Bepi Colombo Orbit Determination during Extended Electric Propulsion Manoeuvres

Gabriele Bellei[(1)], Frank Budnik[(2)]

[(1)]DEIMOS Space, located at ESA-ESOC
Darmstadt, Germany
Email: gabriele.bellei@deimos-space.com

[(2)]European Space Agency
Darmstadt, Germany
Email: frank.budnik@esa.int

***Abstract* – The Orbit Determination of an interplanetary orbiter subject to extended Solar Electric Propulsion manoeuvres presents several challenges, as the disturbance of the propulsion force causes degradation of the radiometric observations, particularly Doppler, while the dynamic modelling of the system is complicated by the irregular evolution of the thrust acceleration, characterised by transients and frequent and unpredictable short interruptions. In the case of Bepi Colombo, this is further complicated by strong guidance constraints and a higher level of thruster degradation than anticipated. The purpose of this paper is to present a retrospective of how the operational Orbit Determination strategy evolved from the early cruise approach, mainly relying on Doppler data and planned thrust interruptions, to the current stage where line-of-sight and ΔDOR measurements are used in combination with telemetry-enhanced dynamic modelling.**

I. Introduction

BepiColombo is a European Space Agency (ESA) cornerstone mission to Mercury which is being conducted in co-operation with the Japan Aerospace Exploration Agency (JAXA) [1]. The mission is composed of two spacecraft, the European Mercury planetary Orbiter and the Japanese Mercury Magnetospheric Orbiter (MIO) that will orbit Mercury independently. After a successful launch in October 2018, the mission has been since in heliocentric cruise in a configuration called Mercury Composite Spacecraft (MCS) that comprises, in addition to the two orbiters, the Mercury Transfer Module (MTM), providing Solar-Electric Propulsion (SEP) and all services not required in Mercury orbit, and the MIO Sunshield and Interface Structure (MOSIF), acting as a thermal protection and a mechanical and electrical interfaces for the MIO [2].

At the time of writing a total of 6 planetary flybys (one at Earth, two at Venus and three at Mercury) and 18 Solar Electric Propulsion manoeuvres have been completed. The arrival at Mercury is foreseen in December 2025, after three additional Mercury flybys, eight deterministic SEP manoeuvres, with a weak gravitational capture and a sequence of chemical propulsion burns. A view of the heliocentric trajectory can be seen in Fig. 1, with the thrust arcs marked in red; additional details on the launch, the cruise main events and the early navigation operations can be found in [3].

The Solar Electric Propulsion system is key to reach Mercury orbit, with a total deterministic ΔV of about 4.4 km/s, and consists of four Qinetiq T6 gridded ion thrusters, mounted on gimbaled Thruster Pointing Mechanisms (TPMs). Each engine can provide a theoretical thrust in the range 75-145 mN, although currently limited to 125 mN due to lifetime qualification constraints, and up to two thrusters can be operated at the same time when the power produced by the solar panels is sufficient, which is always the case below Earth distance. The TPMs are steered in closed loop to ensure that the thrust vector passes through the centre of mass of the satellite and to prevent accumulation of angular momentum on the satellite reaction wheels.

Bepi Colombo SEP manoeuvres can last up to more than two months and in some cases are critically placed just prior to planetary flybys or Mercury arrival, for this reason a reliable redundant implementation and fast reaction to anomalies are required.

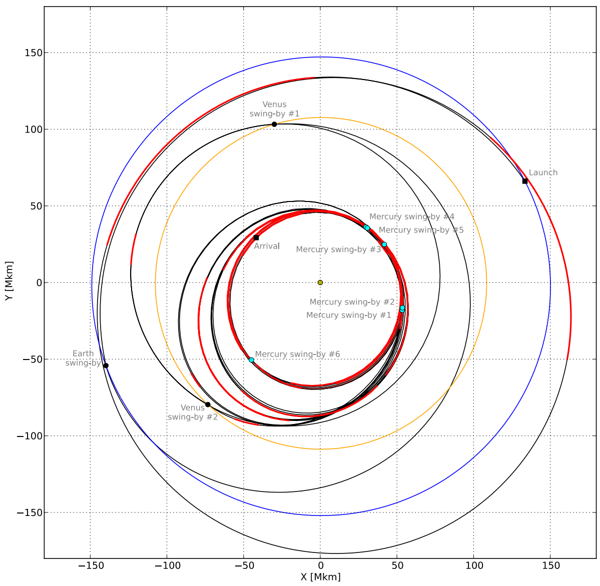

Fig. 1. Bepi Colombo heliocentric cruise trajectory projected onto the ecliptic plane.

