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Over Powering Solar System Exploration

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OVER POWERING SOLAR SYSTEM EXPLORATION

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ABSTRACT

This paper describes the new mission possibilities and technology requirements for a high power Solar Electric Propulsion (SEP) system using a Taurus-class launch vehicle. A 10-kW Hall Effect Thruster with Anode Layer (TAL) running direct-drive off a high voltage solar array could produce enough thrust to enable very rapid missions with a small sciencecraft to various destinations throughout the solar system. After defining the propulsion system characteristics, a palette of different missions were studied to quantify the total wet spacecraft mass and corresponding mission trip times. The missions considered include two main-belt asteroid rendezvous (Vests and Ceres), and a comet rendezvous (Kopff), and with launches to either low Earth orbit (LEO) or Earth escape. Results show that this high power SEP system first enables LEO to escape trajectories to be performed in about 2-3 months, therefore increasing the effective mass capability of small launchers, and second can be used for very rapid transportation of small spacecraft (- 50 kg) to the mainbelt asteroids and short-period comets. Typical flight times for such missions vary from 1.0 to 1.5 years, which is half the time that the New Millennium DS-1 SEP **system (NSTAR) technology would offer.**

Introduction

A new era of deep-space exploration is starting with the flight in 1998 of the New Millennium Deep-Space 1 mission. This mission will be performed with a 30-cm diameter 2.3 kW ion engine, called NSTAR[1] (NASA SEP Technology Application Readiness), and fly-by and asteroid and a comet. For the first time ion propulsion will be used in a planetary mission and if successful, will open the doors to new mission opportunities. A fair amount of work has atready been done to look at potential future *missions* that the NSTAR system could enable. A step further is taken in this study by looking at a high power electric propulsion system that would enable small sciencecraft missions to be launched from a small launcher and that would feature short trip times to their destinations. Short trip times typically means here about half the time that could be obtained with an NSTAR-based ion propulsion system (which already reduces the trip time to these destinations by roughly a factor of two relative to chemical propulsion).

This paper will first describe the propulsion system assumptions that formed the basis of this study, and then look at different mission options and their requirements on the system performances.

Propulsion System Description

The Solar Electric Propulsion (SEP) system envisioned to be suited for future short trip time missions to small bodies and planets is composed of a single Hall Effect Thruster with Anode Layer (TAL) because of its ability to process large amounts of power with a high

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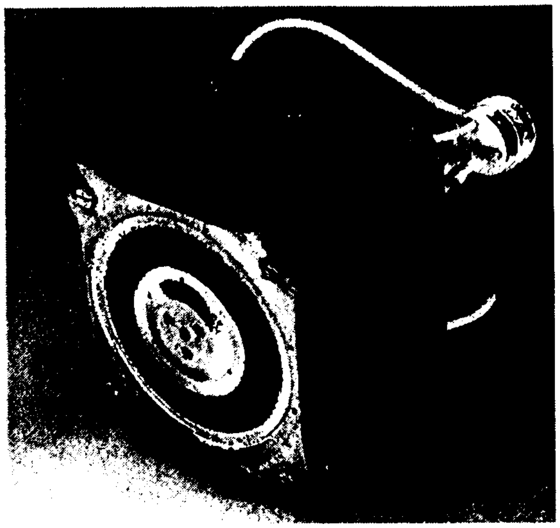

Fig. 1: Picture of the single stage TAL D-1OO

efficiency, and of an advanced lightweight solar array that would be designed to run at high voltages. The thruster would be directly driven off of the solar array, therefore significantly reducing the complexity and mass of the Power Processing Unit (PPU). This section summarizes the technology and assumptions made on the propulsion system.

Thruster with Anode Layer (TAL)

To reach planets and small bodies such as the main-belt asteroids or comets in a relatively short trip time, the performances of the thruster call for high power, high efficiency and relatively high lifetimes (between 1.0 to 1.5 years, or in other terms between about 120 and 500 kg of propellant throughput). The Hall Effect are potential candidates for these applications since they ean process high power in a small lightweight package. A detailed description of the Hall thruster physics and technology can be found in literature[2,3] and will not be discussed here. One key issue that rises concerning TAL technology is the relatively high lifetime requirement and future developments will have to be made to resolve it.

