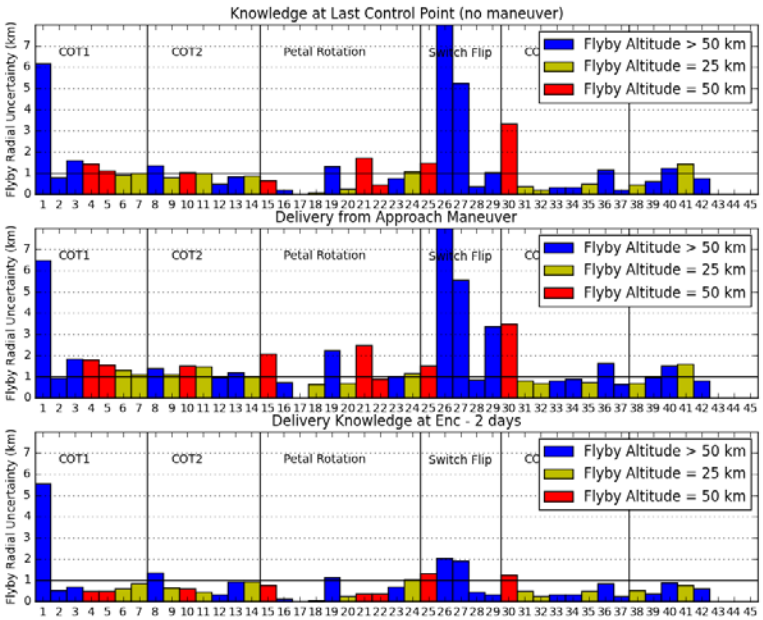

Figure 3. Encounter Altitude Uncertainty at encounter time for three OD solutions: a) OD knowledge at approach maneuver design DCO, b) Delivery error with maneuver execution error included, c) OD knowledge at two days prior to encounter with post maneuver tracking (and one day post maneuver)

Maneuver Analysis

The primary goal of a maneuver analysis is computing ΔV statistics, and thus the expected mission fuel requirement. As a necessary first step, an experience-based strategy is used to determine statistical maneuvers and adjust deterministic maneuver placement and targeting strategy. Judicious placement accounts for the time of dynamic events that tend to magnify dispersions in the spacecraft ephemeris, such as close flybys and large maneuvers, as well as accounting for expected tracking schedule. Placement variations are assessed to find a suitable strategy. Many of the Europa transfers occur in a 4:1 resonance with the orbit of Europa about Jupiter, resulting in a 14 day orbit period which lends itself to placement of the deterministic maneuvers at 11 and 7 days before encounter, and a statistical maneuver at 3 days prior. Some transfers, used to shape the tour, have different durations and deterministic maneuver ideal locations, and so require different treatment.

Once a maneuver location has been determined, a Monte Carlo analysis is performed starting with initial states from a sampled state covariance at a time prior to the maneuver. A maneuver is searched for each sample using the implemented targeting strategy. Finally, an execution error model is sampled to provide the final distribution of that particular maneuver. The two main inputs to the Monte Carlo analysis are the engine error model, which is the input to vary the values of maneuver samples, and the expected DCO for ephemeris knowledge used to design the maneuver, which determines the distribution of initial states for the Monte Carlo. For the purposes of this study, maneuver execution errors were modeled parametrically with a Gates Model.[7] The Europa Mission concept flight system is currently being designed to perform maneuvers by slewing to the maneuver attitude and firing a set of four 22 N thrusters for maneuvers less than

0.21 m/s or eight 22 N thrusters for larger maneuvers. Gates Model parameter values characterizing the thrusters within the current flight system design for both the execution modes was provided by the guidance and navigation team as input to this analysis (Table 2). These values assume some in-flight calibrations to be performed during the interplanetary mission phase. This system predicts much larger execution dispersions for very small maneuvers than the analysis in Ref 4. The DCO used to select the initial state covariance was either one or two days prior to the execution time of the maneuver, where one day prior is the baseline case, and two days prior is a margined case examined to check sensitivity. The margined case retains a one day DCO for the approach maneuver.

Figure 4 shows the expected ΔV99 accumulation through the 15F10-S22 tour along with the deterministic ΔV for the baseline case (one-day DCO for all maneuvers) and a margined case (two day DCO for non-approach maneuvers). In some cases, substantial improvement in maneuver dispersion results from placing the design DCO closer to the maneuver, however, in many cases, the performance is similar. Although allowing two days between the OD DCO and maneuver execution improves operations margin, previous analysis shows that there is a substantial cost to using that margin for approach maneuvers, so we have not considered the case here.