II. THE ORBIT DETERMINATION CHALLENGE

The Orbit Determination (OD) of Bepi Colombo during extended SEP arcs that can last up to two months presents several challenges. The dynamic modelling of the system is complicated by the irregular evolution of the thrust acceleration, characterised by transients and very frequent short interruptions (beam-outs), which are not easily predictable a-priori during the command sequence generation, and by the possible unplanned thrust outages caused by SEP system anomalies to which the spacecraft reacts autonomously. The unmodelled portion of the propulsion force disturbance causes a degradation of the radiometric observation residuals used in the OD filter, with a particularly detrimental effect on Doppler.

The satellite attitude is very constrained during manoeuvres, which in some cases results in the unavailability of antenna coverage and consequently of ground contact. Finally, a higher level of thruster degradation than anticipated has been observed, which posed further constraints on the number of planned thrust interruptions.

An example of the Doppler residuals evolution during thrust is shown in Fig. 2. Although the Doppler noise level is not much higher, at least in this specific pass, than what observed during quiet cruise, the frequent thrust interruptions caused by the beam-outs are visible in terms of discontinuities in the data. Such Doppler data cannot be used with normal weighting in the OD due to the extent and non-Gaussian nature of the disturbance.

During the first two extended SEP manoeuvres executed in 2019 a more traditional OD approach was employed, where weekly thrust interruptions were planned to allow clean range and Doppler acquisition and to uplink a guidance update which would compensate the errors accumulated up until the previous OD tracking data cut-off, with a total 1-week latency. A SEP acceleration model based on data packets available in telemetry was already in use, as it had already proved beneficial in reducing the errors deriving from transients and variations on the thrust level, mainly caused by the frequent beam outs. The legacy OD approach and the results obtained during the early SEP manoeuvres are described in full detail in [4].

Already after the second extended SEP manoeuvre, the recommendation from the thruster manufacturer of greatly reducing the number of re-ignition cycles, in order to prevent creeping in the engine neutraliser, was implemented in form of a constraint to not interrupt the thrust more than once every two weeks on average. This would also benefit the last two manoeuvres before the end of the cruise, where very little margin for the recovery of unplanned SEP outages exists.

In view of these limitations the OD strategy needed to be rethought.

As a first step, a new OD approach inspired by a technique exploited by JAXA Hayabusa mission [5] has been validated in-flight, and consists in performing a quasi-kinematic orbit reconstruction, where ΔDOR measurements from two baselines are combined over a very short time interval with range measurements. Doppler measurements could be theoretically considered but only over few-minutes time scales, to prevent that the signatures induced by the irregular thrust degrade the orbit accuracy in the plane-of-sky. This technique gives a very accurate position estimate and a loose velocity estimate, which would be in general sufficient to command a guidance update and compensate for the bulk of the accumulated dispersion errors, even in the absence of an accurate thrust model.

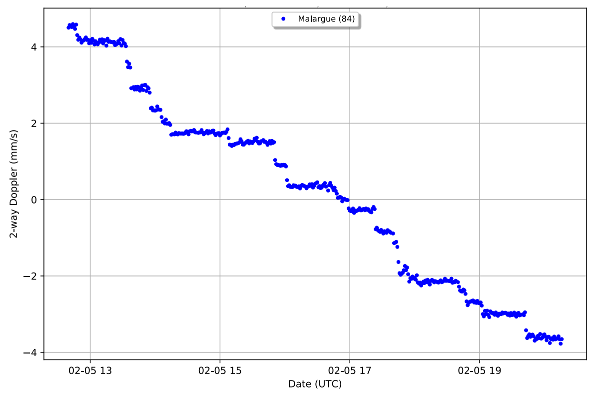

Fig. 2. Pre-fit Doppler residuals during a SEP manoeuvre.

The technique and the results of a dedicated OD experiment are presented in section III.

This method however is not robust to any loss of measurements and requires accurate orbit predictions to resolve the ΔDOR ambiguity; furthermore, it does not allow any calibration of the SEP system because of the very short observation intervals.

The currently adopted OD strategy makes still use of the same ΔDOR and range data combination, but now the OD fit covers long time spans and is augmented by the telemetry-based SEP acceleration model, scaled with an opportune combination of linear scaling factors that allow to estimate reliably the miss-performance of the SEP system. The Doppler data collected during the SEP monitoring passes is not directly used in the fit but is passed-through the reconstructed orbit as a further verification of the achieved OD accuracy. The consistency of the obtained scaling factors values indicates that the estimated SEP performance could be fed forward to reduce prediction errors and possibly reduce the number of planned guidance updates during long burns. A description of the current OD approach and results is the subject of sections V and VI.

All operational and experimental OD activities for Bepi Colombo are run using the ESOC legacy orbit determination software AMFIN [6]. All the results presented in the paper are computed with that software suite, whose main components are a Square Root Information Filter (SRIF) used as a batch estimator, and a Nordsiek integrator for the trajectory propagation part.

