Low-thrust trajectories for human missions to Ceres

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ABSTRACT

A low-thrust trajectory design study is performed for a mission to send humans to Ceres and back. The flight times are constrained to 270 days for each leg, and a grid search is performed over propulsion system power, ranging from 6 to 14 MW, and departure \( V_1 \), ranging from 0 to 3 km/s. A propulsion system specific mass of 5 kg/kW is assumed. Each mission delivers a 75 Mg payload to Ceres, not including propulsion system mass. An elliptical spiral method for transferring from low Earth orbit to an interplanetary trajectory is described and used for the mission design. A mission with a power of 11.7 MW and departure \( V_1 \) of 3 km/s is found to offer a minimum initial mass in low Earth orbit of 289 Mg. A preliminary supply mission delivering 80 Mg of supplies to Ceres is also designed with an initial mass in low Earth orbit of 127 Mg. Based on these results, it appears that a human mission to Ceres is not significantly more difficult than current plans to send humans to Mars.

1. Introduction

Among potential destinations for humans to explore in the Solar System, Ceres stands out as one well-suited to human exploration. However, there has been little research that has addressed the problem of sending a human crew to Ceres. Benton has proposed a nuclear thermal rocket (NTR) vehicle design that could reach Ceres [1], but to our knowledge there has been little else on the matter. Other destinations have been subject to more study. Chief among them is Mars, which has long been considered the natural next step for exploration after the Moon [2–8]. We have also seen proposals to send astronauts to a near-Earth asteroid (NEA) [9] and to a Lagrange point in the Earth–Moon system [10]. While authors have looked at electric propulsion missions to Ceres at least as far back as 1971 [11], they have focused on robotic probes such as Dawn, which will reach Ceres in 2015 [12].

We aim to address this gap by presenting a low-thrust mission architecture that assesses the feasibility of a human mission to Ceres. Ceres possesses resources to aid in human exploration. Earth-based observations have demonstrated a high likelihood that significant quantities of water ice are present in the crust of Ceres [13,14]. When Dawn reaches Ceres in 2015, we will greatly expand our knowledge of the dwarf planet.

Reaching Ceres is a challenge because its very low gravity offers little assistance to a vehicle attempting to capture into orbit. Ceres’ orbit also has an inclination of about 10.6°. At the same time, it lacks any appreciable atmosphere, so landing on Ceres would be similar to landing on the Moon or a large asteroid. On Mars, spacecraft can use the atmosphere to decelerate before landing, saving propellant. However, the atmosphere introduces significant uncertainty during landing, resulting in a target radius on the order of 10 km. On Ceres, thrusters must provide all deceleration, but in principle a more accurate landing should be possible.

We present a high-level mission concept to send human astronauts to Ceres and back. We focus on the low-thrust trajectory design but do not present a detailed
design of a transfer vehicle. However, estimates of the masses of the vehicles, the propellant costs, and the total initial mass in low Earth orbit (IMLEO) are provided. In addition, we provide a method to scale the mass results up or down to accommodate a payload mass different from the one assumed here. Our primary goal is to determine whether a human mission to Ceres is feasible given current technology and to identify which technological areas require further development.

2. Design methodology

2.1. Mission architecture overview

This mission presented here is built around the assumption of a two-vehicle, low-thrust propulsion concept. The first vehicle is the supply transfer vehicle (STV) and its mission is to deliver all supplies necessary to sustain the crew while on Ceres as well as any propellant or equipment required to return to Earth. We assume that its mission must be successfully completed before the astronauts depart.

The second vehicle is the crew transfer vehicle (CTV). We begin our mission analysis assuming that the CTV is already assembled in low-Earth orbit. It departs the Earth under the power of its electric propulsion using an elliptical spiral escape, performs an impulsive burn to achieve some departure $V_{\infty}$, and uses electric propulsion to transfer to Ceres and back again to Earth. We will provide greater detail on each of these mission phases later in the paper.

