SOLAR ELECTRIC PROPULSION PERFORMANCE FOR MEDLITE AND DELTA CLASS PLANETARY MISSIONS

Carl G. Sauer, Jr*
Member of the Technical Staff
Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California 91109

ABSTRACT

'l'he current emphasis on small, low-cost planetary missions using Delta and Medlite Class launch vehicles has prompted the examination of the use of Solar Electric Propulsion (SEP) spacecraft to either enable or enhance the performance of some of the more demanding planetary missions. Planetary missions that appear most attractive for a small solar electric propulsion system include those missions that require a large post-launch AV commitment from the spacecraft propulsion system such as small body rendezvous and sample return missions. Other missions that may benefit from use of SEP would be a Mercury orbiter mission and various outer planet orbiter and flyby missions. The use of SEP for this latter class of missions could result in either increased performance *or* use of a smaller, lower cost launch vehicle as compared with the more conventional ballistic mission.

Preliminary estimates of planetary mission performance for this class of SEP spacecraft were presented in two technical papers by the author[1,2] in 1993 and 1994. In the above references trajectories were calculated based on a conceptually simple model of the propulsion system using a constant specific impulse and efficiency. Although the results presented in these papers demonstrated the feasibility of using this technology for small planetary missions, better knowledge of the actual delivery capability using real thruster throttling performance is necessary for more detailed mission studies.

Extensive measurements of the throttling behavior of electric propulsion 2.5 kW 30 cm ion thrusters (NSTAR) have been made at the NASA Lewis Research Center in Cleveland and at the Jet Propulsion Laboratory in Pasadena to support a technology verification of Solar Electric Propulsion on the New Millennium Deep Space 1 mission (DS 1). Based upon these measurements, realistic SEP throttle performance has been incorporated into the trajectory optimization software currently being used for SEP mission studies. The modeling of the throttling behavior consists of approximating both thrust and mass flowrate as polynomial functions of power processor input power. An example of polynomial fits to some early measurements of thruster throttling is shown in figures 1 and 2. In these figures a linear fit of thrust force and mass flowrate to power processor input power was sufficient to approximate

* Senior member AAS and AIAA Tel, (818) 354-4875, Fax (818) 393-9900
Email: cgs@sauer.jpl.nasa.gov

the throttle behavior. The specific impulse and overall thruster efficiency which result from the thrust and mass flowrate shown in figures 1 and 2 are shown in figures 3 and 4 respectively and serve to indicate the degree that the thruster performance degrades at the lower throttle levels. Since some of the planetary missions being examined rely on low power operation of these thrusters, it is important to model thruster performance to an accuracy sufficient to get reliable estimates of delivery capability. This paper describes the inclusion of thruster throttling into the trajectory optimization code and discusses the necessary conditions to be satisfied for optimization of thruster staging (I E. changing the number of operating thrusters) during the trajectory.

Several different thruster combinations have been used in these missions studies depending upon the launch vehicle and array power used. In genera] the 2.5 kW thrusters are oversized for some of the small spacecraft that are being considered for launch from a Medlite class launch vehicle. As a consequence only one operating thruster is considered for these spacecraft with a solar array power level in the range of 2-5 kW. Larger Delta 7925 class launch vehicles will have nearly double the injection capability and two operating 30 cm thrusters with an array power level of 5-8 kW can be used. These 2.5 kW thrusters are currently being designed to cover a throttle range of approximately 5 to 1. For some missions, such as a Ceres rendezvous mission, the variation of solar array output power can exceed this range and the solar array power must be increased so that sufficient power is available at the maximum required thrusting distance.

Optimized Solar Electric Propulsion trajectories have been calculated for numerous planetary missions during the past several years in order to determine the SEP delivery capability. A sample of these missions are described in this paper and include the following representative missions:

1. A comet Kopff rendezvous mission
2. An asteroid Vesta rendezvous mission
3. An asteroid Ceres rendezvous mission with and without a Mars gravity assist.
4. A Pluto flyby mission using two Venus and a Jupiter gravity assist.
5. A Mercury orbiter mission using a single Venus gravity assist.
6. A Jupiter flyby/orbiter mission using an Earth gravity assist.
7. A Uranus flyby/orbiter mission using an Earth gravity assist.
8. A comet Tempel 1 rendezvous and sample return mission.

A brief description of four of the above missions is included.