III. ΔDOR BASED QUASI-KINEMATIC OD

In [5] a quasi-kinematic navigation approach was proposed and tested in-flight for the JAXA mission Hayabusa, also controlled with ion electric propulsion, where the orbit determination is based on very short sessions of range (15 min) and ΔDOR (45 min) acquired simultaneously from three baselines. This way an accurate "instantaneous" position reconstruction can be obtained with minimum station booking time, while a weak velocity observability is provided by the finite duration of the range sessions and the separation in time of the different ΔDOR scans. This method would practically eliminate the need to interrupt manoeuvres to acquire radiometric data and the need of an accurate thrust model, as the OD is relatively insensitive to errors in the non-gravitational acceleration model, given the short duration of the measurements.

To assess the feasibility of a similar approach for Bepi Colombo, an OD experiment was made in late 2019, when range and ΔDOR data were collected while thrusting during the second SEP manoeuvre, at an Earth distance of 0.36-0.38 AU, as detailed in Table 1.

Table 1. Tracking data acquired during thrust for quasi-kinematic OD experiment.

Date	ΔDOR CEB-NNO	ΔDOR CEB-MLG	Range NNO
04 Oct. 2019	15:37-16:37	18:20-19:20	05:19-12:32
11 Oct. 2019	15:35-16:35	18:00-19:00	04:46-12:25
18 Oct. 2019	15:10-16:10	18:15-19:15	04:37-12:15

A key difference with respect to JAXA approach lies in the fact that simultaneous ΔDOR from ESTRACK deep space stations cannot in general be achieved, except from very specific geometric situations. The solutions presented here thus rely on range and ΔDOR acquisitions that are sufficiently close in time to be meaningful, but not simultaneous.

The experiment methodology consisted in running orbit determinations by using range and ΔDOR only, from a single day. The orbit differences between the OD solutions and the finally reconstructed operational orbit, which is based on an OD with weekly SEP interruptions and acquisition of clean radiometric data during those, were compared with the OD formal uncertainties. Different runs were made with different settings on the dynamic model, in particular SEP acceleration, and on the duration of the range data included in the arc. The longer duration of the data arc when using the full range pass required to scale the acceleration with at least one linear scaling factor, in this case in magnitude only. The configuration of the runs presented here is summarised in Table 2, where the letter "n" in the run identifier indicates the incremental day of the experiment.

Table 2. Test cases definition for quasi-kinematic OD experiment.

Run identifier	ODn_N	ODn_T	ODn_F
Range duration	1h	1h	full
SEP model	Nominal	Telemetry	Nominal
SEP solve-for scale factors	None	None	1(Magnitude)

ΔDOR sessions consist of three measurements, separated by 20 minutes and weighted between 0.07 ns and 0.1 ns. The range measurements are sampled every 20 minutes and have a weight of 1m.

Fig. 3 illustrates the state vector error between each OD run and the finally reconstructed orbit, treated as a reference and obtained with the legacy OD approach enhanced by the ΔDOR measurements as described in [4]. The results are projected in the geocentric Line-of-Sight (LOS) and plane-of-sky North-South (N-S) and East-West (E-W) directions. The 1-σ formal position and velocity uncertainties are also illustrated in the form of error bars. The results demonstrate the viability of the proposed OD scheme, with formal 1-σ accuracies of about 1 km in position and about 0.1 m/s in velocity, and orbit differences with the reference orbit always within the uncertainty, except for the radial direction (as the range data acquired during thrust were not used in the operational orbit reconstruction). The OD results also proved quite insensitive to the adopted SEP acceleration model. The inclusion of a longer range dataset shows the minor benefit of a better formal accuracy in the radial velocity estimation, but requires to scale with at least a single scaling factor (here in magnitude only) the SEP acceleration model.

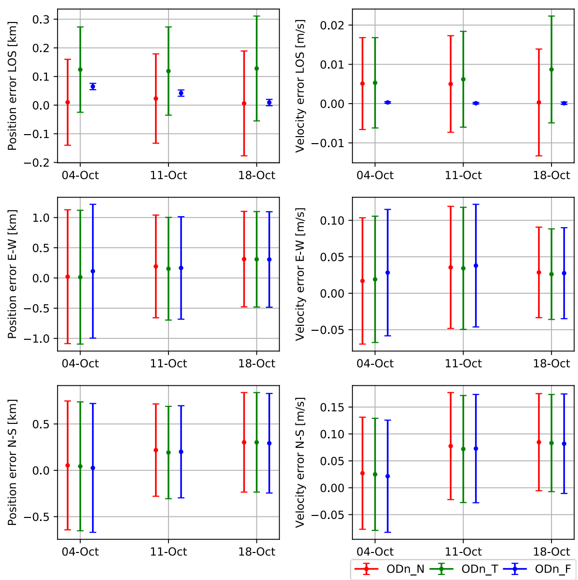

Fig. 3. State vector difference between solutions from OD experiment and reconstructed trajectory. Error bars represent 1-σ formal OD uncertainty.

