Trajectory design for a Ceres lander mission

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***Abstract* – The impact crater Occator constitutes a target of high scientific interest on asteroid 1/Ceres. The present paper outlines the trajectory and orbit design aspects of a potential future landing mission.**

The interplanetary transfer to the asteroid strongly depends on the type of propulsion envisaged. The paper trades low-thrust against conventional options, assuming a launch into direct Earth escape in the late 2030s/early 2040s.

Following Ceres arrival, a pre-landing orbit is targeted. The factors driving the selection of such orbit are the focus of the second part of the paper; these include the orbit stability, the accessibility to the landing site and the presence of backup landing opportunities within a reasonable timeframe.

Data related to fields other than mission analysis are not presented (systems, power, electric propulsion, asteroid science, etc.), neither are the powered landing and the surface phases.

I. INTRODUCTION

Ceres is by far the largest body in the asteroid belt; it is rich in water (it is considered to be the closest ocean world [1]) and shows signs of recent geological activity [2] [3] [4]. For these and other reasons, Ceres remains a priority target for scientific exploration. NASA's mission Dawn [5] arrived at the asteroid in 2015 and orbited it until end of mission in 2018, providing crucial scientific observations, fundamental for the current knowledge of Ceres. Surface sites of particular interest are the optically bright regions within Occator, a 92 km wide impact crater at 19.86 deg N latitude, which can be seen in white in Fig. 1.

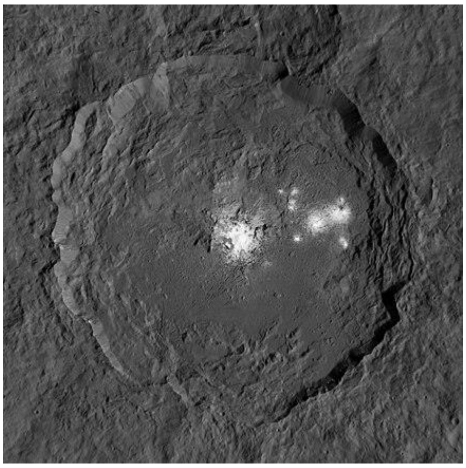

Fig. 1 Occator imaged by Dawn. Credits: NASA

The trajectory design of a possible future mission that would target a landing in the regions of interest on Ceres is presented in the current paper; the focus is the interplanetary transfer trajectory and the pre-landing orbit at the asteroid. Other parts of the trajectory design, including the powered descent to the surface are outside the scope of the current investigations.

II. TRANSFER ASSUMPTIONS AND APPROACH

Assumptions are made to constrain the search space for the interplanetary transfer and to trade the available options. The transfer is assumed to start in all cases after the spacecraft is placed into an Earth escape trajectory by the launcher, assumed to be Ariane 62. If possible, a low escape declination is preferred as this maximizes the launcher performance from the Kourou spaceport. The interplanetary transfer is concluded when Ceres is reached and capture occurs, whether via a standard orbit insertion manoeuvre or as a gravitational capture requiring a sufficiently low arrival velocity. Note that the spacecraft components are not defined in detail, as this is outside the scope of the current work (e.g. if the lander is a separate element of the mission carried by the spacecraft or not).

The interplanetary transfer to Ceres can be achieved with a series of swing-bys at the inner planets coupled with deltaV provided by the spacecraft itself. The way these manoeuvres are applied drives the approach used to search for possible solutions: a mission analyst will use different tools to obtain initial guesses for conventional transfers (deltaV provided by high-thrust chemical propulsion burns) or low-thrust ones (long thrust arcs spanning a large portion of the transfer itself).

For conventional transfers, deep space manoeuvres (DSMs) are approximated as impulsive deltaV performed with chemical propulsion (CP). Conversely, low-thrust trajectories require proper modelling of the available thrust with solar electric propulsion (SEP). To analyse these low-thrust cases and generate initial guesses, the following assumptions for the SEP system were made:

- Available power: for the analyses of the current document power is assumed to be generated via solar arrays and scaling with the inverse of the Sun distance squared; at one astronomical unit (AU), 23 kW are assumed as available.
- Available thrust: thrust is assumed to be directly proportional to the available power; a thrust-to-power value of 30 mN/kW is assumed as available. This assumption is consistent with current gridded electrostatic ion thrusters. A reduction by 10% of this theoretical value is applied throughout to account for SEP outages.
- Specific impulse: a constant value of 4000 s is assumed, consistent with use of a gridded electrostatic ion thruster.