The power level assumed in this study was nominally 10 kW. This level appeared to be a good compromise between relatively high thrust and low solar array masses. Current single stage or double stage TAL technology such as the TAL D-100 (Fig. 1) developed by TsNIIMASH[4,5] show total thruster efficiencies (including cathode flow rates) ranging from 0.6 to 0.7 with specific impulses (Isp) varying from 2000 to 4000 seconds, and have shown to be capable of a throttling range over 5:1. Experiments[6] have also shown that in general for these thrusters, the higher the discharge voltage, the higher the thruster efficiency, which will somewhat drive the solar array design. In these tests the propellant used was xenon. It is to be noted also that TALs have the advantage of requiring essentially a single voltage input to operate steady state, simplifying the design of the power supplies (small low-power electromagnet supplies are also needed). The current is controlled by the propellant flow rate.

The requirements on the operating point of the thruster are related to the type of mission trajectory and total AV, and to the characteristics of the thruster. It is assumed here that this operating point can be modulated within the advertised boundaries and optimized for the trajectory found.

Power system

Direct-drive solar array technology is slowly emerging and shows some great potential for SEP systems[7,8]. A direct drive system greatly reduces the demand and therefore mass of the power processing system by directly connecting the thruster load to the solar array.

High voltage solar arrays (100 V range) are currently being developed mostly by the commercial satellites companies along with high voltage buses. For planetary exploration, the SCARLET[9] concentrator array (developed by AEC-Able Engineering Co., Inc.) will run between 90 and 120 V for the duration of the New Millennium DS-1 mission. This array (in its current conservative design for DS - 1) shows performances around 50 W/kg. For future applications, expectations are that 70-80 W/kg is achievable for this concentrator array. For SEP direct drive applications, the array voltages need to be in the 500-1000 V to yield the desired Isp. Running arrays at these voltages might require better protection (coverglass) to avoid or diminish the arching effects. The PASP+ experiment[10] tested several cell technologies in space for one year at +/-500 V and showed that concentrator arrays were quite stable and resistant to radiation arrd to plasma interaction without much changes in their design. It can, therefore, be expected that if the demand is strong, high voltage direct-drive solar (concentrator) arrays could be developed and that high specific masses could be obtained. In this study, a specific mass of 100 W/kg (at the array level, including contingency) was assumed.

Onc concern though might be that the array voltage will increase as the spacecraft goes further from the sun (20-30% increase can be expected) due to lower array temperatures. To compensate for this voltage increase, the mass flow rate must be adjusted. The details of the thruster tuning are beyond the scope of this work, but to be consistent with the generated constant specific impulse trajectories, the thruster must be throttled at constant Isp with an increasing input voltage.

The Power Processing Unit (PPU) typically represents a large fraction of the system dry mass. With the direct-drive option, the complexity of the PPU is reduced to an EMI filter and a switch, that insures EMI compatibility with the spacecraft and a matching network, which is tuned to optimally run the thruster. A PPU efficiency of 0.95 was assumed here, which is consistent with current analysis[8], although actual measurements in laboratory show an efficiency of 0.99" for a 10-kW PPU running in the unregulated mode. The mass of the PPU hardware was assumed to be 0.5 kg/kW, still based on a developed laboratory hardware design".

Feed system

Significant mass savings were assumed in the feed system hardware relative to the NSTAR feed system by assuming the use of micro-gas rheostat and magnetostrictive valves that are currently being developed.