Comet Kopff Rendezvous: A heliocentric ecliptic plot of this trajectory is shown in figure 5. This spacecraft would be launched on a Medlite* launch vehicle in May 2000 and rendezvous with Kopff in July 2003. A 3.375 kW silicon solar array and a single operating 2.5 kW thruster and would be capable of delivering a total spacecraft dry mass of slightly over 300 kg with a propellant expenditure of 128 kg of Xenon. Rendezvous with the comet is around 220 days past perihelion after a flight time of 3.2 years. There are two optimized coast phases in this trajectory with a short thrust phase between them. Estimates of the SEP system mass

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*The Medlite launch vehicle is assumed to be a Delta 7326 range from around 150 to 200 kg and would result in a net, spacecraft mass of somewhere around]00-150 kg.

Asteroid Ceres rendezvous: An ecliptic projection of the trajectory for this 3 year Ceres rendezvous mission is shown in figure 6. The spacecraft would be launched on a Medlite launch vehicle in May 2003 and employ a Mars gravity assist midway through the trajectory to improve performance. In order to provide sufficient power to a single 2.5 kW thruster at the distance of Ceres the power level of the solar array must be increased to 5 kW. As a consequence, the SEP mass will be slightly higher than that for the Kopff mission due to the increased size of the solar array. There is only a single coast phase of around 70 days near the end of the trajectory. The total dry spacecraft mass for this mission is slightly under 300 kg and the propellant expenditure is 150 kg, both masses similar to that of the Kopff mission.

Mercury rendezvous: An ecliptic projection of the path of the spacecraft is shown in figure 7 for this 800 day Mercury mission. The spacecraft would be launched in August 2002 using a Delta 7325 launch vehicle and arrive in November 2004. Like the previous missions, this spacecraft would use a single operating 2.5 kW thruster with a solar array power level of 2.6 kW. Although the array power level increases as the spacecraft goes closer to the sun, the expected degradation of the array will probably have the effect decreasing the output power to about its initial value. Because of the decrease of efficiency of the solar array at low solar distances, it is necessary to feather the array at distances less than around .7 AU so as to constrain the maximum array temperature.

I n this example a Venus gravity assist in March 2003 is used to decrease the perihelion distance of the transfer trajectory to nearly that of Mercury. The spacecraft thrusts mostly around perihelion in order to decrease spacecraft aphelion distance to that of Mercury. Note the presence of coast arcs around spacecraft aphelion indicating that thrusting at this time is not effective. The injected mass of slightly over 600 kg is higher than for the previous two examples and the thrust acceleration level is thus lower. Previous studies of Mercury rendezvous trajectories employed a higher thrust acceleration and had transfer time of 600 to 700 days without the Venus gravity assist. The propellant expenditure for this example is slightly over 190 kg, requiring at least 3 thrusters to satisfy present lifetime constraints on the 30 cm thrusters. As a consequence the propulsion system mass will be higher for this spacecraft as compared with those in the previous examples, not only because of additional thrusters but possibly also due to additional thermal protection required because of the high temperatures encountered on this mission.

Pluto flyby: The last mission presented in this abstract is a 10 year flyby mission to Pluto and involves two gravity assists at Venus and a gravity assist at Jupiter. An ecliptic projection of the initial phase of the mission is shown in figure 8. Only this phase is shown in order to illustrate the thrusting phases of the trajectory. This trajectory is similar to a ballistic trajectory using the same gravity assist bodies and the intent was to see if an adequate payload could be delivered using a small and less expensive launch vehicle than that required for a ballistic mission. Similar power and propulsion parameters are used for this mission as for the Kopff mission, that is, one operating 2.5 kW thruster and a 3.375 kW solar array. Like the first two missions, this spacecraft is launched on a Medlite launch vehicle. The spacecraft energy requirements are actually less for this Pluto mission than for the two small body rendezvous missions the propellant expended is less, being slightly over 100 kg. The total delivered spacecraft and SEP mass is approximately 400 kg.

The proposed paper will go into greater detail for these missions and the other missions mentioned previously. The paper will also include a comprehensive definition of the modeling of the thruster throttling in the trajectory optimization software.

1. Sauer, Carl G., Jr, *Planetary Mission Performance for Small Solar Electric Propulsion Spacecraft,* Paper AAS 93-561, AAS/AIAA Astrodynamics Specialist Conference, Victoria, BC, Canada, Aug 16-19, 1993

2. Sauer, Carl G. Sauer, Jr. and Yen, Chen-Wan 1,., *Planetary Mission Capability of Small Low Power Solar Electric Propulsion Systems,* Paper IAA-L-0706, IAA International Conference on Low-Cost Planetary Missions, John Hopkins Applied Physics Laboratory, Laurel, MD, April 12-15, 1994