Later tests demonstrated that the same radial velocity accuracy can be achieved by including a Doppler dataset of very short duration (5-10 minutes). Doppler arcs of longer duration would negatively affect the estimation of the position in the plane of sky.

Although the results of the OD experiment were very positive, concerns over the robustness of this approach discouraged its adoption as the baseline OD method. As the orbit determination relies on a very reduced dataset, a failure in the acquisition of any of the measurements would dramatically compromise the state vector observability and would require the fallback to a very poor accuracy OD based on the Doppler acquired during active SEP. Moreover, the ΔDOR correlation function has a relatively narrow ambiguity window (about 263ns for 3.8 MHz DOR tones); this requires accurate orbit prediction to keep the orbit error within the half ambiguity, otherwise the reduced measurements could be completely wrong. Even in the assumption of weekly ΔDOR acquisition, the orbit prediction uncertainty for certain orbital geometries proved to be very marginal or even outside the ambiguity limit, when considering the manoeuvre implementation errors of an uncalibrated SEP system. The calibration of the SEP acceleration on the other hand is not possible with such short OD data arcs, as demonstrated by the complete insensitivity of the results to the acceleration model.

On top of that, the restrictive spacecraft attitude constraints and the need to thrust even across superior solar conjunctions do not always allow to acquire weekly ΔDOR measurements.

For all these reasons, it was eventually decided to adopt an OD strategy that combines the reliability of a more traditional approach with long data arcs, and the position accuracy given by the ΔDOR measurements and a telemetry-based SEP dynamic model.

IV. SEP Dynamic Model

Already during pre-launch activities, it was anticipated that a telemetry-based SEP acceleration model would be beneficial to improve the OD accuracy, due to its capability of capturing the transients during engine restart, the magnitude variations mainly caused by the beam-outs and the direction errors coming from thrust vector migration, these last predicted to be large for this type of engine.

The SEP thrust magnitude per thruster can be recovered from telemetry through a model that makes use of the measured grid current and voltage. An example is shown in Fig. 4. On the 2nd of February, an autonomous SEP reconfiguration is visible as a short burst for SEP Thruster (SEPT) combination 1+3, followed by a restart on SEPT1+4. The large number and irregular distribution of the beam-out events is also evident as a reduction or complete interruption of thrust.

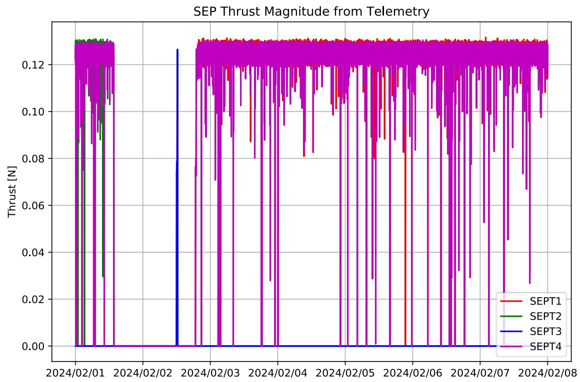

Fig. 4. Example of SEP Thrust magnitude from telemetry.

The thrust direction is obtained by reading the measured Thruster Pointer Mechanism (TPM) angles and assuming, in the absence of better information, that the ion beam is aligned with the thruster assembly axis. Based on pre-launch information, this assumption could theoretically lead to errors up to more than two degrees due to beam migration, however in-flight estimates indicate smaller deviations.

The magnitude and direction information are finally combined in a single acceleration file provided, with a time resolution of 90-120 seconds, in a spacecraft body-fixed reference frame. The file can be then further compressed by the OD team to lower resolution according to the accuracy needs. While at the beginning of the cruise a high resolution of 300 seconds was used, it was observed that the noisy acceleration profile and especially the beam-outs would create numerical problems to the orbit integrator when interpolating the file. Later analyses showed that averaging the thrust over longer periods would not cause a significant loss of accuracy in the OD and would benefit both run-time and numerical stability. The data is currently compressed to 6-hours long constant acceleration blocks, except during the initial transient where shortened intervals are used. An example of the final format of the acceleration file is shown in Fig. 5.

The acceleration from the model can be calibrated in the OD by applying linear scaling factors. A dedicated ΔV calibration reference frame is defined for this purpose, with the Z axis along the modelled thrust direction, the X axis taken as the projection of S/C body frame X axis onto the plane perpendicular to Z, and the Y axis to complete the right-handed frame.

Given the SEP acceleration from the telemetry model, the calibrated acceleration in the ΔV calibration frame can be obtained as:

$$\boldsymbol{a}_{cal} = \begin{bmatrix} 0 \\ 0 \\ \|\boldsymbol{a}_{model}\| \end{bmatrix} + \begin{bmatrix} \lambda_x \\ \lambda_x \\ \lambda_z \end{bmatrix} \cdot \|\boldsymbol{a}_{model}\| \qquad (1)$$

Where λ_x, λ_y and λ_z are the linear scaling factors estimated in the OD.

