Journal of Spacecraft and Rockets
Vol. 43, No. 5, September–October 2006

Solar Electric and Chemical Propulsion for a Titan Mission

Michael L. Cupples* and Shaun E. Green†
Science Applications International Corporation, Huntsville, Alabama 35806
Victoria L. Coverstone‡
University of Illinois at Urbana–Champaign, Urbana, Illinois 61801
and
Benjamin B. Donahue§
The Boeing Company, Huntsville, Alabama 35806

Systems analyses were performed for a Titan Explorer Mission characterized by Earth–Saturn transfer stages using solar electric power generation and propulsion systems for primary interplanetary propulsion, as well as chemical propulsion for capture at Titan. An examination of a range of system factors was performed to determine their effect on the payload delivery capability to Titan. The effect of varying launch vehicle type, solar array power level, ion thruster number, specific impulse, trip time, and Titan capture stage chemical propellant choice was investigated. The major purpose was to demonstrate the efficacy of applying advanced ion propulsion system technologies like NASA's evolutionary xenon thruster (NEXT), coupled with state-of-the-art (SOA) and advanced chemical technologies to a Flagship class mission. This study demonstrated that a NASA design reference mission (DRM) payload of 406 kg could be successfully delivered to Titan using the baseline advanced ion propulsion system in conjunction with SOA chemical propulsion for Titan capture. In addition, the solar electric propulsion system (SEPS)/chemical system of this study is compared to an all-chemical NASA DRM. Results showed that the NEXT-based SEPS/chemical system was able to deliver the required payload to Titan in 5 years less transfer time and on a smaller launch vehicle than the SOA chemical option.

Introduction

THE Titan Explorer Mission has generated significant interest in the space science community and has been analyzed using various propulsion systems.[1] Previous analyses covered a wide range of propulsion systems, but did not specifcally investigate the application of an advanced ion propulsion systems like NASA's Evolutionary Xenon Thruster (NEXT) to the Titan Explorer Misson. This study examines propulsion performance for the Titan explorer mission for a NEXT-based solar electric propulsion system (SEPS) stage combined with state-of-the-art (SOA) and advanced chemical propulsion options. The chemical propulsion system (CPS) is integrated into a full-up spacecraft and is used for Titan capture. At Saturn distances the solar flux is too small to provide adequate solar power to the ion thrusters for propulsion and, hence, neccessitates an alternative Titan capture propulsion mode.

This study will, in general, investigate the sensitivities of the SEPS delivered mass and Titan payload to various system and mission parameters. The SEPS delivered mass is defined as the CPS spacecraft and the total science payload (Titan payload). The Titan payload consists of a direct entry Titan lander and science instruments that remain in Titan orbit. The science instruments that remain in orbit at Titan are referred to in this study as the orbiter science instruments. The first part of this study investigates SEPS delivered mass sensitivity to launch vehicle, opportunity year, transfer time, number of operational ion engines, and ion engine maximum I_{sp}. The second part of the analysis examines Titan payload sensitivity to propellant combination, tank technology, and transfer time.

In this study, launches from both Atlas-V and Delta-IV medium class launch vehicles were evaluated. The launch vehicle injects the SEPS interplanetary transfer stage and CPS spacecraft, along with the Titan payload, on a minimum-propellant optimized trajectory to Saturn with eventual capture of the CPS spacecraft into a circular 2000-km Titan orbit. As a typical example, an Atlas-V 431 launch vehicle injected a total mass of about 4333 kg to an Earth departure specific energy (C3) of 12.3 km^2/s^2. After launch vehicle injection, the SEPS stage is used to propel the CPS spacecraft and Titan payload through a trajectory that includes a Venus gravity assist (VGA) en route to Saturn. The SEPS stage is separated from the orbiter at approximately 4 astronomical units (AU) after completing its task. A larger gain in velocity could most likely have been realized with an Earth gravity assist (EGA) rather than with a VGA, but the VGA was chosen because of concerns of performing an EGA with a spacecaft that includes a plutonium-based radioisotope thermal generator (RTG) power source. (Two RTGs are baselined in this study for the chemical capture stage.) On arrival in the Saturn vicinity, the CPS-based spacecraft releases the the Titan lander (with aeroshell) onto a Titan direct entry trajectory to Titan, and then the main insertion burn of the CPS places the CPS spacecraft into Titan orbit. Total baseline payload mass that must be delivered to Titan, as taken from the NASA's design reference mission,[2] is 406 kg. This payload consists of 42 kg of instruments (orbiter science instruments mass) on the CPS stage in Titan orbit and a separate 346-kg direct entry lander.