Figure 1. Thrust Level

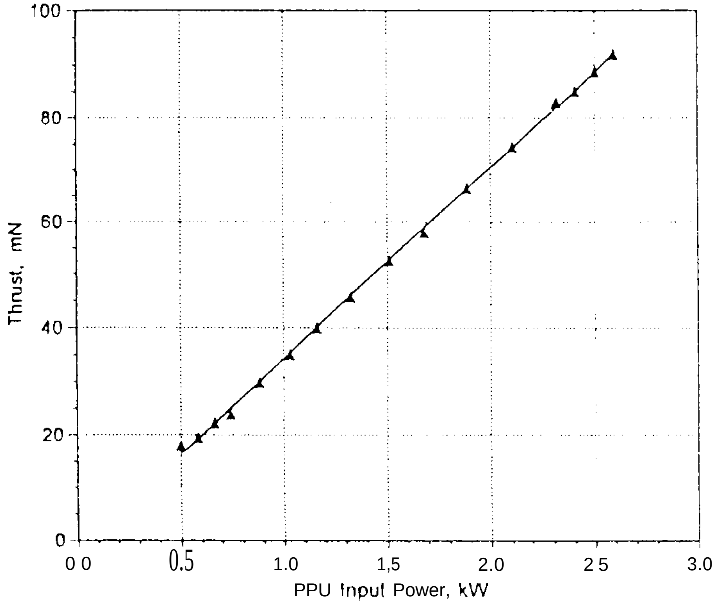


Figure 2. Mass Flow Rate

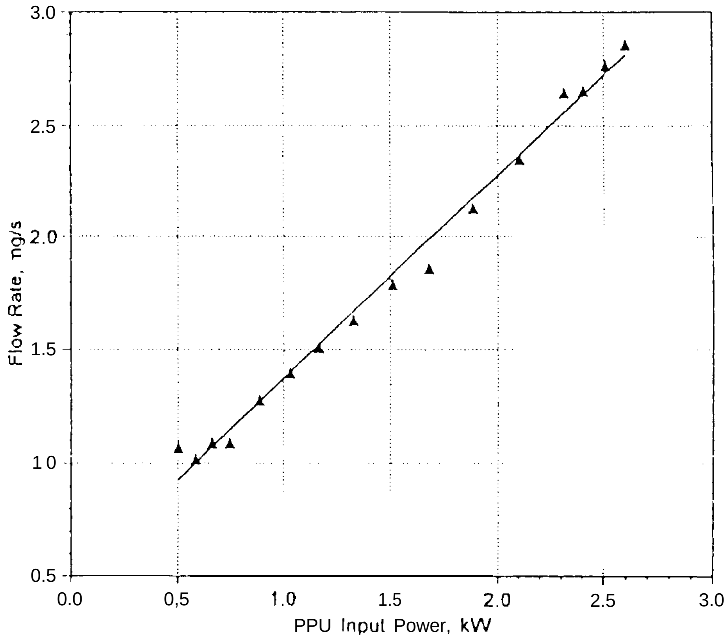


Figure 3. Specific Impulse