2.2. Constraints

The need to protect the crew from a lengthy period of deep-space radiation exposure is the main factor driving the trajectory design for the crew mission. While there is great uncertainty in the effects of deep-space radiation on the human body, most authors indicate that such exposure would likely lead to fatal cases of cancer as well as other non-cancerous diseases [15]. We have constrained the Ceres-bound and Earth-bound legs to be no more than 270 days each, and the total time spent by the crew away from Earth on the Ceres mission to be no more than 2 years. For comparison, the NASA Mars Design Reference Architecture 5.0 (DRA5) specifies a maximum 180-day time of flight each way. While DRA5 has a total of six months less time in deep-space, the crew remains on the Martian surface for over a year, so the total time away from Earth is longer than the 2 year constraint we use here. A preliminary analysis indicated that a 270-day constraint provides a good balance between minimizing crew exposure to radiation while still requiring a reasonable IMLEO cost. We will return to the question of how this constraint affects IMLEO later in the paper.

Cucinotta and Durante [16] estimate that, given flight times similar to what we use here, the increased risk of developing a fatal case of cancer caused by exposure to deep-space radiation is about 4.0% for men and 4.9% for women, although these numbers are highly uncertain. While limiting flight times is one possible way to mitigate the risks faced by the crew, we acknowledge that the risk and uncertainty associated with deep-space radiation remains a major dilemma for human exploration of the Solar System.

Upon arrival at Ceres, $V_{\infty}$ is constrained to be zero. This constraint is required because aerobraking is not possible at the atmosphere-free Ceres, and its gravity is so low that an impulsive capture maneuver is prohibitively expensive.

2.3. Technology assumptions

To make this mission possible while meeting the constraints, a nuclear electric propulsion (NEP) system is used throughout all stages of the mission. The low gravity and non-existent atmosphere on Ceres means that an impulsively propelled mission would require a significant amount of propellant to capture into orbit around Ceres and land. Unlike Mars, no aerocapture or aerobraking is possible. For these reasons, an electric propulsion system is selected because it allows the spacecraft to reach Ceres on a zero-$V_{\infty}$ approach and spiral down to a low parking orbit. A nuclear power system is chosen over a solar-electric one because its specific mass is lower and its power output remains constant. For this study, we assume a propulsion system specific mass of $\alpha = 5\,\text{kg/kW}$. This would likely lead to fatal cases of cancer as well as other non-cancerous diseases [15].

<table>
<thead>
<tr>
<th>Nomenclature</th>
<th>Description</th>
</tr>
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<tbody>
<tr>
<td>$V_{\infty}$</td>
<td>hyperbolic excess velocity, km/s</td>
</tr>
<tr>
<td>$e$</td>
<td>eccentricity</td>
</tr>
<tr>
<td>$\Omega$</td>
<td>longitude of the ascending node, rad</td>
</tr>
<tr>
<td>$\omega$</td>
<td>argument of periapsis, rad</td>
</tr>
<tr>
<td>$\theta^*$</td>
<td>true anomaly, rad</td>
</tr>
<tr>
<td>$p$</td>
<td>semi-latus rectum, km</td>
</tr>
<tr>
<td>$L$</td>
<td>true longitude $\Omega + \omega + \theta^*$, rad</td>
</tr>
<tr>
<td>$i$</td>
<td>inclination, rad</td>
</tr>
<tr>
<td>$f$</td>
<td>modified equinoctial element $e \cos(\omega + \Omega)$</td>
</tr>
<tr>
<td>$g$</td>
<td>modified equinoctial element $e \sin(\omega + \Omega)$</td>
</tr>
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<td>$h$</td>
<td>modified equinoctial element $\tan(i/2) \cos \Omega$</td>
</tr>
<tr>
<td>$k$</td>
<td>modified equinoctial element $\tan(i/2) \sin \Omega$</td>
</tr>
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<td>$a_i$</td>
<td>$i$th acceleration component, km/s$^2$</td>
</tr>
<tr>
<td>$a$</td>
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</tr>
<tr>
<td>$\beta$</td>
<td>steering angle, rad</td>
</tr>
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<td>$m_{pl}$</td>
<td>usable payload mass, Mg</td>
</tr>
<tr>
<td>$m_f$</td>
<td>total final mass at Ceres, Mg</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>propulsion system specific mass, kg/kW</td>
</tr>
<tr>
<td>$P$</td>
<td>spacecraft power, MW</td>
</tr>
<tr>
<td>$\mu_p$</td>
<td>propellant tank mass factor</td>
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<td>propellant mass, Mg</td>
</tr>
<tr>
<td>$T$</td>
<td>thrust, N</td>
</tr>
<tr>
<td>$m_{leo}$</td>
<td>initial mass in low Earth orbit, Mg</td>
</tr>
<tr>
<td>$m_1$</td>
<td>total mass after spiral, Mg</td>
</tr>
<tr>
<td>$m_2$</td>
<td>total mass after chemical escape, Mg</td>
</tr>
</tbody>
</table>
specific mass is in line with proposed methods of space-based nuclear power generation [17,18].

