Acta Astronautica 95 (2014) 124–132

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Low-thrust trajectories for human missions to Ceres

Frank E. Laipert*, James M. Longuski

Purdue University, School of Aeronautics and Astronautics, 701 W. Stadium Ave., West Lafayette, IN 47907-2045, United States

ARTICLE INFO

Article history:
Received 13 February 2013
Received in revised form
18 October 2013
Accepted 2 November 2013
Available online 8 November 2013

Keywords:
Astrodynamics
Human space exploration
Low-thrust mission design
Ceres

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_∞, 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_∞ 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

* Corresponding author. Tel.: +1 630 670 7548.
E-mail addresses: frank.laipert@outlook.com, flaipert@purdue.edu (F.E. Laipert), longuski@purdue.edu (J.M. Longuski).

0094-5765/$ - see front matter © 2013 IAA. Published by Elsevier Ltd. All rights reserved.
http://dx.doi.org/10.1016/j.actaastro.2013.11.003

Nomenclature

V_∞	hyperbolic excess velocity, km/s
e	eccentricity
Ω	longitude of the ascending node, rad
ω	argument of periapsis, rad
θ^*	true anomaly, rad
p	semi-latus rectum, km
L	true longitude $\Omega+\omega+\theta^*$, rad
i	inclination, rad
f	modified equinoctial element $e\cos(\omega+\Omega)$
g	modified equinoctial element $e\sin(\omega+\Omega)$
h	modified equinoctial element $\tan(i/2)\cos\Omega$
k	modified equinoctial element $\tan(i/2)\sin\Omega$
a_i	ith acceleration component, km/s^2
a	semi-major axis, km
β	steering angle, rad
m_{pl}	usable payload mass, Mg
m_f	total final mass at Ceres, Mg
α	propulsion system specific mass, kg/kW
P	spacecraft power, MW
μ_p	propellant tank mass factor
m_p	propellant mass, Mg
T	thrust, N
μ_{sc}	trajectory scaling factor
m_{leo}	initial mass in low Earth orbit, Mg
m_1	total mass after spiral, Mg
m_2	total mass after chemical escape, Mg

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_∞, 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_∞ 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_∞ 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 $\alpha=5$ kg/kW. This