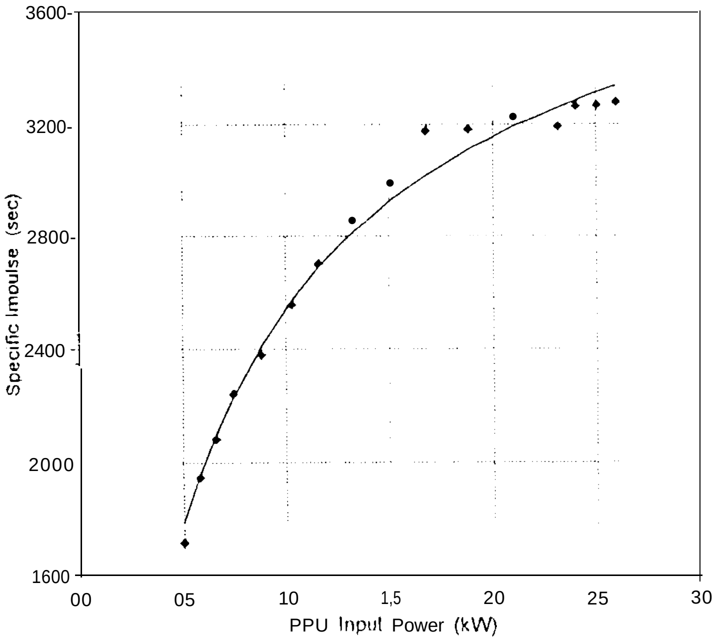


Figure 4. Overall Efficiency

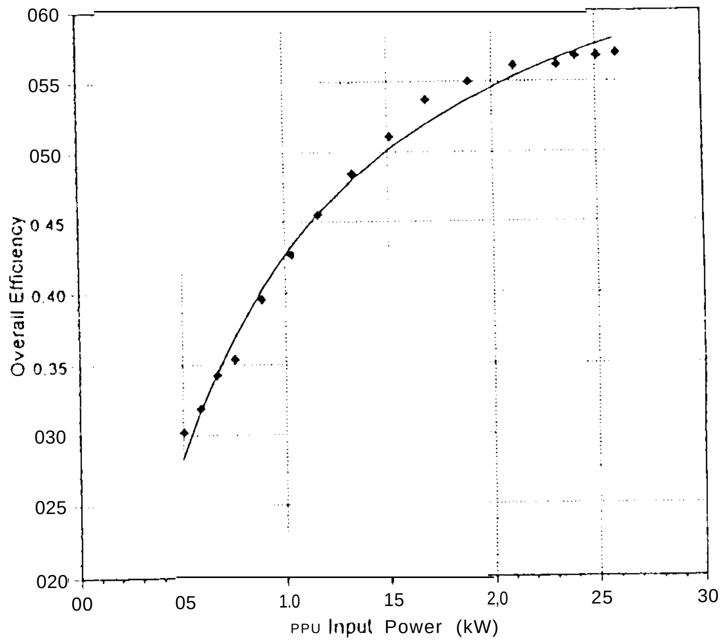
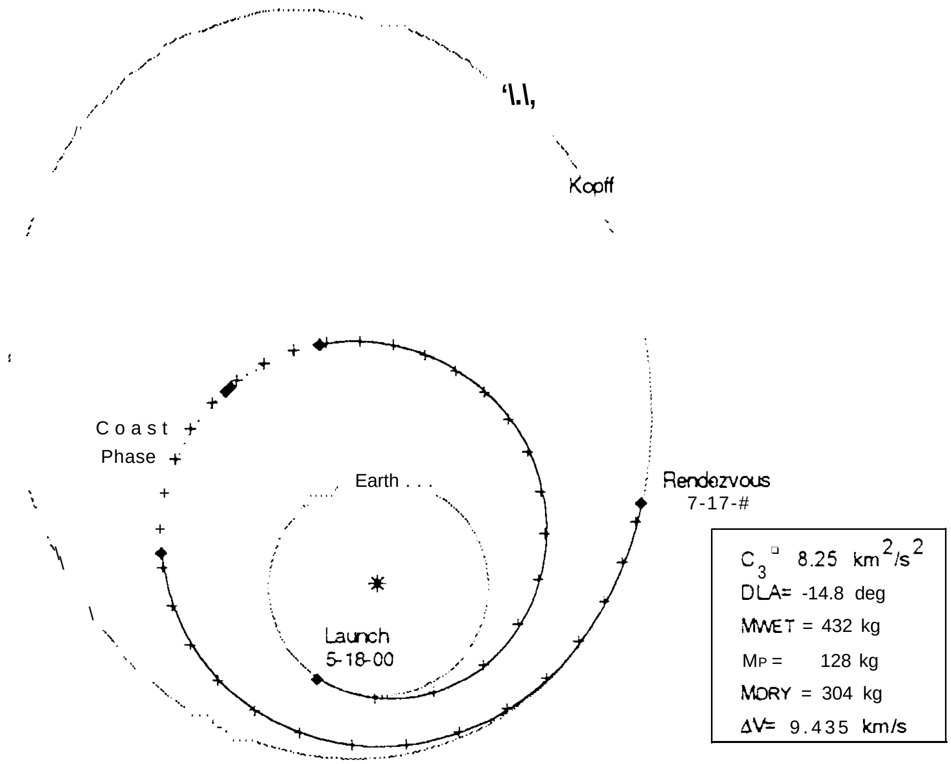
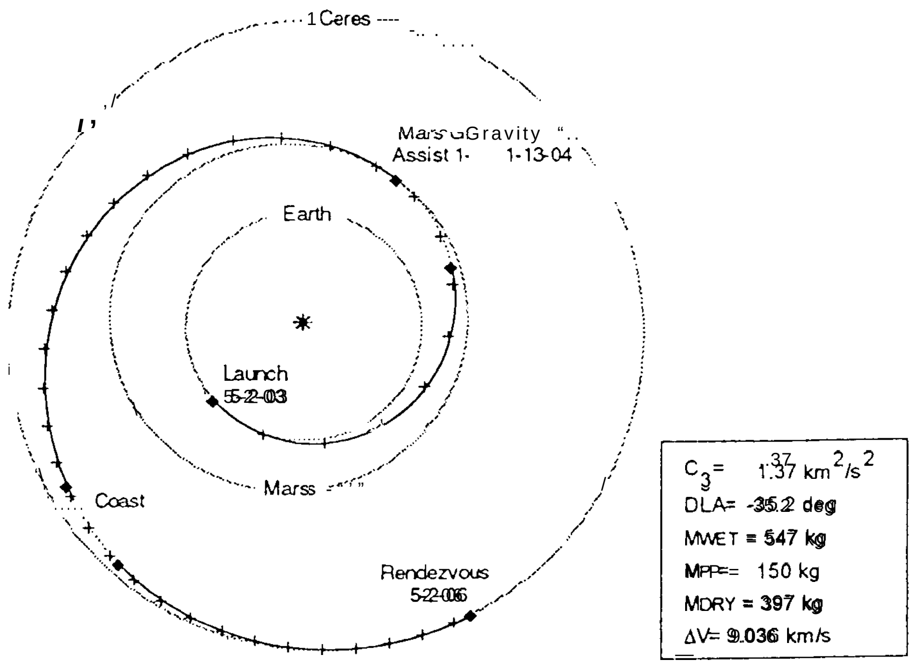