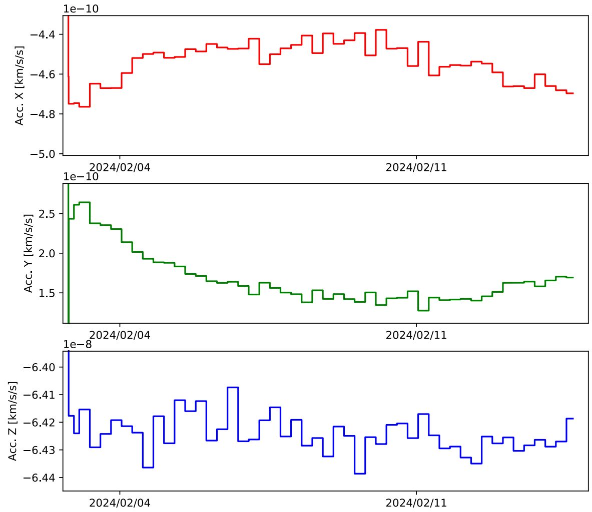

Fig. 5. Processed TM-based SEP acceleration for OD use.

Under the assumption of small angles between real and modelled thrust directions, λ_x and λ_y represent good approximations of the angular deviation in radians of the thrust vector from the telemetry model, in a reference frame which is easy to interpret, being close to the S/C body frame. Similarly, λ_z is a direct measure of the magnitude error.

The scaling factors can be applied over user-selected time batches. The choice of the number and distribution of scaling batches is left to the OD analyst, given the variability of the geometric and viewing conditions, number of radiometric and ΔDOR observations in the data arc, duration of uninterrupted thrust sub-arcs etc. As a general strategy, a scheme with four scaling factors between each session of navigation measurements (Range + ΔDOR) is typically employed: two factors λ_{z1} and λ_{z2} along the thrust direction cover half of the interval duration, whereas two factors λ_x and λ_y cover the interval duration. However, in some cases it has been observed that more reliable results can be obtained by reducing the number of angular batches, being the thrust direction rather stable after an initial transient.

V. THE CURRENT OD APPROACH

The OD strategy currently used during extended SEP arcs relies on long observation data arcs. Two-way Doppler and range are the main tracking data types employed outside of the SEP arcs; two-way range and ΔDOR are mainly used during the thrust arcs. It is requested, at scheduling level, to provide weekly ΔDOR observation sessions from two baselines (1-hour each) and a range pass within the shortest possible time span, to exploit some of the benefit of the quasi-kinematic approach. A very short two-way Doppler dataset (typically 5 minutes) is added to improve line-of-sight velocity estimation without degrading the plane-of-sky information provided by ΔDOR. All the range and the degraded Doppler data from telemetry monitoring passes are discarded in the OD, but their residuals are always evaluated via a pass-through against the reconstructed orbit to detect possible inconsistencies. Sometimes due to spacecraft constraints or station network load this tracking data scheme cannot be fulfilled exactly and some ad-hoc adjustments are needed. Usually a relatively long manoeuvre-free data arc is included in the OD before the start of a SEP burn to constrain the trajectory and avoid filter instability.

The number and timing of SEP thrust interruption for guidance update depends mainly on spacecraft antenna restrictions (that could prevent tracking) and on the capability to recover execution errors and missed thrust in the downstream portion of the trajectory. Ideally thrust should not be interrupted more than once/twice per manoeuvre with the long-term goal of avoiding interruptions entirely.

The OD in preparation of a guidance update has a data cut-off typically 7-10 days earlier than the time of the update, to allow for all the steps of ground processing and uplink. The spacecraft SEP telemetry is normally made available with short latency, thus the TM-based SEP model is considered to be always available at the time of the OD and is employed as baseline, with a fallback to a sequence-based model in case of unavailability.

The high OD accuracy enabled by ΔDOR measurements permits to characterise in detail the performance of the SEP system, both in terms of magnitude and direction. The calibration of the SEP system is obtained by scaling the telemetry-based acceleration in the already defined ΔV calibration frame and directly comparing the scaled acceleration vector with the nominal one obtained from the commanding sequence. If ΔDOR can be acquired throughout an entire manoeuvre, uncertainties as small as 0.05% in magnitude and 15 mdeg in direction can be achieved. Along the cruise it has been observed that different manoeuvres with the same thruster combination give very similar angular error, probably driven by the error in the modelling of the satellite centre of mass and by a relatively stable thruster beam direction error for each thruster. The thrust magnitude is slowly but steadily reducing, probably due to grid degradation. An effort to feed this information in the commanding of future SEP manoeuvre is currently ongoing, with the magnitude error already pre-compensated in recent SEP manoeuvres and the direction error pre-compensation being implemented at the time of writing. This should reduce the number of interruptions planned for guidance update, while also decreasing the size of the chemical correction manoeuvres needed to target the upcoming Mercury swing-bys.