In a later step, not covered in this paper, the mission design will take into account specific constraints of the propulsion system. For instance, there may be upper and lower limits for the input power to the propulsion system or a limit on the number of thrusters operated simultaneously. This must be included in the trajectory design. Also, the system design must be consistent with the mission requirements – for instance, if application of a certain deltaV within a given time frame in the vicinity of Ceres is required, the propulsion system must be capable of that.

In order to efficiently reduce the load on the spacecraft propulsive system, both in case of CP and SEP transfers, the inclusion of gravity assists (GAs) is assumed. Scanning the full space of solutions including GAs coupled with DSMs or low-thrust arcs is a very complex problem and dedicated software tools are used for the task. In general, given the launch timeframe, the possible solutions are generated by automated procedures and are then filtered based on different criteria, such as total transfer time, deltaV or specific constraints, such as those related to launch conditions, or the absence of solar conjunctions during critical mission phases and launch conditions.

The launch conditions assumed for the transfer are not the same for the CP and SEP options: for the CP case the transfer-optimal ones are assumed, driving the performance of the launcher and therefore the feasible initial spacecraft mass; for the SEP case a fixed C_3 of 9 km^2/s^2 is assumed to be reached after launch at -4 deg declination on the Earth equator. In both cases a refinement of the launch conditions is possible but outside the scope of the current analysis: tuning the escape v-infinity or its declination can be done in the frame of launch window analysis when the spacecraft systems are being designed.

Since the thrust-to-mass ratio is a fundamental quantity required for the design of a low-thrust transfer, the initial spacecraft mass is in this case fixed and assumed to be 2.8 tons, feasible with Ariane 62 performance for a direct escape at 3 km/s and low declination.

The most promising solutions for CP and SEP transfers in the current paper can be used for comparison purposes between the two options: the trade-off between the two systems is however outside the scope of the current paper as it can only be done once the architecture is defined for the two and all the necessary assumptions can be made (e.g. payload mass, refined thrusters configuration, etc.).

III. Low-thrust Transfer Options

The crucial assumption that drives the design of low-thrust transfers is the available thrust-to-mass ratio. The space of solutions can look completely different based on this parameter. The solutions described in this paper are valid for thrust-to-mass ratios similar to the one assumed; in fact, if for whatever reason the available power (and therefore thrust authority) is reduced too much and the thrust arcs become saturated, the solution becomes unfeasible. This aspect is to be considered when designing a reference trajectory and is linked to the missed-thrust robustness of the transfer.

In order to generate initial guesses for possible low-thrust transfers from Earth to Ceres, SWING, a tool developed in the Mission Analysis section for solving such problems, is used; it works in steps: first it generates possible combinations of transfer segments (between planetary encounters) based on user configuration ignoring the phasing constraint. Then it closes the time gaps of the unphased initial guesses using a gradient-based optimiser. The same techniques are currently being consolidated in SALTO [6], part of the recently developed Midas software [7].

Under the assumptions listed in section II, many possible transfers are feasible with launch in the late 2030s/early 2040s; such transfers can be grouped based on the planets used for the GAs, as done in Tab. 1, and rough ranges of transfer time and total deltaV can be provided.

Tab. 1 Solution families for SEP transfers

GA bodies	Sub-type	Transfer time [y]	Total deltaV [km/s]
None		5 – 7	9.5 – 11.0
Only Earth	Fast	6 – 9	9.5 – 10.0
	Slow	9 – 11	7.5 – 9.5
Earth and Mars		5.5 – 7.5	6.0 – 8.0

As can be seen in Tab. 1, using only the Earth gravity assists (EGAs) offers limited advantage with respect to direct options with no GAs: to be effective in reducing the deltaV requirements, high penalties on the transfer time have to be accepted. The best solutions are obtained by adding Mars gravity assists (MGAs): in most cases one is sufficient, though for some solutions two are needed.