Propulsion system and spacecraft mass summary

Table 1 summarizes the propulsion system masses and spacecraft mass assumed in this study. Most of the component masses are derived or scaled from existing hardware, except in the areas of technology development discussed earlier. No or minimal redundancy was applied on this system in this context of high risk low cost planetary missions, and 30% contingency was applied on all items except the solar arrays (which are assumed to already include a mass contingency). A 50 kg allocation was made for the rest of the spacecraft, which will be composed of all the subsystems minus the propulsion and power system (an allocation was also made as part of the propulsion system for the spacecraft Power Management and Distribution (PMAD) hardware to reduce the high voltage solar array output to a common 28-V bus). 250-W were also bookkept for running the spacecraft. Tbe xenon tanks mass were assumed to be 10% of the propellant mass, which is consistent with the current NSTAR technology'.

Note that 10-kW BOL array output power is equivalent to about 1.75 kW EOL at 2 AU, 750 W EOL at 3 AU (asteroids, comets), 280 W EOL at 5 AU (Jupiter), assuming 30% lifetime degradation and no benefits from the colder array operating temperatures.

Item	Mass/Unit (kg)	Total m: (kg)
Engine	8	10.4
Gimbals	1.5	2
Gimbal drive electron.	0.4	0.5
PPU	5	6.5
PPU thermal control	3	3.9
PPU micrometeroid protection	0.8	1.0
DCIU	0.5	0.7
Propellant tanks	1 O% propel.	20.4
Fixed feed system	2	2.6
Feed system/engine	1	1.3
PMAD	5	6.5
Subtotal		55.8
Cabling	5% subtotal	2.8
Structure	15% subtotal	8.4
Thermal	5% subtotal	2.8
Solar array (10.25kW)	100 W/kg	102.5
Solar array drive	5	6.5
Residual Xenon		7
Propul. Dry Mass		185.7
Propellant		150.0
Spacecraft		50.0
LV adapter	2.5% wet	9.6
Total S/C Mass		**395.3**

Table 1: Propulsion and spacecraft mass breakdown for 150 kg of propellant (Xenon)

Mission Analysis

The aim of this study was to look at the mission possibilities assuming the existence of the high-power, high performance propulsion system described above. The constraints were to use a small, inexpensive launch vehicle (Pegasus to Taurus class) and to achieve missions in a relatively short trip time (1 year to 2 years to mainbelt asteroids). The missions considered focus on scientifically interesting targets, and include rendezvous with two of the inner mainbelt asteroids, 4-Vesta, with a semi-major axis of 2.36 AU, 1-Ceres, with a semi-major axis of 2.77 AU, and a rendezvous with the short period comet Kopff (semi-major axis of 3.47 AU). A more careful examination of all the mission opportunities to other small bodies or planets was beyond the scope of this work.

The first phase of the study was to assess through a parametric analysis what the propulsion system performances would need to be to achieve a reasonable acceleration (close to or more than 1 mm/s^2) and spacecraft total mass. The second phase focused on finding trajectories and their corresponding trip times with the characteristics resulting from the parametric analysis. These trajectories could either start from a 200-km altitude low Earth orbit (LEO) or from an Earth escape trajectory (at C3 of O km^2/s^2).

Parametric analysis

The aim of this parametric analysis was to get an estimate of the trades space for this problem, and to get some input conditions for the trajectory optimization. The variables were the propellant mass (related to the trajectory performance), the total propulsion system dry mass, the initial trajectory state, the specific impulse (Isp), the system power, and the TAL system efficiency.

Two different starting conditions were considered: 1) starting from LEO, 2) starting from a slightly positive C3. For each of these two conditions, two trajectory performances in terms of final over initial mass ratio (m_f/m_i), were selected (this choice was not random but rather represented boundaries to what could be expected from the real trajectory calculations). The outputs were given in terms of resulting acceleration, and in terms of

C3 = 0 km²/s²

Isp	m_f/m_i = 0.5			m_f/m_i = 0.65		
(s)	M_{min}	Acc	M_{prop}	M_{min}	Acc	M_{prop}
2000	536	1.1	268	'372	1.6	130
2500	536	0.91	268	372	1.3	130
3000	536	0.76	268	372	1.1	130
3500	536	0.65	268	372	0.94	130
4000	536	0.57	268	372	0.82	130
Units	kg	mm/s^2	kg	kg	mm/s^2	kg