Several propellant combinations for the CPS were considered. These included SOA nitrogen-tetroxide/hydrazine (NTO/N_2H_4), liquid-oxygen/hydrazine (LOX/N_2H_4), fluorine/hydrazine (F_2/N_2H_4), and a high I_{sp} monopropellant[3] system. F_2 was analyzed for its high performance potential, and the advanced monopropellant was analyzed for its potential of global propulsion system simplicity. For the Titan capture stage, two pressure-fed engines of 100-psia ($\sim 6.9 \times 10^5$-N/m^2) chamber pressure P_c and 100-lbf (445-N) thrust were assumed throughout the analyses. Advanced CPS tank technologies considered in this study included tank liner thickness and composite tank overwrap strength.

Presented as Paper 2003-4728 at the AIAA/ASME/SAE/ASEE 39th Joint Propulsion Conference, Huntsville, AL, 20–23 July 2003; received 20 October 2003; revision received 5 October 2004; accepted for publication 8 October 2004.

*Lead Systems Engineer, Engineering and Technologies Section.
†Systems Engineer, Engineering and Technologies Section.
‡Associate Professor, Department of Aerospace Engineering. Associate Fellow AIAA.
§Third-Generation Space Technology Task Lead, Boeing Phantom Works.

Table 1 Baseline SEPS stage power and propulsion systems definition

System	Definition
Power	30 kWe at 1-AU end-of-life (EOL) arrays, 25-kWe maximum into ion propulsion system (IPS); housekeeping power is assumed to be covered by five additional kWe array reserves
Array	Multijunction GaAs arrays, AEC-Able Engineering, Inc. (AEC), ultraflex design, arrays feathered at distances <1 AU
Thrusters	four thrusters with one spare, ~6 kWe at 4000-s I_{sp}; high I_{sp} throttling, NEXT design; molybdenum grids
PPU	four PPUs with one spare; cross-strapping PPUs; ~6.25-kWe maximum power, NEXT design
Radiators	SOA heat pipe radiators
DCIU	SOA DCIU
Tank and propellant	Tank fraction = 5% (SOA), supercritical Xe propellant
Propellant management	NEXT design

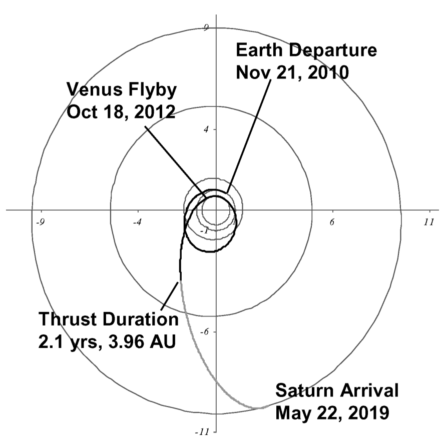

Fig. 1 Typical Earth–Venus–Saturn trajectory.

Systems Analysis

Launch Vehicle Models

Launch mass vs launch C3 performance data taken from the vendor's literature[4,5] for the Delta-IV 4450, Atlas-V 421, and Atlas-V 431 launch vehicles were used in performing trajectory optimization. Appropriate curve fits of the launch vehicle data were used directly in the trajectory optimization code to provide the optimum mass and C3 required to perform the mission based on other system and mission assumptions, including destination and transfer time.