Among the available transfers that include Mars GAs, one is selected as reference and its trajectory is reproduced and optimised with high fidelity in GODOT [8], the reference tool for astrodynamics calculations at ESOC Flight Dynamics Division. The reference solution taken is not the absolute optimal in terms of deltaV or transfer time, but rather the best compromise considering many factors, among which the most relevant are:

- Segments between swing-bys far from being saturated with thrust arcs (indicatively considered ok if the thrusters are required to be active for less than 80% of the time); this increases the robustness of the solution to changes in the available thrust (or power indirectly) due to variations in the system design and to contingencies occurring during operations (missed-thrust).
- No solar conjunctions in proximity of swing-bys or Ceres arrival; it would not be acceptable for operational safety to have no communication possibility close to a critical event.
- The selected option not representing a particularly favourable phasing case that only rarely occurs; re-occurrence of transfers with similar or lower deltaV and transfer times ensures robustness to unexpected delays of the launch beyond initially considered backups.

In order to create a refined reference trajectory based on the selected initial guess, number and type of thrusters are selected to refine the thrust model; up to two RIT-2X Xenon thrusters are assumed to be operable at the same time, while a single PPS1350 Hall effect thruster is used for the approach to Ceres; the choice of adding a second type of thruster (with different thrust-to-power ratio, power limits and specific impulse) is driven by the fact that operating a single RIT-2X at ~3 AU from the Sun is not possible even at the lowest power mode; usage of the very efficient gridded ion thrusters is however maintained wherever else possible.

In Fig. 2 the full projection of the generated reference trajectory on the ecliptic plane is shown; the evolution of the distance to the Sun and relevant major bodies as function of time is shown Fig. 3. As it can be seen, during the thrust arc preceding Ceres arrival the distance to the Sun is between 2.5 and 3 AU, leading to an available power for the SEP module between 2.6 and 3.7 kW, below the minimum operating point of RIT-2X.

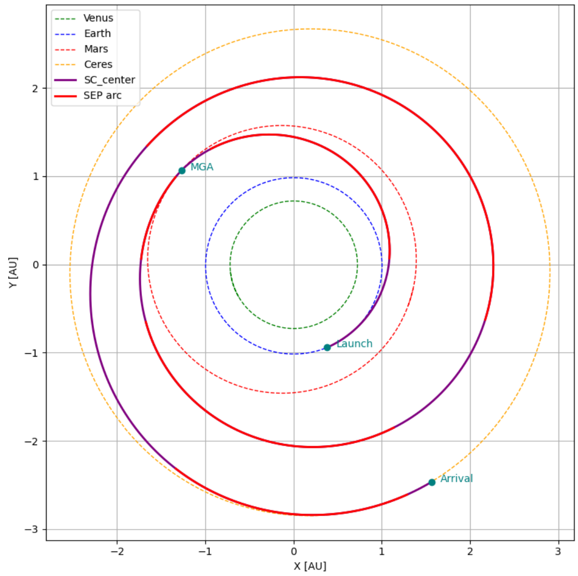

Fig. 2 SEP transfer ecliptic plane projection

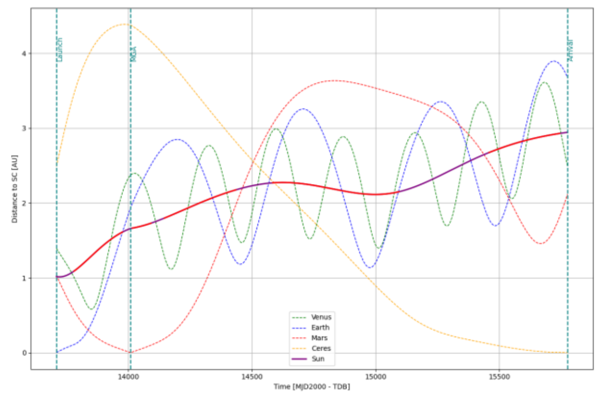

Fig. 3 Distance evolution to major bodies

The contribution of the MGA and the SEP thrust arcs on the transfer can be directly visualised plotting the evolution of the semi-major axis and inclination on the ecliptic as function of time, as done in Fig. 4; the MGA is essentially exploited to increase the orbit semi-major axis, while the substantial inclination change (Ceres lies at ~10 deg inclination on the ecliptic plane) is performed completely by the spacecraft propulsive system.