Table 2: Parametric analysi results for a C3 of O km^2/s^2

LEO* (200 *km altitude)

Isp	m_f/m_i = 0.5			m_f/m_i = 0.65		
(s)	M_{min}	Acc	M_{prop}	M_{min}	Acc	M_{prop}
2000	973	0.9	635	625	1.4	342
2500	853	0.77	534	559	1.2	287
3000	785	0.66	477	521	1.0	255
3500	741	0.58	440	495	0.87	234
4000	710	0.52	414	477	0.77	218
Units	kg	mm/s^2	kg	kg	mm/s^2	kg

Table 3: Parametric analysis results for LEO

the minimum total wet mass that the spacecraft should have for the mission to be feasible. This minimum mass represents the point of intersection between the curve of total spacecraft wet mass as a function of propellant mass (independently of the mission, computed as in Table 1), and the curve of total spacecraft wet mass as a function of the AV requirements. The following cases assume a SEP power of 10 kW, and a total system efficiency of 0.6 (0.95 for PPU efficiency and 0.63 for thruster efficiency).

These results show that a desirable Isp for both cases is around 2500 seconds since it stays in between reasonable boundaries for the acceleration. The chosen input point for the interplanetary trajectory calculations was therefore: Isp of 2500 sec. SEP power (solar array output power) of 10 kW, total system efficiency of 0.6, and an acceleration of 1 mm/s^2, which lead to an initial mass of 490 kg.

Earth Escape

The Earth escape part of the trajectory was estimated using the Edelbaum equation[12]. This equation gives the low-thrust trajectories characteristic velocity (AV) needed to go from one circular orbit to another. It assumes that the orbits remain quasi-circular for all intermediate changes, and that the thrust angle is held constant during each revolution. For this study, the final circular orbit was assumed to be at infinity (conservative assumption). This approximation lead to a total "Earth escape" AV of about 7.8 km/s. Small changes in inclination (as for the considered mainbelt asteroids) did not affect this result significantly. The mass flow rate for this propulsion system (with the characteristics mentioned above) is about 20 mg/s, leading to a trip time to reach Earth escape of about 77 days, i.e. 2.5 months.

Interplanetary trajectories

Analytical tool

The low-thrust trajectory tool initially used in this study to compute and optimize the trajectories is called DIFINC and is based on the differential inclusion concept. This software was chosen for this work mainly because its ease of use. Once an optimum trajectory (that looked attractive) was found, the results of DIFINC would be used to initialize the more elaborate and complex VARITOP software.

VARITOP is currently the most frequently used program for low thrust trajectory calculation for preliminary mission studies at the Jet Propulsion Laboratory (JPL). It contains several launch vehicle models and for a chosen launch vehicle, optimizes the injected mass as a function of launch energy. It can compute fly-by, rendezvous as well as gravity-assist maneuvers. On a thruster point of view, VARITOP assumes a constant specific impulse and includes several models of Solar Array output power as function of distance from the Sun. VARITOP optimizes flight times and launch dates. It is to be noted that VARITOP takes into account the changes in orbital inclination.

From a trajectory optimization point of view, VARITOP is based on an indirect method algorithm for finding the optimized trajectory (that gives the lowest propellant mass fraction). Indirect methods use the calculus of variations (Lagrange multipliers) to obtain a set of necessary conditions whose solution insures a local extremum of the objective function (i.e. the program finds the absolute minimum over the whole range of varying parameters).

DIFINC, like VARITOP, is a constant Isp low-thrust trajectory program that can compute fly-by and rendezvous types of trajectories and takes into account the changes in inclination. It does not optimize flight times nor launch dates, and is somewhat easier to use and requires less expertise to run than VARITOP. It also includes a model of the solar array power as function of the distance from the Sun.