Weekly ODs are run even when a guidance commanding cycle is not required. This is necessary to provide accurate orbit predictions to the ΔDOR correlator, keeping the DOR error within the half ambiguity value of 132ns, which is equivalent to about 600 km at a 1AU geocentric distance and scales linearly with the distance. To improve the prediction error, the orbit propagation can be run on the same day of the DDOR to be acquired: the tracking data cut-off of the OD is one-week in the past (the time of the previous ΔDOR) but a telemetry based model is available until present thanks to the regular monitoring passes and can be used for the orbit prediction, extrapolating the determined scaling factors to the future thanks to the relative stability of the SEP acceleration. This allowed, in the recent months, to keep the ΔDOR pre-fit residuals within just a few nanoseconds, thus safely far from the ambiguity boundaries.

VI. RESULTS FROM SEP #17-18 MANOEUVRES

The OD strategy described above was applied successfully, with some variations, in all extended SEP manoeuvres since late 2022. Here some OD results on the final reconstruction of the recent SEP#17 and SEP#18 manoeuvres are presented. The manoeuvres, of 26- and 12-days duration respectively, were executed between Jan 6th and Feb 14th 2024, for a total ΔV of 213 m/s. They were separated by a day of planned interruption for guidance update on 1st Feb 2024. The tracking dataset used in the OD for the final orbit reconstruction starts on the 27th Oct. 2023 and ends on the 29th Feb. 2024. A superior solar conjunction right before the start of the manoeuvre required to extend the data arc well in the past, due to the de-weighted observations, in order to provide a solid constraint to the reconstructed trajectory. A total of eight ΔDOR sessions from two baselines were acquired in the tracking campaign, one before the start of the manoeuvre, three during SEP#17, two during SEP#18 and two after the end of the manoeuvre. All sessions were spaced by one week, except for a gap at the end of SEP#17 where no S/C antenna coverage was available.
An overview of the main OD filter settings is provided in Table 3.

Table 3. Main OD filter settings for the reconstruction of SEP#17-SEP#18 manoeuvres.

Observation data Arc	2023/11/27 – 2024/02/29
Doppler data	Compressed at 60 s and weighted at 0.2 mm/s (ad-hoc weight settings during solar conjunction)
Range data	Sampled at 1200 s and weighted at 5 m (ad-hoc weight and bias settings during solar conjunction)
ΔDOR data	Three points per baseline/session, weighted at 0.1ns
Solve-for parameters (in brackets, 1σ a-priori uncertainties)	State at epoch (unconstrained), SRP in S/C Y (2%) and Z (2%) directions, range biases per pass (5 m), commanded WOLs (0.1 mm/s, spherical), SEP as below
SEP λ_z scaling factors a-priori sigma	0.03
SEP λ_x, λ_y scaling factors a-priori sigma	0.0175 (equivalent to 1 degree)
Consider parameters (in brackets, 1σ a-priori uncertainties)	SRP in S/C X (0.5%), station coordinates (0.1 m), troposphere (4 cm), ionosphere (25%), group (10 ns) and phase (0.1 ms) delays, pole errors (30 nrad) and UT1 errors (75 ms).

The SEP dynamic model was telemetry-based, and the scheme of the applied scaling factors batches, together with the estimates from the OD, is illustrated in Fig. 6, where the shadowed regions represent the 1-σ uncertainties and the black lines the estimated values; the results from SEP#17 and SEP#18 are shown in separate columns with different time scales due to the different thruster combination used, and consequently different estimated values.
The tracking data residuals from the OD are shown in Fig. 7. The period of the solar conjunction before the manoeuvres is excluded from the view as it would make figures unreadable without added value to the discussion. The adopted scaling scheme permits a very good fit of all data types with a relatively reduced number of parameters. The formal 1-σ state vector covariance, projected in the radial and plane-of-sky directions, is illustrated in Fig. 8 and shows a km-level accuracy throughout the entire duration of the two manoeuvres. The uncertainty on the estimated SEP scaling parameters is also very small and permits a very accurate calibration of the SEP system, with magnitude uncertainties below 0.02% and angular uncertainties up to 90 nrad. These figures could be possibly optimistic since the scaling parameters are applied over long time intervals and as such represent an average and do not capture short-term variations of the SEP error; however, the low tracking data residuals of the monitoring passes and the small orbit variations with different selection of the scaling batches are good indicators of the reliability of the estimates. The accurate SEP scaling estimates permit to characterise in detail the performance of the SEP thrusters. The already known magnitude under-performance is taken into account during trajectory optimisation, which results in a longer duration of the manoeuvre.