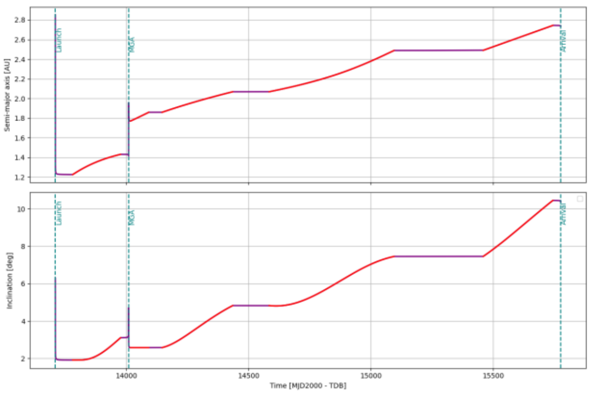

Fig. 4 Semi-major axis and inclination evolution

A summary of the main events during the transfer is given in Tab. 2; it is worth noting that the only mission that visited Ceres so far, NASA's Dawn, also employed SEP, but visited Vesta before Ceres: the transfer as a whole cannot be therefore compared directly with the one described in the current section.

Tab. 2 2037 EMC low-thrust transfer summary

Event	Epoch Start	Epoch End	V-inf [km/s]	Minimum altitude [km]	Duration [days]	DeltaV [m/s]
Launch	2037-07-14		3	-	-	-
SEP#1	2037-09-23	2038-04-07	-	-	195	2110
MGA	2038-05-10		1.918	428	-	-
SEP#2	2038-05-18	2038-07-31	-	-	75	519
SEP#3	2038-09-24	2039-07-09	-	-	289	1497
SEP#4	2039-12-06	2041-04-29	-	-	510	2293
SEP#5	2042-04-28	2043-02-07	-	-	285	1366
Arrival	2043-03-11		0.1	-	-	-
Total	-		-	-	**2066**	**7785**

As already mentioned, backup opportunities in following years are present requiring similar (or lower) deltaVs and comparable transfer times; for example an opportunity with launch 2039 is present with a 6.8 years transfer requiring 7.7 km/s deltaV. If a very similar mission profile is required for the baseline and backup and/or more frequent launch windows are needed, it is conceivable to prepend a 1:1 resonant arc with the Earth and anticipate the launch by one year: this transfer would be a year longer but the deltaV required would be similar if not slightly better than the original case (additional opportunity for thrust arc during the first heliocentric segment). Building on the reference presented, a case with launch in 2036 can be constructed; in that case the baseline would be the longer transfer starting in 2036 and the backup would be the one summarised in Tab. 2, launching in 2037.

The shown transfer is concluded with an approach of Ceres at low relative speed (v-infinity of 100 m/s assumed). It can be envisaged to use SEP to complete the insertion into orbit around Ceres and the spiral down towards the pre-lading orbit described in section V. Preliminary estimates for this phase are a deltaV cost of around 250 m/s and a duration of few months, if no specific intermediate orbits are needed for science; usage of the smaller Hall effect thruster PPS1350 is possible for this phase.

IV. Conventional Transfer Options

To explore all possible options, various sequences of swing-bys at Venus, Earth and Mars and conventional chemical propulsion have been studied using the global transfer analysis tool SOURCE [9] [10]. The most promising strategy identified, assuming a launch date in the late 2030s/early 2040s involves two Mars swing-bys.

Tab. 3 summarizes the best Earth-Mars-Mars-Ceres (EMMC) case in the studied launch time from 2036 to 2041. Even considering the arrival velocity at Ceres, it has not only the lowest deltaV requirement, but also the shortest transfer duration and a moderate Earth escape velocity at 34.9 deg declination on the Earth equator; the total transfer time is 5 years.