In addition to a nuclear power source, the mission requires an electric engine capable of meeting the thrust and specific impulse requirements. The VASIMR engine [19] is one such propulsion concept under development, and the magnetoplasmadynamic (MPD) thruster is another technology that has seen some study [20]. We assume a propulsive efficiency of 70%. As we will see, the architecture presented here—to minimize IMLEO—requires a thruster capable of processing 11.7 MW of input power at a constant specific impulse of 6800 s.

2.4. Outbound trajectory design

To minimize IMLEO for the mission, we perform a straightforward grid search of the trajectory design space with departure $V_{\infty}$ and spacecraft power as our design variables. In creating this grid, we search over a range of 6–14 MW for power, in increments of 2 MW, and 0–3 km/s for $V_{\infty}$, in increments of 1 km/s. This range yields a total of 20 possible design points. In the grid study, we proceed by first designing the interplanetary trajectory, next computing the $\Delta V$ for the Earth-orbit departure, and then designing an elliptical spiral that uses electric propulsion to escape low-Earth orbit. These steps are described in detail in Sections 2.4.1–2.4.3, as follows.

2.4.1. Interplanetary trajectory

At each design point, we use the software package MALTO (Mission Analysis Low-thrust Trajectory Optimization) to design a low-thrust trajectory. MALTO is developed and maintained by the Jet Propulsion Laboratory, and uses a direct method to produce optimal low-thrust trajectories [21]. For this study, we direct MALTO to produce a trajectory from Earth to Ceres that delivers 125 Mg (i.e. metric tons) of mass in 270 days, while minimizing the total departure mass. This 125 Mg includes the propulsion system mass. During the optimization process, the launch date is free, while the specific impulse is chosen from the range of 2000–8000 s. At the end of this step, we have the mass of the propellant required for the interplanetary leg and can compute the associated tank mass. We assumed a structural mass factor of 15%.

2.4.2. Earth orbit departure

Once we know the mass at the start of the interplanetary leg, we need to compute the mass of the chemical stage which places the transfer vehicle on an escape trajectory with the appropriate departure $V_{\infty}$. The maneuver occurs at the perigee of a highly elliptical orbit, discussed in the next section. First, the escape $\Delta V$ is computed by taking the difference between the hyperbolic perigee velocity and the elliptical perigee velocity:

$$\Delta V = \sqrt{\frac{V_{\infty}^2}{2} + \frac{2\mu}{r_p}} \sqrt{\frac{(1+e)\mu}{r_p}}$$  \hspace{1cm} (1)

To achieve a 3 km/s $V_{\infty}$ from an orbit with 0.95 eccentricity and 350 km perigee altitude, a $\Delta V$ of 543 m/s is required.

Then, the propellant mass is computed using the rocket equation, assuming an LH2/LOX propulsion stage with $I_{sp} = 450$ s:

$$m_0 = m_f \exp(\Delta V / (I_{sp}g_0))$$ \hspace{1cm} (2)

Again, we compute the structural mass assuming a factor of 15%.

2.4.3. Elliptical spiral

While it is possible to simply spiral away from the Earth with the electric propulsion engine by applying tangential thrust, this method would make it difficult for the crew to rendezvous with the transfer vehicle at the end of the spiral stage. The tangential spiral also makes it difficult to impart a departure $V_{\infty}$ on the vehicle because the transfer from elliptical to hyperbolic orbit occurs at a point far away from Earth. A direct injection from low Earth orbit (LEO) to interplanetary transfer is also possible with a chemical propulsion stage, but this would be an inefficient use of propellant, especially for a spacecraft already equipped with a highly efficient electric propulsion system.