From a trajectory optimization point of view, DIFINC is based on a direct solving method. Direct methods use gradients to search a parameter space and locate a local extremum. They typically transform optimal control problems into non linear programming problems. These methods exploit finite approximations to the state differential equation and the objective function is directly minimized by varying discrete values for the states and controls. This means that the program typically finds a local minimum for the trajectory instead of a global minimum. More details on these software can be found in literature’ [3].

Vulidation of the analvtical tool

In order to get a feeling for the accuracy provided by the DIFINC analytical tool, we ran several trajectories originally done with VARITOP, keeping the same initial conditions. The VARITOP trajectories were run by Carl G. Sauer from the Low-Thrust Trajectories Group at JPL. The system efficiencies were the same for both software runs. Table 4 summarizes the initial conditions and Fig. 2 shows the results of this comparison.

Mission	Launch	TOF (days)	Power (l AU, kW)	Isp (s)	Ml (kg)
Ceres	1/13/01	913	8	3188	1022
Ceres	10/10/01	1279	3.375	3185	477
Vesta	6/1/01	913	3.375	3185	416
Kopff	5/3/00	1158	3.375	3185	432
Wilson - Harrington	8/30/02	965	10.4	3280	989

Table 4: Summary of the initial conditions for DIFINC and VARITOP (Ml: launch mass)

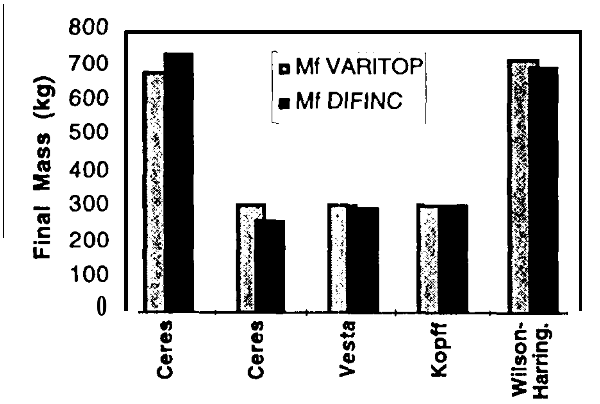

Fig. 2: Differences in final mass given by running both DIFINC and VARITOP

These results show that the results of DIFINC is within about 10% of those from VARITOP, and can be optimistic or pessimistic. To a first approximation, this level of error was considered acceptable for this study and therefore the trajectories were first generated with DIFINC, and then refined and/or corrected with VARITOP.

Results

A comprehensive assessment of the delivery options available for ballistic mainbelt asteroid rendezvous missions has been made at JPL by Chen-wan Yen[14]. A similar examination of the mission options using SEP technology was performed by Carl Sauer[15,16] using VARITOP or its derivatives. Results for asteroids Vesta and Ceres, and comet Kopff are summarized in Tables 5 and 6. The trip times for these missions are typically around 2 to 3. l years with an SEP system and 6 to 7 years in the ballistic case.

	Grav. Assist	Trip Time (days)	Launch Vehicle	S/C net mass (kg)
Vesta	Mars²	-2300	Delta II 7925	230
Ceres	Mars²	-2770	Delta II 7925	160

Table 5: Ballistic mission performances (S/C: spacecraft, Mars²: double Mars Gravity Assist)

	SEP Power	Trip Time (days)	Launch Vehicle s	SIC net mass _ ! & L . -
Vesta	5 kW	~750	Delta 117925	300
	5 kW	~ 910	Delta II 7925	350
Ceres	10 kW	~ 700	Atlas II AS	300
	8 kW	1022	Delta 1[7925	300
Kopff	10 kW	1140	Atlas II AS	350
	5 kW	1130	Delta II 7925	350

Table 6: Mission performances using an NSTAR like SEP system

Since this work foeused on smaller and cheaper spacecraft, a palette of launch vehicles smaller than the Delta II 7925 was considered. Table 7 summarizes their launch capabilities (with a 10% contingency on the delivered mass) and estimated costs. Except for the Pegasus XL, all of the launch vehicles are under development.