SEPS Models

SEP vehicle synthesis models were used to provide estimates of the spacecraft mass. Table 1 shows the baseline power and propulsion system assumptions used for analyses. Figure 1 shows the main SEPS and subsystem elements modeled. After computing the mass of the electric power system (power generation, conditioning, and distribution), propulsion system [power processing units (PPU), thrusters, gimbals, actuators, digital control interface units (DCIU), and cables/harness], propellant management (fluid management and tank thermal conditioning), and structures (bus, adaptors, mechanisms, thruster support, PPU support, tank support, and other component attachment), the remaining mass allocation represents the usable mass that can be delivered to the destination. The mass referred to herein as the SEPS delivered mass is the difference between the launch vehicle interplanetary injection mass and the wet mass of the SEPS stage. Determining the SEPS vehicle's wet mass was a primary task of this study, and the discussion that immediately follows focuses on the SEPS primary power and propulsion systems.

SEPS Power

High-efficiency solar photovoltaic arrays provide power for propulsion and vehicle housekeeping (with the exception of battery power that must be provided for array deployment). The array model was based on high-efficiency triple-junction galium arsenite (GaAs) cells with an ultraflex[6] structure; the initial power selected was 31.5 kWe, representing the 30-kWe baseline plus a 5% margin. A 2% per year degradation factor was applied to all solar array masses to account for expected radiation degredation during the mission, including increased degradation due to less than 1-AU operation of the propulsion system during certain peroids of time before and after the VGA. The ultraflex model used provides a current SOA in lightweight solar array technology. Low-intensity low-temperature[7] effects were also accounted for in the power generation model.

SEPS Propulsion

The electric propulsion subsystem models include the following: PPU, thruster, gimbals and gimbal actuators, DCIU, propellant management system, cable and harness, and support structure. For all cases evaluated, including variations on thruster number, a spare propulsion string consisting of a thruster, PPU, and propellant management system is included in the SEPS. The baseline SEPS configuration used an array of five NEXT[8,9] ion thrusters (four active and one spare) operating in the high I_{sp} throttling mode. The NEXT-class advanced ion engines operate at a maximum power level of about 6 kWe. Two cases of maxiumum I_{sp} were investigated, 3600 and 4000 s, to determine the SEPS stage sensitivity to thruster I_{sp} level. The NASA Solar Electric Propulsion Technology Application Readiness (NSTAR) ion thruster has a demonstrated maximum thruster power level of about 2.3 kWe at a maximum I_{sp} of 3120 s.

Other SEP Systems

Other SEPS vehicle subsystem models play a critical role in determining the overall mass of the spacecraft and, therefore, affect delivered mass to the destination. All subsystems accounted for in the SEPS model are shown in Fig. 2.

For each subsystem block indicated in Fig. 2, the vehicle synthesis code computes the mass. These masses are then deducted from the total launch vehicle injected mass to obtain the final SEPS delivered mass to the Saturn vacinity. SEPS analyses required various assumptions concerning mass and power margins, contingencies, and vehicle system redundancies. These assumptions are listed in Table 2.

Titan Capture Stage CPS Models

Planetary capture CPS models were derived from curve fits of historical data and physics-based models. For example, the composite overwrap tank model is scaled from the Advanced X-ray Astrophysics Facility[10] vehicle's composite tank. The baseline CPS

Table 2 SEPS contingencies, margins, and other assumptions

System	Contingencies
Launch vehicle	2% nominal capacity
Xe propellant reserve, residual, navigation and trajectory corrections	10% of deterministic propellant
Chemical capture stage ΔV	2.5% g-loss; out-of-tangency Titan rendezvous 2%
Array EOL contingency	14% of baseline power mass
SEPS propulsion duty cycle	95% of total time SEPS is active
	Margins
Dry mass	30% of nondelivered dry mass
SEPS power	5% of baseline power dry mass
	Other
SEPS redundancy	One extra thruster, PPU, gimbal, and DCIU
Attitude control system	Provided by IPS during low-thrust operation, by RCS during ion engine off (scheduled operations coasts)