Table 2. Maneuver Execution Error Model (One-sigma)

		8x22N Configuration	4x22N Configuration
Magnitude	Fixed Error (mm/s)	4.67	4.67
	Proportional Error	0.33%	1.00%
Pointing	Fixed Error (mm/s)	3.33	3.33
	Proportional Error (mrad)	6.67	6.67

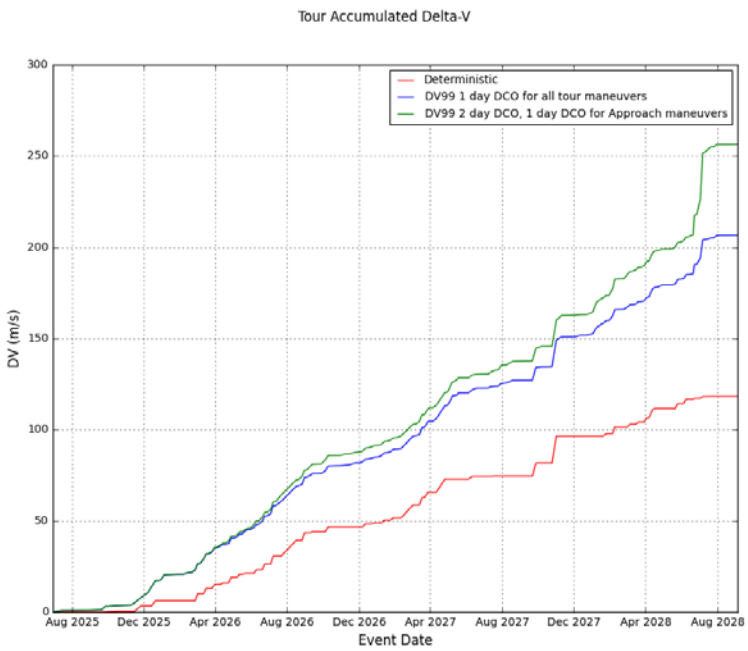

Figure 4. The cumulative ΔV in the reference trajectory is shown in red as a function of time in the tour. The blue curve shows the cumulative ΔV99 when all maneuvers are designed with OD solution DCO one day prior to maneuver execution. The green curve shows the cumulative ΔV99 when approach maneuvers are designed with OD solution DCO one day prior to maneuver execution, while others are designed based on OD solutions with DCO of 2 days prior to the maneuver. The total accumulations are 118 m/s deterministic, 206 m/s baseline case (blue), 256 m/s margined case (green).

CONCLUSION

Employing a ground-based navigation strategy using Earth-based radiometric tracking with the current propulsion system execution error model, it is feasible to baseline providing 1 km, at one-sigma, radial position knowledge at most targeted Europa encounters for a statistical ΔV budget on par with the deterministic cost of the tour. However, the strategy would require tracking and an uplink later than the approach maneuver, possibly creating an operations challenge and would leave little margin against the guidelines provided to instrument proposals. Since the mission is in Phase-A, the observation requirements are developing and margins against this navigation strategy are still to be assessed. Achieving a leap in navigation performance, such as across the board improvement of 50% or more, may require either a strategy employing either additional, ideally target relative, tracking observables or a significant reduction in maneuver execution dispersion. Reducing the maneuver execution dispersion via an on-orbit calibration strategy, was employed during the Cassini tour with good results, and may restore margin to the present strategy after the early part of the tour.[8]

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APPENDIX 1: B-PLANE GEOMETRY

Figure 5 depicts B-plane coordinates. The B-plane is the plane perpendicular to the asymptote of the incoming trajectory. The B-vector is defined as the vector joining the body center and the point where the asymptote meets the (perpendicular) B-plane. Note that physically, the B-vector indicates the closest point if the body had no mass. Three coordinate vectors are then defined, with S along the incoming velocity, T lying in the ecliptic plane, and R completing the triad. Using this geometry, the target point is described by the R and T components of the B-vector, B • R, B • T. The B-plane error is expressed in terms of those quantities, and the time of flight.

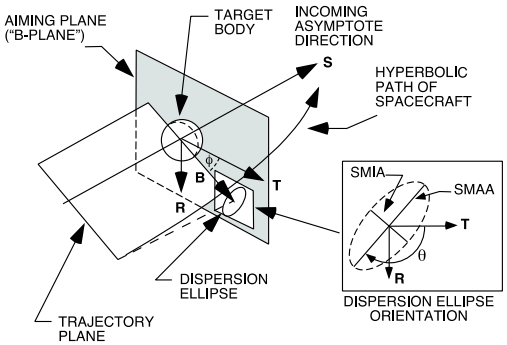

Figure 5. B-plane geometry representation.

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