	Delivered Mass at LEO (200 km) (kg)	Delivered Mass at C3=0 km ‘/s² (kg)	cost (1999 $M)
Pegasus XL	423		20
Taurus XL	1278		30
‘Taurus XL /Orion 38	1458	121	33
Taurus XL /Star 37	1620	319	35
Taurus XL /Orion 38 /star 37		345	38
LMLV-1	585		20
LMLV-2 /star 37		273	24+5
LMLV-2 /Star 48		387	24 + 8

Table 7: Launch vehicles performances and cost, including a 10% launch vehicle margin

Three example trajectories generated by VARITOP are presented here. They all start with a C3 of O km^2/s^2. The input conditions were a specific impulse of 2500 see, a solar array output power of 10 kW, a total system efficiency of 0.6, a throttling ratio of 5:1, and an initial mass of 490 kg. This resulted in a thrust of 0.49 N and an, initial acceleration of 1 mm/s^2.

The trajectories generated can be scaled to any spacecraft mass as long a the acceleration profile **and the specific** impulse stay the same as the ones used to do the trajectory. To keep the acceleration profile the same, it is sufficient to **keep the ratio of the beam power to the mass the same.** Thus the calculated trajectory performances will be presented and scaling equations will be applied to adjust the performances to the launch vehicle capabilities.

<u>Vesta Rendezvous</u>

Figure 3 shows an optimum trajectory for a Vesta rendezvous launching in January 2007 and arriving 470 days later. The final mass is 313.2 kg (m_f/m_i = 0.64) **and propellant mass 176.8 kg.**

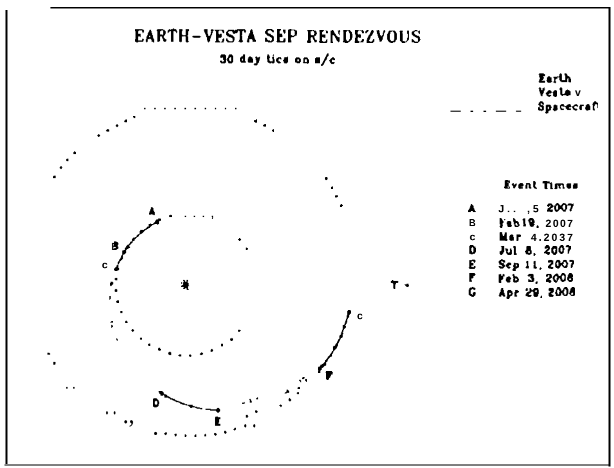

Fig. 3: VARITOP Optimum Vesta rendezvous (10-k W propulsion system)

Scaling **this result to the selected launch** vehicles, one finds that a 50-kg sciencecraft mass could be launched from a Taurus XL/Orion 28/Star 37 and reach Vesta in about 1.3 years, about half the time of an NSTAR-like SEP system. In this case, the total wet spacecraft mass is about 335 kg, with about 124 kg of propellant. Other characteristics and launch options are summarized in Table 8.

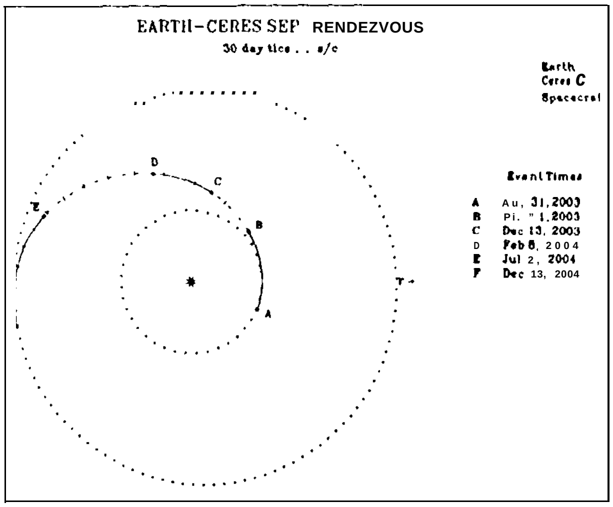

Fig. 4: VARITOP Optimum Ceres rendezvous (10-kW propulsion system)