Figure 5.
20003 .2 Year Kopff Rendezvous
Delta 7326 / SEP 3.375 kW



33 day tics on spacecraft path

Figure 6.
2001 3 Year Ceres Rendezvous
Delta 7326 / SEP 5 kW



33 day tics on spacecraft path

Figure 7.
2002 Earth-Venus-Mercury Rendezvous
800 Day Flight Time
Delta 7325 / SEP -2.6 kW

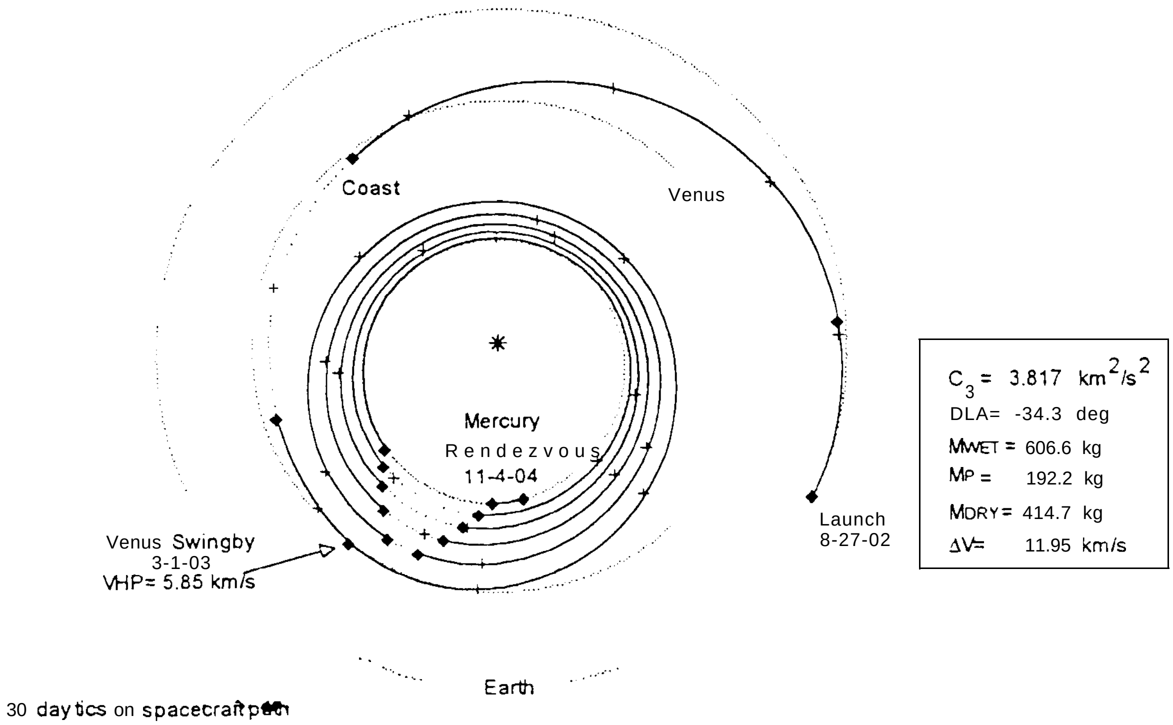


Figure *8.*
10.2 **Year VVJGA Pluto Flyby**
Delta 7326 / SEP -3.375 kW