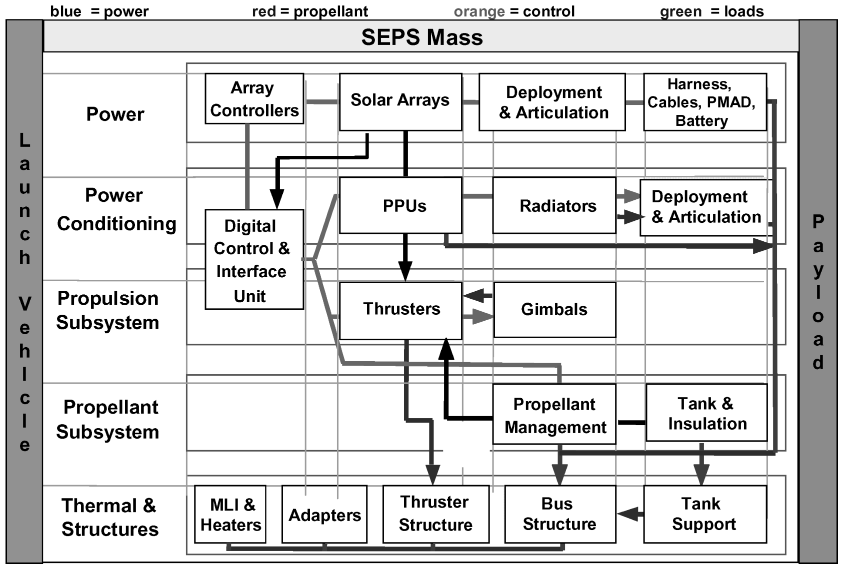

Fig. 2 Total SEPS definition model diagram.

consists of two 100-lbf (~445-N) class main pressure-fed engines. The thrust level was chosen to keep the finite burn gravity losses to a low percent of total characteristic (ideal) ΔV during the Titan capture burn. Other elements include the thrust vector control system, propellant thermal conditioning and pressurization systems, and the reaction control systems (RCS). The RCS consists of 16 hydrazine monopropellant thrusters that operate at an I_{sp} of 220 s. Lightweight tanks using internal tank pressures on the order of 230 psia are baselined. Thermal conditioning is provided for tanks, lines, valves, and thrusters. The propellant pressurization system assumes 4500-psia (3.1×10^8-N/m^2) regulated gaseous helium. Given this relatively high-pressure assumption, the pressurant system mass is based on real gas[11] considerations, including helium molecule finite size.

Mission Analysis

Optimization of SEPS Trajectories

The Solar Electric Propulsion Trajectory Optimization Program[12,13] (SEPTOP) was used to generate optimal trajectories to Saturn for all cases studied. The trajectory optimization process includes, as an optimization constraint, launch vehicle injection mass capability as a function of C3. SEPTOP was used to generate the interplanetary trajectories for a variety of relevant launch dates, trip times, departure C3, arrival velocities, power levels, thruster combinations, and thruster I_{sp}. SEPS stage propellant, change in velocity ΔV, thruster operation time, and thruster throttling and sequencing data, among other data, are also generated as outputs of SEPTOP. Specific thruster models are imbedded into the propulsion system modeling routines, and this allowed detailed sensitivity analyses to be easily performed. Constraints, such as the maximum allowed power output from the solar arrays (done through feathering) and the maximum and minimum thruster operational power levels, can be placed on major system elements.

Earth–Venus–Saturn Trajectories

Optimal gravity assists for the outer planet mission investigated tend to be in a class of trajectories termed energy pumping. This term implies that the vehicle expends time in the inner solar system building energy before the VGA occurs. The optimal energy gain occurs as the vehicle is directed by the SEPS stage into a transfer path that takes it into solar distances between approximately 0.7 and 2 AU. Most of the vehicles heliocentric velocity increase is gained in this high solar flux region. After building orbital energy, the spacecraft performs a VGA to acquire the additional energy needed to reach Saturn within the prescribed transfer time. An example energy pumping trajectory is shown in Fig. 1, illustrating a typical 8.5-year trajectory[14] used in this study.