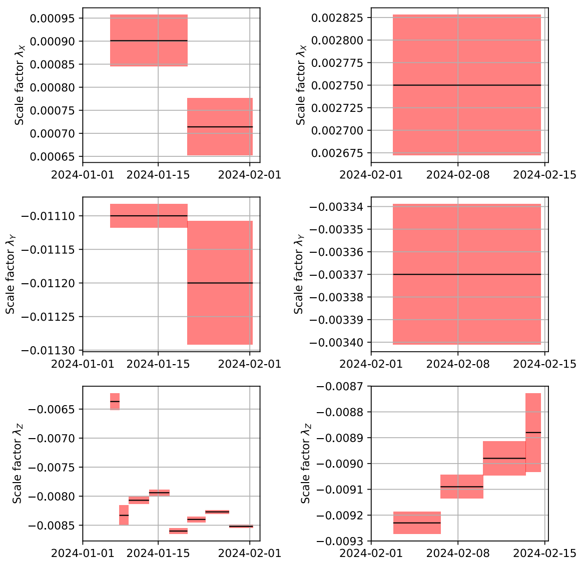

Fig. 6. Distribution and estimated values of SEP scaling parameter batches in SEP#17 (left) and SEP#18 (right) manoeuvres. Shadowed region represents 1-σ uncertainty.

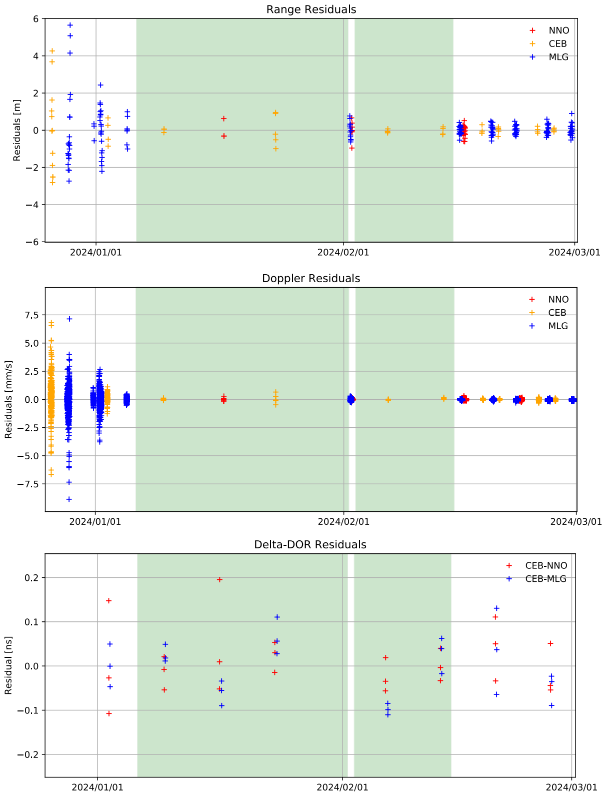

Fig. 7. OD residuals for orbit reconstruction of SEP#17 – SEP#18 manoeuvres. Zoom around manoeuvre periods. Green areas are time periods of active SEP thrust.

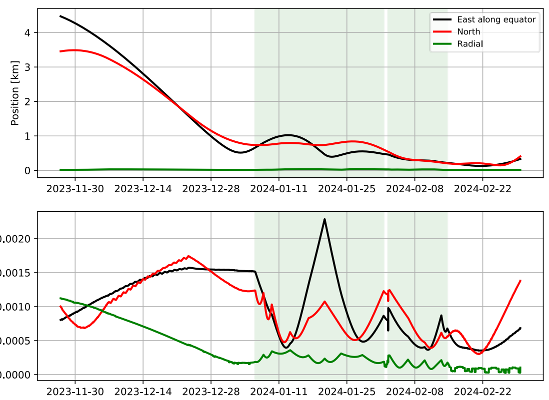

Fig. 8. State vector covariance (1- σ) for the OD calibrating SEP#17-SEP#18. Green areas are time periods of active SEP thrust.

Fig. 9 shows the nominal thrust acceleration of 250 mN already pre-scaled by -1.4% during SEP#18, compared with the thrust computed from telemetry and the one finally obtained in the OD. The pre-compensation resulted very accurate and most of the manoeuvre execution errors derived from an unplanned thrust interruption at the start of the burn (resulting in about 1.5 m/s loss) and from the angular error.

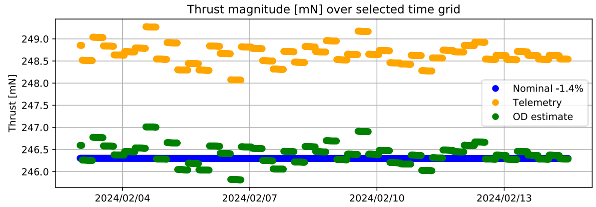

Fig. 9. Calibration of SEP#18 thrust magnitude.