Instead, we employ an elliptical escape spiral similar to that proposed by Sweetser et al. [22]. Following their method, the spacecraft begins in LEO, and uses its electric propulsion system to spiral away from the Earth (without crew on board) with a steering law that keeps perigee constant while increasing apogee and eccentricity. An example of such a trajectory is shown in Fig. 1.

Because MALTO is not capable of computing the propellant required for an elliptical escape spiral, we use MATLAB to integrate the trajectory with the steering law from Sweetser et al., where we derive the steering law in terms of the modified equinoctial elements provided by Walker et al. [23]. The modified equinoctial elements can be obtained by a simple transformation of the classical orbital elements

$$p = a(1-e^2)$$  \hspace{1cm} (3)

![Fig. 1](image-url)
The steering angle, $\beta$, is defined here as the angle from the local horizon to the thrust vector, and is given by

$$\tan \beta = \frac{a_r}{a_o} = \frac{e^2 + 2e - w^2 + 1}{w(f \sin L - g \cos L)}$$

In the form originally given by Sweetser et al. [22], the steering law produced a singularity when the spacecraft reached periapse and apoapse. This singularity required the user to modify the steering law slightly when implementing it numerically. Using the form presented in Eq. (22), numerical difficulties did not occur when propagating a trajectory using the elliptical spiral steering law.

While it is not obvious upon examining the control law as presented here, intuitively we can surmise that when the spacecraft is at periage, thrust should be in the tangential direction, and when the spacecraft is near apoage, thrust cannot be in the tangential direction, or else it will raise periage. So in general, the thrust direction for true anomalies near 180° will tend towards the radial direction. However, radial thrust does not increase the energy of the orbit and is an inefficient use of propellant if we want the spacecraft to escape. Therefore, we can save propellant by setting a maximum true anomaly, $\theta_{\text{max}}$, beyond which the spacecraft will coast. It will resume thrusting after it passes $2\pi - \theta_{\text{max}}$. While this modification to the control law does save propellant, it comes at the expense of increased time in the elliptical spiral stage. For this mission we have found $\theta_{\text{max}} = 60°$ to strike a good balance between propellant savings and increased time of flight.

We note that there may be a better and more efficient way to increase apoage and eccentricity than the Sweetser method, however the goal of this paper is to find the optimal solution, but a practical one that demonstrates the feasibility of the human mission to Ceres we are proposing here. There may exist a steering law that achieves the same target orbit as the elliptical spiral used here, but at a reduced propellant cost. Additionally, the Moon may be used as a gravity-assist body during the escape phase. These escape strategies should only reduce propellant usage, and hence IMLEO, for the mission. Because this study is intended as a high-level feasibility analysis, we will proceed with the non-optimal escape strategy which, as we will see, is adequate for the mission given our assumptions.

The CTV travels on a spiral trajectory for about 2 years to reach a highly eccentric orbit with $e = 0.95$ and a periage altitude of 350 km. (Subsequently, the crew is launched in a small capsule to rendezvous with and board the CTV.)

At each point in the grid study, we compute the propellant mass (and tank mass) required to bring the
transfer vehicle and chemical departure stage from LEO to
the highly elliptical orbit. This computation is done by
backwards propagation from an elliptical orbit \( e=0.95 \)
with the final mass required for the impulsive escape
maneuver to a circular orbit of 350 km altitude. The initial
mass (computed at the end of the backward propagation)
is a reasonable estimate of the total IMLEO required to
complete the mission.

2.5. Computing payload mass

While each trajectory delivers 125 Mg to Ceres, the
payload mass varies because we must deduct the inert
mass of the propulsion system from the final mass:

\[ m_{pl} = m_f - \alpha P - \mu p m_p \]  

(23)

The propulsion system inert mass includes the nuclear
reactor, the thrusters, and the propellant tanks. So while
high-power missions will tend to use less propellant than
low-power missions, they will require a larger power
system mass. These competing effects generally result in
an optimum propulsion power level that balances propel-

lant mass with inert mass.