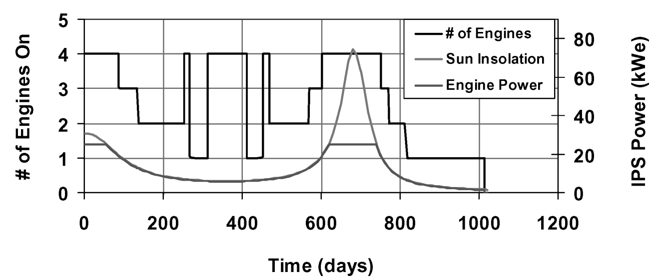

Fig. 3 Number of ion thrusters and power as function of time.

The energy pumping maneuver can be further described in the following sequence of events: 1) The spacecraft increases its potential energy by looping away from the sun. 2) The spacecraft then moves toward the sun (increasing its kinetic energy), achieving VGA (near heliocentric perihelion) and providing a relatively large net heliocentric velocity increase. For the cases studied, the flyby radius at Venus was constrained to no less than 6352 km, and the typical velocity increase during the VGA ranges from 4.5 to 5.0 km/s. Sample SEPS trajectory data showing thruster power-on-time, engine power, and number of engines on as a function of mission time are shown in Fig. 3.

As indicted in Fig. 3, IPS power is held to a maximum of 25 kWe after SEPS separation from the launch vehicle and array deployment. As the spacecraft moves away from the sun (0–350 days), the number of operational thrusters is reduced as array power decreases, until a single thruster is operating. As the spacecraft returns to a lower AU, the number of thrusters increases back to the maximum (four) at 253, 300, and 460 days, as shown in Figs. 3 and 4.

For example, the total power into the thrusters at 253 days was 6.8 kWe. Although two thrusters would be sufficient to handle this power level, SEPTOP selected four active thrusters, assuring an optimal trajectory based on the global set of power, propulsion, and constraints. Essentially, at this point in the trajectory and at the related power level, higher thrust provided by four thrusters operating at lower power was more important than the higher efficiency operation of two thrusters operating at higher power. The longest duration of four active thrusters occurs from about 600 to 750 days (during which period the spacecraft will reach its closest distance from the sun) and the VGA occurs at 700 days. For IPS power limiting and array thermal control purposes, solar array feathering is required during low AU phases of the mission. Figure 3 shows that IPS power is held at 25 kWe during the feathered periods, 0–60 and 620–740 days. The SEPS stage is jettisoned, and CPS coasting

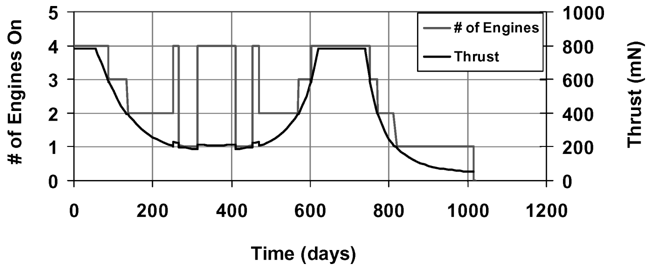

Fig. 4 Number of ion thrusters and thrust as function of time.

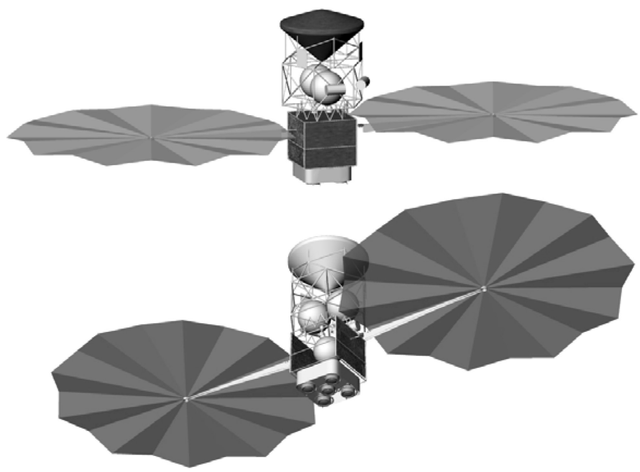

Fig. 5 Titan Explorer mission propulsion system concept.

begins at about 4 AU (1015 days, 2.8 years) as shown in Figs. 1, 3, and 4. At this time, the CPS with the total Titan payload begins an additional 2088-day (5.7-year) coast before the Titan chemical propulsion capture maneuver. During this coast, power is provided by the RTG system.

