Research Article

Outer Planet Missions with Electric Propulsion Systems—Part I

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For interplanetary missions, efficient electric propulsion systems can be used to increase the mass delivered to the destination. Outer planet exploration has experienced new interest with the launch of the Cassini and New Horizons Missions. At the present, new technologies are studied for better use of electric propulsion systems in missions to the outer planets. This paper presents low-thrust trajectories using the method of the transporting trajectory to Uranus, Neptune, and Pluto. They use nuclear and radio isotopic electric propulsion. These direct transfers have continuous electric propulsion of low power along the entire trajectory. The main goal of the paper is to optimize the transfers, that is, to provide maximum mass to be delivered to the outer planets.

1. Introduction

In October 1998, NASA launched Deep Space 1 (DS1) that was the first deep space mission having been propelled by solar electric propulsion [1, 2]. Later, Smart-1 (ESA-2003) and Hayabusa (Japan-2003) were launched. With the successful demonstration made by the Deep Space 1, many studies have been performed to show the applicability and performance of Solar Electric Propulsion (SEP) for deep space missions [3].

A Neptune Orbiter mission is examined utilizing single propulsion systems and combinations of Solar Electric Propulsion (SEP), Radioisotope Electric Propulsion (REP), and chemical systems to compare the concepts [5]. Electric propulsion systems, while

Figure 1: (a) Electric power level versus duration of use and (b) Solar energy flux versus distance from the Sun (Schmidt and Houts [4]).

highly efficient, produce only a small thrust value; this is why electric propulsion is also called low thrust. As a consequence, the engine operates in long arcs of the trajectory.

Several U.S.A space missions used Radioisotope Electric Propulsion (REP) system. In particular, some of them used generator of the radioisotope thermoelectric type (Lincoln Experimental Satellite "LES" 8 and 9, Voyager 1 and 2, Galileo, Ulysses, Cassini, New Horizons). Nuclear Electric Propulsion (NEP) uses a reactor power system to provide the electricity for thrusters that ionize and accelerate propellant to produce thrust. The combination of nuclear electric power systems and electric thrusters has been studied by

Figure 2: Ratio m_f/m_i (continuous lines) and ΔV (dashed lines), considering REP for a mission to Uranus, with effective specific power (P) = 6.67 W/kg, TOF = 10 years, $V_{\text{inf}} = 6$ km/s. The dates are (a) 2020–2025 and (b) 2026–2030.

NASA and other agencies since the 1950s. Reactor power system design and development has received considerable attention, despite only one U.S space flight experience, the SNAP-10A (Systems Nuclear Auxiliary Power Program).

Figure 1(a) shows the availability of energy of several sources. Notice that the radio isotopic (REP) and nuclear electric propulsion (NEP) have more advantage when compared

Figure 3: Ratio m_f/m_i (continuous lines) and ΔV (dashed lines), considering NEP for a mission to Uranus, with effective specific power (P) = 40 W/kg, TOF = 10 years, V_{inf} = 6 km/s.

to the solar electric propulsion (SEP) for a long time trip. The variation of the solar power with the distance from the Sun is shown on Figure 1(b).

2. Goal of the Study

The main goal of the present research is to compare low-thrust transfers (limited power) to Uranus, Neptune, and Pluto with different power sources, such as REP and NEP.

The method of the transporting trajectory was used for the optimization. A spacecraft flight between two given positions for a given time in a central gravity field using the low thrust is considered. The equations of the spacecraft motion are linearized near a Keplerian transfer orbit (reference orbit or transporting trajectory). The optimal vector of the low thrust is obtained analytically for the linearized equations using a general solution to the conjugate variational equation. This vector can be used as a first guess to the accurate solution of the problem; the corresponding propellant consumption gives a lower limit for the low thrust subject to constraints. To improve the linearization accuracy, it is possible to split the times of flight into subintervals and then solving the problem separately on every subinterval. Later, they are joined together using the condition of minimizing the performance index of the original problem [6–8].

3. Basic Assumptions

In this paper, the decay of the nuclear/radioisotope source is not considered. The nuclear/radioisotope power is considered constant. The unit of power used in this paper is the effective specific power (P), that is the *(efficiency) × (total power/total mass)*. Besides, for

Figure 4: Ratio m_f/m_i for several TOF for a trip to Uranus. The effective specific power (P) used is (a) 4 W/kg (REP), (b) 6.67 W/kg (REP), (c) 40 W/kg (NEP), and (d) 50 W/kg (NEP).

all cases, the efficiency is considered 100% (ideally cases). The planetary state vector obtained from the planetary ephemerides (J2000) is used, together with Chebyshev's interpolation. For all cases, it is assumed that the spacecraft leaves the Earth's sphere of influence with variable velocity at infinity (V_{inf}) and approaches the targets planets (Uranus, Neptune, Pluto) with zero velocity at infinity. Several analyses were considered as a function of the ratio between the final and initial mass, time of flight (TOF), ΔV , velocity at infinity, and effective specific power.

With the goal of finding the optimal dates between 2020–2030, several figures, as a function of the ratio between the final and initial mass and ΔV versus time of flight, were made considering that the velocity at infinity near Earth is 6 km/s and time of flight is 10

Figure 5: Ratio m_f/m_i (continuous lines) and ΔV (dashed lines), considering REP for a mission to Neptune, with effective specific power (P) = 6.67 W/kg, TOF = 10 years, V_{inf} = 6 km/s.