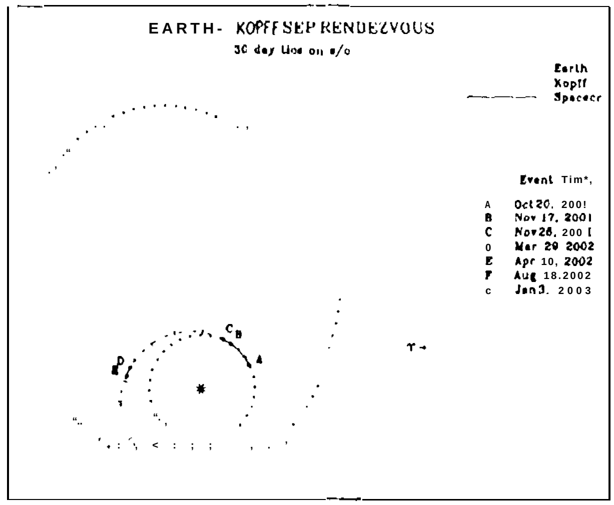

Fig. 5: VARITOP Optimum Kopff rendezvous (10-kW propulsion system)

Ceres Rendezvous

Figure 4 shows an optimum trajectory for a Ceres rendezvous leaving end of August 2003 and arriving 470 days later. The final mass is 256 kg (m_f/m_i = 0.52) and propellant mass 234 kg.

Ceres, being a little bit further from the Sun than Vesta, is a little bit more propellant demanding. The results for this case show that it is possible to transport a 50-kg sciencecraft to rendezvous with Ceres in 1.5 years. The total spacecraft mass is about 840 kg (with 520 kg of propellant) and the launch can be done on a Taurus XL (to LEO). Note that in this case, it is preferable to leave from LEO, and it is also desirable to have a propulsion system more efficient (total efficiency closer to 0.75). Other results are summarized in Table 9.

Kopff Rendezvous

Figure 5 shows an optimum trajectory for a comet Kopff rendezvous starting end of October 2001 and arriving 440 days later. The final mass is 292 kg (m_f/m_i = 0.6) and propellant mass 198 kg.

These results show that a 50-kg sciencecraft can rendezvous with Comet Kopff in 1.2 years if launched from a LMLV-2 to Earth escape, The total spacecraft mass is about 370 kg, with 150 kg of propellant mass. Other launch options for Kopff rendezvous are summarized in Table 10.

Discussion

The results for all three cases are only slightly dependent on the efficiency that the thruster and PPU have for a given power level. The assumption was here that the total efficiency would be 0.6 whatever the power level (unless indicated otherwise), and the results would not change significantly if the system efficiency was different (in the range of 0.6 to 0.7).

The initial operating power level was optimized here to fit within the launch vehicle capabilities, but not on a trajectory point of view. A similar exercise could look at the variation in arrival mass to the body as function of the power level and trip time, i.e. varying the acceleration. The specific impulse could also be optimized. But from these results, a propulsion system that features a power level in-between 7 and 10-kW was sufficient enough to get short trip times and fit the spacecraft on small inexpensive launchers.

Unexpectedly, some missions appear to be attractive starting from an escape trajectory, whereas others can only be done starting from LEO. Here is one more advantage of a high power system that allows LEO to escape trajectories to be performed quickly, dramatically increasing the launch capability of small launchers, and therefore enabling the mission.

A final note is that VARITOP trajectories showed higher performances than the DIFINC trajectories (in terms of final mass) for these short trip time missions. Typically, an increase in final mass from 5 to 12% could be found with VARITOP (this percentage represents the relative mass increase with respect to the VARITOP final mass), which is consistent with earlier results.