A representation of the full Titan Explorer spacecraft concept is shown in Fig. 5. The SEPS stage is shown on the bottom of the stack with its dual 15-kWe solar arrays deployed. Five ion thrusters, enclosed on their perimeter by a sun shield, are shown at the bottom of the craft. On top of the SEPS stage is a cut-away view of the CPS spacecraft showing the position of its two main propellant tanks. On the top of the CPS spacecraft, the conical-shaped Titan lander module is shown.

Titan Capture Scenario

Table 3 summarizes a typical ΔV budget for a CPS Titan stage of the type studied. These data are for a system using an Atlas-V 431 launch vehicle, a 30-kWe SEPS stage, and the baseline NTO/N_2H_4CPS. Aimpoint[15] approach calculations indicated a 3455-m/s capture ΔV. The calculation includes Saturn and Titan orbital geometry considerations, Titan approach uncertainty contingency, gravity losses (g losses), and navigation/midcourse ΔV correction. An assumed contingency for an out-of-tangency Titan rendezvous of 2% of ideal ΔV (66 m/s) was included. The capture burn maneuver includes approximately 8 deg of plane change. In addition to the aforementioned ΔV margins, the CPS carries an additional 5.6% for propellant reserve, residuals, and margin.

Gravity Loss Analysis for Titan Capture

The Titan approach was modeled, and an integrated trajectory was generated to estimate the actual finite burn losses (g losses) at capture. A g-loss calculation example is shown in Fig. 6. Calculations were based on a total mass of CPS plus orbiter science instruments and a Titan capture burn with the two aforementioned pressure-fed engines. The required burn time to capture was 4210 s; the calculated ΔV loss was 72 m/s, or 2.1% greater than the ideal ΔV. Therefore, to be conservative, all CPS main propellant requirements assumed g losses of 2.5% of ideal ΔV.

Table 3 Typical Titan capture ΔV budget

Parameter	Value
Titan gravitational parameter	9.027×10^3 km^3/s^2
Titan radius	2575 km
Circular capture orbit altitude	2000 km
Excess velocity (V_{hp}) relative to Titan	4250 m/s
Hyperbolic velocity at Titan periapsis	4691 m/s
Circular velocity at Titan capture orbit	1405 m/s
Characteristic capture ΔV	3287 m/s
Contingency: orbit insertion uncertainty	66 m/s
Finite burn losses (g losses)	82 m/s
Navigation and trajectory correction ΔV	20 m/s
Total ΔV (propulsive capture)	3455 m/s

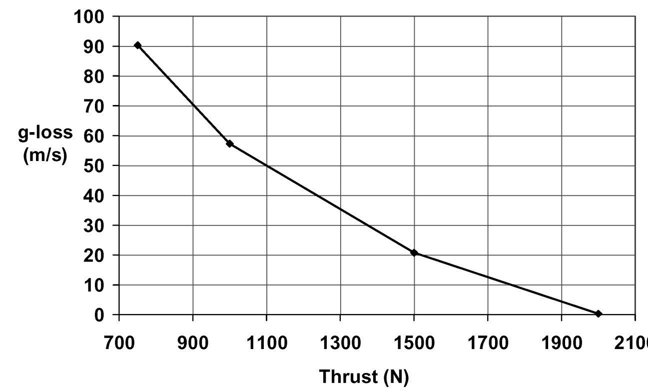

Fig. 6 Finite burn g loss at Titan capture vs thrust magnitude.