The angular error for SEP#18 is shown in Fig. 10, where the axis of the polar plot is the spacecraft body frame X direction, the angle is the right ascension in the body frame X-Y plane, and the radius is the angular separation of the acceleration vector from the body frame Z direction.

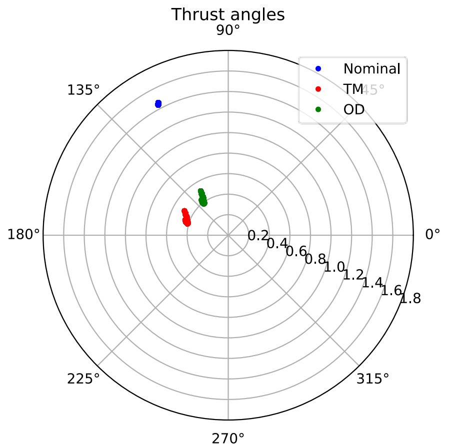

Fig. 10. Calibration of SEP#18 (SEPT1+4) angular error in spacecraft body frame.

It can be observed that the actual thrust acceleration estimated in the OD deviates by approximately one degree from the expected direction.

The direction errors have proven to be systematic and stable when using the same thruster combination. This will permit, starting from the upcoming SEP#19, to pre-compensate them during trajectory optimisation, greatly reducing the largest remaining source of manoeuvre execution errors, besides the non-nominal interruptions caused by SEP system anomalies.

VII. CONCLUSIONS

The challenges faced by the Bepi Colombo mission and navigation teams during long SEP manoeuvres required continuous adaptations and improvements of the orbit determination strategy. Where initially a more traditional approach was used, relying on range and Doppler data acquired during dedicated thrust interruption, the OD system has now evolved in the attempt to reduce the number of SEP restarts. The use of ΔDOR data and a telemetry-based SEP acceleration model is currently allowing not only the accurate estimation of the satellite state vector, but also the detailed characterisation of the SEP thrusters behaviour, which will in turn permit more accurate manoeuvre execution via pre-compensation of all the known systematic errors. This will have immediate benefit during critical mission phases like planetary flybys approach and final Mercury targeting.

VIII. List of acronyms

CEB	Cebreros
DOR	Differential One-way Range
E-W	East-West
LOS	Line-of-Sight
MLG	Malargue
N-S	North-South
NNO	New-Norcia
OD	Orbit Determination
SEP	Solar Electric Propulsion
SEPT	Solar Electric Propulsion Thruster
SRIF	Square Root Information Filter
TM	Telemetry
TPM	Thruster Pointing Mechanism

IX. Acknowledgements

The authors would like to acknowledge all the colleagues in the ESOC Flight Dynamics team for interplanetary missions, as well as the BepiColombo Flight Control Team and the other ESA and Industrial support teams, who contributed to the success of all the cruise milestones accomplished so far.

X. References

[1] J. Benkhoff, G. Murakami, W. Baumjohann, S. Besse, E. Bunce, M. Casale, G. Cremosese, K.-H. Glassmeier, H. Hayakawa, D. Heyner, H. Hiesinger, J. Huovelin, H. Hussmann, V. Iafolla, L. Iess, Y. Kasaba, M. Kobayashi, A. Milillo, I. Mitrofanov, J. Zender, "BepiColombo - Mission Overview and Science Goals", *Space Science Reviews. 217:90*. 10.1007/s11214-021-00861-4, 2021.

[2] R. J. Wilson, and M. Schelklelark, "The BepiColombo Spacecraft, its Mission to Mercury and its Thermal Verification", Presented at 46th Lunar and Planetary Science Conference, The Woodlands, TX, 2015.

[3] F. Budnik, G. Bellei, F. Castellini, and T. Morley, "BepiColombo: Flight Dynamics Operations during Launch and Early Orbit Phase," 27th International Symposium on Space Flight Dynamics 2019 (ISSFD), Melbourne, Australia, 2019.

[4] F. Castellini, G. Bellei, and F. Budnik, "BepiColombo Orbit Determination Activities during Electric Propulsion Arcs", Presented at 2020 AIAA SciTech, Orlando FL, Jan. 2020.

[5] H. Takeuchi, T. Yamaguchi, M. Yoshikawa, T. Ichikawa, N. Ogawa, K. Nishiyama, T. Saiki, Y. Tsuda, S. Taniguchi, N. Fujii, and T. Yagami, "A Quasi-Kinematic Orbit Determination Method for Deep Space Probes", Presented at 26th International Symposium on Space Flight Dynamics, ISSFD 2017-097, June 3-9, 2017, Matsuyama, Japan.

[6] F. Budnik, T. Morley, and R. Mackenzie, "ESOC's System for Interplanetary Orbit Determination", 18th International Symposium on Space Flight Dynamics ISSFD, Munich, Germany, 2005.