VESTA 1.3 Years	Taurus XL Orion+Star	LMLV-2 /Star 48	Pegasus XL	Taurus XL	LMLV-1
Initial state	C3 = 0	C3 = 0	LEO	LEO	LEO
Solar array output power (kW)	7.1	7.9"	6.3	9.0	8.7
Total system efficiency	0.6	0.6	0.6	0.6	0.6
Initial thrust (N)	0.348	0.387	0.309	0.441	0.426
Mass flow rate (mg/s)	14.2	15.8	12.6	18.0	17.4
Total propellant mass (kg)	123.8	139.3	226.1	32.0.7	312.7
Total spacecraft wet mass (kg)	334.8	361.1	448.1	588.6	57S.8
Launch vehicle capability (kg)	345	387	423	1278	585
Margin to LV capability (kg)	10.2	25.9	- 25.1	50,4	9.2
Number of S/C per launch	"1"	1	"	2	1

Table 8: 1.3 years Vesta rendezvous spacecraft options as function of launch vehicle

CERES 1.3 years	Taurus XL Orion+Star	LMLV-2 /Star 48	Taurus XL
Initial state	C3=0	C3=0	LEO
Solar array output power (kW)	10.0	10.0	10.0
Total system efficiency	0.62	0.62	0.75
Initial thrust (N)	0.506	0.506	0.612
Mass flow rate (mg/s)	20.6	20.6	25.0
Total propellant mass (kg)	242.9	242.9	519.8
Total spacecraft wet mass (kg)	506	506	836
Launch vehicle capability (kg)	345	387	1278
Margin (kg)	- *161*	-'719	442
Number of S/C per launch			1

Table 9: 1.3 years Ceres rendezvous spacecraft options as function of launch vehicle

KOPFF 1.2-years	Taurus XL Orion+Star	LMLV-2 /Star 48	Taurus XL Orion 38
Initial state	C3 = 0	C3=0	LEO
Solar array output power (kW)	7.6	7.6	9.9
Total system efficiency	0.6	0.6	0,6
Initial thrust (N)	0.372	0.372	0.485
Mass flow rate (g/s)	15.2	15.2	19.8
Total propellant mass (kg)	149.9	149.9	376.2
Total spacecraft wet mass (kg)	371	371	664
Launch vehicle capability (kg)	345	387	1458
Margin (kg)	- 26	16	65
Number of S/C per launch		1	2

Table 10: 1.2 years Kopff rendezvous spacecraft options as function of launch vehicle

Conclusion

The aim of this study was to assess if high power SEP systems could be used to deliver small spacecraft to scientifically interesting places in the Solar System from small launch vehicles. The high power propulsion system assumed featured a 7- to 10-kW Thruster with Anode Layer (TAL), that would be driven directly off a high voltage light-weight solar array. The TAL thruster would be similar to the existing D-l Of) developed by TsNIIMASH, and would feature efficiencies in between 0.6 and 0.7 at specific impulses around 2500 seconds. The high voltage solar array assumed here would need to be developed and its specific mass would need to reach the 100 W/kg. Specific masses of 70-80 W/kg can be expected with current concentrator array technology. The power processing unit design **would be simplified** thanks to the direct drive configuration, and significant mass reduction would result.

Mission analysis of this concept focused on two mainbelt asteroids (Vesta and Ceres) and a short period comet (Kopff). Results show that a 50-kg sciencecraft could be carried to asteroids or comets in about 1.2 to 1.5 years and be launched from a Taurus class vehicle. In some cases, and depending on the trajectory performances, the mission was enabled by a launch to LEO, and in other cases it was attractive to be launched on an escape trajectory. Launch masses varied depending on the mission. The asteroid Vesta and comet Kopff rendezvous featured launch masses to Earth escape around 350 kg (xenon propellant masses around 120-150 kg). The asteroid Ceres rendezvous called for a spacecraft mass of about 840 kg (520 kg of propellant) and a launch to LEO. Other destinations could be looked at in future work along with additional optimization of the power level and specific impulse per mission.

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