2.6. Scaling the results

MALTO is able to compute a trajectory that minimizes
initial mass given a fixed final mass, but it is not able to do
the same given a fixed payload mass. Because of this, the
set of trajectories produced by the grid search are unequal
in the sense that they do not deliver the same usable
payload to Ceres. To account for this fact, we adopt the
scaling method presented by Landau et al. [24]. This
method allows the mission planner to take a trajectory
that delivers a particular payload mass, \( m_{pl} \), and scale it up
in such a way that the scaled trajectory follows the same
course and has the same time of flight, but delivers a new
payload mass, \( m_{pl}^* \), to the destination. The first step is to
compute the scaling factor:

\[ \mu_{sc} = m_{pl}^*/m_{pl} \]  

(24)

Then, this factor can be used to scale up other key mission
parameters, such as spacecraft power and thrust:

\[ P^* = \mu_{sc} P, \quad T^* = \mu_{sc} T \]  

(25)

We are also able to scale up the propellant masses and
total IMLEO for the trajectory:

\[ m_{pl}^* = \mu_{sc} m_p, \quad m_{LEO}^* = \mu_{sc} m_{LEO} \]  

(26)

For each design point in the grid search, we have scaled
the final payload mass to 75 Mg and adjusted the space-
craft power for each design accordingly.

2.7. Return trajectory

To design the return trajectory, we again use MALTO. In
a manner similar to designing the outbound trajectory, we
fix the mass at Earth arrival to be 125 Mg, constrain the
flight time to be no more than 270 days, and direct MALTO
to minimize the propellant mass. We assume that the
transfer vehicle restocks the propellant it needs for
the return leg at Ceres. The return propellant may either
be delivered directly in the supply mission, or it may be
produced by an in situ propellant facility delivered on the
supply mission. Here we assume that it is delivered on the
supply mission. Upon return to Earth, the arrival \( V_\infty \)
is constrained to less than 4.5 km/s, which results in an
atmospheric entry velocity of around 12 km/s. For refer-
ence, the Apollo entry velocity was about 11 km/s [25]. The
ability to use the Earth’s atmosphere to capture signifi-
cantly reduces the propellant required on the return trip
compared to the outbound trip.

2.8. Supply mission

Before any mission carrying astronauts departs for
Ceres, a mission to bring supplies and resources to Ceres
should have been successfully completed. We perform
another grid search to estimate the IMLEO of such a cargo
mission in a manner much the same as that of the human
mission. The supply mission analysis is different from the
human mission in that (1) the time of flight is constrained
to 2 years and (2) the grid search is one-dimensional
over power.

Departure \( V_\infty \) is constrained to be zero. This constraint
allows the supply vehicle to depart Earth on a circular
spiral, which is possible because the absence of crew
removes the need for the elliptical spiral. The power level
is varied from 1 to 3 MW in increments of 0.5 MW in
search of an optimal solution. Our search range is lower
than that of the human mission because the longer time of
flight allows for lower thrust and lower power.

3. Results

3.1. Human mission

Table 1 contains the complete listing of results from the
outgoing grid search. Of the original 20 design points, 4

<table>
<thead>
<tr>
<th>P (MW)</th>
<th>( V_\infty ) \text{ (km/s)}</th>
<th>( m_{le0} )</th>
<th>( m_1 )</th>
<th>( m_2 )</th>
<th>( m_{pl} )</th>
<th>( m_{pl}/m_{le0} )</th>
<th>Return</th>
<th>( m_{pl}^* )</th>
</tr>
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<tbody>
<tr>
<td>6</td>
<td>3</td>
<td>454</td>
<td>394</td>
<td>340</td>
<td>63</td>
<td>0.14</td>
<td>120.0</td>
<td>542</td>
</tr>
<tr>
<td>8</td>
<td>1</td>
<td>391</td>
<td>347</td>
<td>330</td>
<td>54</td>
<td>0.14</td>
<td>87.4</td>
<td>541</td>
</tr>
<tr>
<td>8</td>
<td>2</td>
<td>320</td>
<td>290</td>
<td>266</td>
<td>64</td>
<td>0.20</td>
<td>72.7</td>
<td>376</td>
</tr>
<tr>
<td>8</td>
<td>3</td>
<td>295</td>
<td>271</td>
<td>234</td>
<td>69</td>
<td>0.23</td>
<td>64.8</td>
<td>323</td>
</tr>
<tr>
<td>10</td>
<td>0</td>
<td>318</td>
<td>289</td>
<td>279</td>
<td>52</td>
<td>0.16</td>
<td>63.5</td>
<td>459</td>
</tr>
<tr>
<td>10</td>
<td>1</td>
<td>273</td>
<td>252</td>
<td>240</td>
<td>58</td>
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<td>54.2</td>
<td>354</td>
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<tr>
<td>10</td>
<td>2</td>
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<td>235</td>
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<td>0</td>
<td>253</td>
<td>235</td>
<td>227</td>
<td>60</td>
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<tr>
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<td>230</td>
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<td>171</td>
<td>48</td>
<td>0.23</td>
<td>29.3</td>
<td>323</td>
</tr>
</tbody>
</table>