Figure 6: Ratio m_f/m_i (continuous lines) and ΔV (dashed lines), considering NEP for a mission to Neptune, with effective specific power (P) = 40 W/kg, TOF = 10 years, V_{inf} = 6 km/s.

Figure 7: Ratio m_f/m_i for several TOF for a trip to Neptune. The effective specific power (P) used is (a) 4 W/kg (REP), (b) 6.67 W/kg (REP), (c) 40 W/kg (NEP), (d) 50 W/kg (NEP).

years (Figures 2, 3, 5, 6, 8, and 9). With this information, the ratio between the final and initial mass versus velocity at infinity near Earth for several times of flight was studied (Figures 4, 7, 10).

Experimental and theoretical investigations of nuclear and radioisotope power system have been carried out over several years [9–16]. The numerical simulations in this paper used for REP, an effective specific power of 6.67 W/kg, (see [17]) and for NEP, an effective specific power of 40 W/kg (see [18]). Other values ideally used were 4 W/kg and 50 W/kg respectively.

Figure 8: Ratio m_f/m_i and ΔV , considering REP for a mission to Pluto, with effective specific power (P) = 6.67 W/kg, TOF = 10 years, $V_{inf} = 6$ km/s.

4. Mission Analysis

4.1. Mission to Uranus

The seventh planet from the Sun is so distant that it takes 84 years to complete one orbit. Uranus, with no solid surface, is one of the gas giant planets (the others are Jupiter, Saturn, and Neptune). NASA's Voyager 2 spacecraft flew very close to Uranus, in January 1986. Figures 2 and 3 study the problem of the optimal date for maximum mass to be delivered to Uranus between 2020–2030.

The maximum ratio m_f/m_i is obtained for launching in June 07, 2030 (Figures 2 and 3). After obtaining the optimal date, on Figure 4, it is possible to see the behavior of the ratio m_f/m_i versus velocity at infinity near Earth for several times of flights. For larger power available, more mass can be delivered.

4.2. Mission to Neptune

Voyager 2 visited Neptune on August 25, 1989. Neptune is a very interesting object for science because of its turbulent atmosphere and the presence of the large moon Triton. Triton is particularly interesting because of its size and retrograde orbit; moreover the insight into Solar System evolution to be gained through its comparative relationship with Pluto and Charon.

Figures 5 and 6 show that the best launch date that maximize the mass to be delivered to Neptune is April 06, 2030. For clarity, Figures 5 and 6 show some years between 2020–2030. The inverse behavior between ΔV and mass ratio is again found. The use of nuclear power

Figure 9: Ratio m_f/m_i (continuous lines) and ΔV (dashed lines), considering NEP for a mission to Pluto, with effective specific power (P) = 40 W/kg, TOF = 10 years, $V_{\text{inf}} = 6$ km/s.

allows the delivery of larger masses to the planet, due to the high power of this power supply system. So, this is a good way to transport large payloads to Neptune.

After obtaining the optimal date, Figure 7 shows the m_f/m_i ratio versus velocity at infinity near Earth for several times of flights. As expected, larger power allows the delivery of larger mass to Neptune.

4.3. Mission to Pluto

On August 24, 2006, the International Astronomical Union (IAU) formally downgraded Pluto from an official planet to a dwarf planet. No spacecraft have yet visited Pluto. However, NASA launched a mission called New Horizons that will explore both Pluto and the Kuiper Belt region. Launched on January 19, 2006, the spacecraft performed a Jupiter gravity assist in February of 2007. Today, it is traveling to Pluto and it will encounter the planet on 14 of July, 2015.

Now we study some possible ways to reach Pluto with electric propulsion. Figure 8 shows the curves of the maximum mass that can be delivered to Pluto. For clarity, some results are shown for the years between 2020 and 2030. The optimal date is on February 06, 2020. The curves have some minima and maxima, but the optimal date allows to maximize the mass delivered to Pluto and to minimize the propellant consumption. The same analysis is made for NEP system and the results are shown in Figure 9. Figure 9 also shows that, depending on the launch date, the propellant consumption can be high and the mass delivered can be a minimum. High velocity at infinity near Earth (depending on the characteristic of launch vehicle) allows an economy on the consumption of propellant and there is a maximum in the total mass delivered to the planet. This analysis, as function of time of flight, is shown on Figure 10.

Figure 10: Ratio m_f/m_i for several TOF for a trip to Pluto. The effective specific power (P) used is (a) 4 W/kg (REP) , (b) 6.67 W/kg (REP) , (c) 40 W/kg (NEP) , (d) 50 W/kg (NEP) .

5. Conclusions

Low-thrust transfers of the limited power (LP) type was considered in this paper. In order to find the optimal trajectory of a transfer with LP thrust, it is suggested to use the method of the transporting trajectory. No constraint is imposed on the thrust vector direction. A maximum power provides a minimum propellant consumption. Then we searched for the optimal date between 2020–2030 that allow maximum delivered mass to the outer planets. With this optimal date, are analyzed the effect of the velocity at infinity near Earth and the time of flight on the delivered mass. Larger times and higher velocity at infinity allow maximum mass delivered. Due to the small masses delivered, solar electric propulsion was not used for the trip to outer planet.

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