Tab. 3 2041 EMMC transfer summary

Event	Epoch	V-inf [km/s]	Minimum altitude [km]	DeltaV [m/s]
Launch	2041/11/19	3.49	-	-
MGA#1	2042/12/16	4.551	314	-
DSM#1	2045/04/19	-	-	34
MGA#2	2045/09/12	4.472	300	-
DSM#2	2045/10/05	-	-	180
Arrival	2046/12/03	3.181	-	-
Total	-	-	-	**214**

A mission design requires a viable baseline and backup. This could be achieved by placing the baseline launch date in November 2040, adding one more year and an EGA to the baseline transfer; the first revolution around the Sun would be in a 1:1 resonance with the Earth. The backup scenario would then be as outlined in Tab. 3.

Though it is possible that some improvement over the mission characteristics listed above may be found for launch dates outside the 2036-41, the sample case summarised in Tab. 3 can be considered as reference for any comparison with the SEP cases described in section III.

For the conventional transfer option, a Ceres orbit insertion burn is envisaged to capture around the asteroid; as a preliminary allocation this can be considered roughly equivalent to the arrival v-infinity of ~3 km/s, representing more than 90% of the entire deltaV of the transfer. Manoeuvres for all post-capture operations near Ceres are not estimated in the current paper.

V. The Pre-Landing Orbit

Prior to deploying the landing craft, the spacecraft is assumed to be placed into the lowest orbit around Ceres that enables landing in one of the candidates landing sites in Occator crater at latitudes slightly below 20 deg N.

The orbital dynamics are governed by the strong oblateness (larger than that of the Earth by a factor of 25) and third body perturbations. Low circular orbits are not stable, even at inclinations of less than 20 deg, which is the minimum required to remain consistent with landing at one of the identified sites.

Use of the critical inclination of 63.4° to maintain a constant argument of periapsis is also not an option, as the third body perturbations move the inclination, away from the critical value, leading to a fast line of apsides drift. In addition, a highly inclined orbit around Ceres will be unstable due to the eccentricity variations.

However, a class of orbit with low inclination and moderate eccentricity is found that feature a near-frozen eccentricity. A reference orbit is chosen with an inclination near 18.5 deg and mean peri- and apoapsis altitudes of 25 and 400 km, respectively.

This orbit features long-term stability; no station keeping is needed and the spacecraft would not crash in case of a temporary lapse of communications. Node and line-of apsides are inevitably subject to rapid drift, going full circle in just a few days.

The orbital period and inclination must be carefully tuned such that the periapsis passes above the targeted landing site in regular (e.g., monthly) intervals, with the lighting conditions required for the landing guidance navigation and control (GNC). This ensures that a sequence of possible landing opportunities is available. In case one opportunity is missed, the problem can be rectified and the next landing attempt can be planned.

VI. Conclusions

The mission analysis for a possible future mission to land on Ceres was conducted; the focus was the interplanetary transfer and the selection of the pre-landing orbit at Ceres.

Interplanetary transfer options for a mission to Ceres have been analysed; both conventional and low-thrust propulsion representative options have been found. For an architecture based on solar electric propulsion transfer opportunities requiring 5.5 to 8 years and a deltaV below 8 km/s have been presented for the 2036-2039 timeframe. All included Mars gravity assists and some included an additional Earth gravity assist. The transfers are feasible also with reduced power, albeit with lesser margins. As a term of comparison, an option for a conventional transfer with chemical propulsion was also found in the late 2030s/early 2040s, requiring above 3 km/s deltaV, concentrated at Ceres arrival, and a transfer time of 5 to 6 years; in this case two Mars gravity assists are required as minimum.

The possible operational orbit before landing were explored and indications for refining the design of such orbit were given. Stability of low Ceres orbits were assessed, finding good candidate solutions with pericentre altitude at 25 km and apocenter altitude up to 400 km; an inclination of 18.5 deg was considered, the minimum that guarantees access to a landing in the regions of interest at Occator without large deltaV penalties to initiate the descent. The fine tuning of the orbit parameters will be required to obtain some degree of synchronicity between the line of apsides drift and the Ceres body rotation to ensure multiple landing opportunities in a reasonable time frame.

VII. References

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