\( ^* \) Best performing case, that is, lowest \( m_{pl}^* \) for fixed payload mass.
were found to be infeasible given the mission constraints, leaving us with 16 design points. Except for the last column, the results in Table 1 are not scaled and all deliver the same final mass of 125 Mg to Ceres. They differ in payload mass because of the different propulsion system masses, but our interest is in delivering a 75 Mg payload. In Table 1, \( m_1 \) and \( m_2 \) refer to the total spacecraft mass after the elliptical spiral and after the impulsive escape burn, respectively. A useful measure of merit to compare the different missions is the ratio of payload mass to IMLEO. We found that the missions which made the most efficient use of the total initial mass were the same ones that required the least initial mass when all the design points were scaled. The final scaled IMLEO is given in the final column as \( m_{\text{leo}}^s \). For example, in the first row of Table 1, we compute \( \mu_{\text{sc}} = 75/63 = 1.194 \), and then multiply \( m_{\text{leo}} \) by \( \mu_{\text{sc}} \) to get \( m_{\text{leo}}^s = 542 \) Mg. Similarly, \( P^s = 6 \times 1.194 = 7.16 \) MW, \( m_1^s = 470 \) Mg and \( m_2^s = 406 \) Mg (\( V_\infty \) is not scaled, so \( V_\infty = 3 \) km/s).

In Fig. 2, we have taken the data from Table 1 and scaled the masses so that each design point in the plot delivers 75 Mg of usable payload to Ceres. The minimum solution has a power of 11.2 MW (which was scaled up from 10 MW) and a departure \( V_\infty \) of 3 km/s. The results of the supply mission analysis, shown in Fig. 3, indicate that a minimum IMLEO is obtained using a power source of about 2.45 MW. This mission has an interplanetary flight time of 2 years and a circular spiral to depart Earth and capture into Ceres orbit. The mission brings 80 Mg of supplies and return propellant to Ceres and has an IMLEO of 127 Mg.

The supply mission analysis did not examine the effect of departure \( V_\infty \). Since the time constraints and human considerations are far less stringent on the supply mission, a circular spiral is used instead. The circular spiral precludes the use of a chemical escape booster since it cannot leverage a high perigee velocity. The propellant required for the circular spiral stage is computed by MALTO and does not require a separate analysis.

Because it is not evident that a circular escape spiral is more economical than an elliptical escape (as used in the human mission), we perform a simple trade study using an
elliptical spiral escape for the supply mission to examine the effect of this escape method on IMLEO. Using the elliptical escape with no coast arc around apogee, we find that the minimum IMLEO occurs again at $P = 2.45 \text{ MW}$, with an IMLEO of 132 Mg and a flight time of 345 days. This case is both more massive and takes longer than the circular escape method, which has an IMLEO of 127 Mg and escape time of 214 days. We may reduce propellant consumption at the expense of increased flight time by setting a maximum true anomaly past which the spacecraft does not thrust, as described previously. Setting $\theta_{\text{max}} = 135^\circ$ results in an IMLEO of 124 Mg—a savings of 3 Mg over the circular spiral. However these savings come at a cost of a 621 day flight time. Lowering $\theta_{\text{max}}$ to 60° reduces IMLEO to 121 Mg, but the flight time increases to an unacceptable duration of 2318 days.

### 3.4. Design selection

After performing the grid search, we have identified a near-optimal design point for a human mission to Ceres. As we noted earlier, the selected design has an IMLEO of 289 Mg and a power of 11.7 MW and delivers a 75 Mg payload to Ceres. In Table 2 we can see the mass budget for the design after scaling is applied. For the electric propulsion system, the inert mass fraction is 45%. For the overall mission, 47% of the IMLEO is propellant, 27% is inert mass, and 26% is payload. The total IMLEO for the crew mission is 289 Mg, while for the supply mission the total IMLEO is 127 Mg. The combined IMLEO then, with an arbitrary margin of 10%, is 458 Mg. For comparison, the total on-orbit mass of the International Space Station is 450 Mg. We also note that four heavy lift launch vehicles would suffice to enable this mission.

In Table 3 we have the time line for the full mission. The initial supply mission launches in October 2026, the crew mission departs in August 2030, and the crew returns to Earth in May 2032. Since the mission architecture does not involve any gravity-assist bodies, launch opportunities should repeat around the time when Ceres passes the ascending or the descending node, or roughly every 2.3 years.

Fig. 4 depicts the trajectory and events of the crew mission, including the outbound trajectory, the stay on Ceres, and the return trajectory. From a three-dimensional version of the trajectory plot, we learn that the surface operations on Ceres occur when Ceres is near the ecliptic plane and the Sun is between the Earth and the Ceres. A simple analysis for our particular mission showed a minimum Sun–Earth–CTV angle of $2.15^\circ$, which occurs when the CTV is at Ceres. Depending on the communications architecture used, extra communications infrastructure, such as an Earth-trailing satellite, may be required to ensure an uninterrupted link.

### 3.5. Time-of-flight trade study

As we noted earlier, the question of how to constrain the TOF is not easily answered because of the great uncertainty in how the deep-space radiation environment affects the human body. Ideally, we need a way to directly compute the risk of losing a crew member (as a result of exposure to deep-space radiation on any given mission) so we could set a TOF constraint directly based on that risk. However until we are able to perform such an analysis, we resort to simply constraining the time of flight to a feasibly low value.

In Fig. 5, we present the results of a trade study where we vary the time-of-flight constraint while keeping the payload mass and power constant. MALTO was allowed to adjust the launch and arrival dates, the $I_p$, and the stay time on Ceres to minimize the initial mass. The $I_p$ was bounded between 2000 and 8000 s, and the stay time on Ceres was set to a lower bound of 90 days. TOF constraints

### Table 2

<table>
<thead>
<tr>
<th>Item</th>
<th>Mass (Mg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spiral propellant</td>
<td>18.3</td>
</tr>
<tr>
<td>Chemical escape propellant</td>
<td>31.3</td>
</tr>
<tr>
<td>Transfer propellant</td>
<td>87.4</td>
</tr>
<tr>
<td>Chemical escape structure</td>
<td>5.52</td>
</tr>
<tr>
<td>Transfer structure</td>
<td>13.0</td>
</tr>
<tr>
<td>Propulsion inert mass</td>
<td>58.7</td>
</tr>
<tr>
<td>Payload</td>
<td>75.0</td>
</tr>
<tr>
<td>Total</td>
<td>289</td>
</tr>
</tbody>
</table>

### Table 3

<table>
<thead>
<tr>
<th>Event</th>
<th>Date (m/d/y)</th>
</tr>
</thead>
<tbody>
<tr>
<td>STV launches and begins spiral</td>
<td>10/19/2026</td>
</tr>
<tr>
<td>Supply mission departs</td>
<td>5/27/2027</td>
</tr>
<tr>
<td>Supply mission arrives</td>
<td>5/21/2029</td>
</tr>
<tr>
<td>Crew vehicle begins spiral</td>
<td>Before 7/19/2028*</td>
</tr>
<tr>
<td>CTV begins interplanetary leg</td>
<td>8/6/2030</td>
</tr>
<tr>
<td>CTV arrives on Ceres</td>
<td>5/3/2031</td>
</tr>
<tr>
<td>CTV departs Ceres</td>
<td>8/23/2031</td>
</tr>
<tr>
<td>CTV returns to Earth</td>
<td>5/19/2032</td>
</tr>
</tbody>
</table>

* Spiral phase takes 748 days to complete as designed.
reliable method of quantifying the radiation health risk. Such that overall risk to the astronauts is reduced. We longer TOF can be reassigned to bolster radiation shielding so the question becomes whether the mass savings of a constraint will require more propellant and higher IMLEO, general it is reasonable to assume that a shorter TOF TOF constraint as a third design variable. However, in 240 days. For the case study in this paper a TOF of 270 days is assumed.

Fig. 5. The IMLEO increases dramatically for TOF constraints lower than 240 days. For the case study in this paper a TOF of 270 days is assumed.

ranged from 240 days to 360 days in increments of 30 days. For each TOF constraint setting, we calculated the total IMLEO in the same manner as with the initial grid search.

What we see is that the 270-day TOF constraint appears adequate given the assumptions of the trade study. The phasing of the mission is a likely factor preventing significant reductions in IMLEO for longer TOF constraints. MALTO is given freedom to choose the launch dates in the mission, and it has chosen the dates such that the spacecraft arrives and departs Ceres while the dwarf planet is near the ascending node of its orbit. Arriving at Ceres while it is near the ascending node means that most of the plane change required to reach Ceres can be performed at a greater heliocentric distance, where less propellant is needed. Subsequently, when the TOF constraint is loosened, MALTO reduces the time spent on the surface of Ceres so that the arrival still occurs near the ascending node of the orbit, until the lower constraint on stay time (30 days) is reached. When no more time can be taken away from the surface operations and diverted to the transfer, the launch dates must be altered and there is less benefit to an increased TOF.

A more thorough analysis could be achieved by performing an entirely new grid search over power and departure $V_{\infty}$ for each constraint, in essence adding the TOF constraint as a third design variable. However, in general it is reasonable to assume that a shorter TOF constraint will require more propellant and higher IMLEO, so the question becomes whether the mass savings of a longer TOF can be reassigned to bolster radiation shielding such that overall risk to the astronauts is reduced. We must, of necessity, leave this question open pending a reliable method of quantifying the radiation health risk.

4. Discussion

4.1. Key technologies to develop

The primary technology enabling a human mission to Ceres is a nuclear power system capable of generating 11.7 MW of power with a specific mass of 5 kg/kW. A mission may be feasible with a smaller power system of around 8–9 MW, however more propellant and a higher IMLEO would be required. In addition to the power system itself, an electric propulsion technology capable of converting that power into thrust with an efficiency of about 70% is needed.

4.2. In situ resource utilization vs pre-delivered propellant

Given that significant quantities of water ice exist in the regolith of Ceres (i.e. between 3% and 20% of the mass of the regolith), it is possible that a mission to Ceres could use that water to produce propellant for the return mission through electrolysis. The option of in situ resource utilization would be attractive if the cost of sending the required equipment would be less than that of sending return propellant to Ceres. The in situ propellant could feed a liquid-oxygen, liquid-hydrogen chemical propulsion system, or it may serve as the propellant for an electric system in the form of $H_2$. In the latter case, the electric propulsion system would be limited to high specific impulse because of the very low atomic mass of hydrogen.

5. Conclusion

We can draw the following conclusions from this mission design study.

1. A human mission to Ceres could be made feasible with the appropriate investment in propulsion and in-space power technology. Given the assumptions used here, the total IMLEO, with a 10% margin, would be 458 Mg. The mission architecture presented here would deliver a total of 155 Mg of payload to Ceres over two missions. Four heavy-lift launch vehicles would suffice to carry out such a mission.

2. Nuclear electric propulsion technology enables human exploration at Ceres because it has a relatively low specific mass (i.e. about 5 kg/kW) and it avoids a costly impulsive capture maneuver at Ceres. Electric propulsion technologies capable of processing input power up to 11.7 MW (or more) should be further developed to open the possibility of exploring Ceres.

3. Total mission times of less than 2 years (for the crew) are possible with nuclear electric propulsion. In the absence of a proven method of blocking deep-space radiation, limiting mission times is the best way to limit the danger to the crew.

References