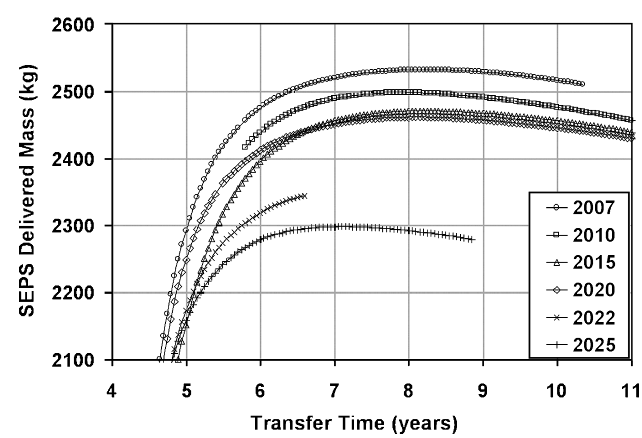

Fig. 7 Launch opportunity comparison.

SEPS Mass Delivery Sensitivity Analyses Results

SEPS Delivered Mass Sensitivity to Launch Date and Launch Vehicle

A launch opportunity analysis was conducted for Earth departure dates of 2007–2025. Figure 7 shows SEPS-delivered mass to Saturn, including the mass of the inert SEPS stage for a number of cases. Both the Atlas-V 421 and 431 and the Delta-IV 4450 medium-class launch vehicles generally provide greater than 2400 kg of usable mass over transfer times from 6 to 9 years and for the most favorable launch opportunities 2007, 2010, and 2015. SEPS delivered mass capability generally decreases as the Earth-departure year increases over the range examined. The percent difference in the delivered mass over this range is about 8%. The 2007 mission provided the largest delivered mass, 2540 kg, and the 2010 opportunity provided the next largest mass, 2500 kg (both at approximately 8-year trip times).

All results shown hereafter will be from the 2010 launch case, which is considered a representative opportunity. For this 2010

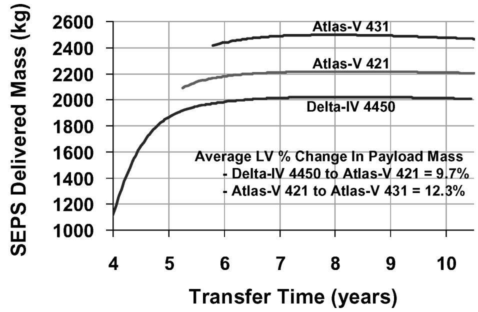

Fig. 8 Variation in SEPS delivered mass with launch vehicle.

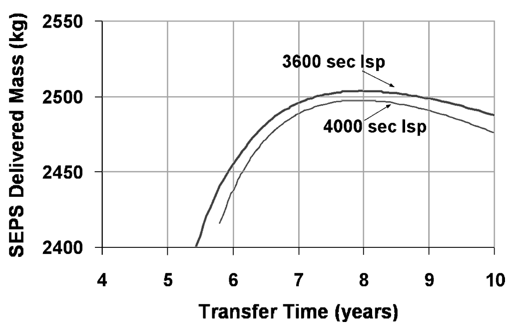

Fig. 9 Impact of thruster I_{sp} on SEPS delivered mass.

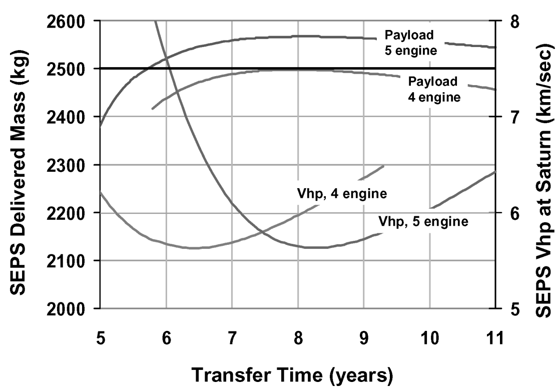

Fig. 10 SEPS delivered mass sensitivity to number of operational thrusters.

opportunity, an analysis was conducted to determine the sensitivity of SEPS delivered mass for three launch vehicle selections. The resulting SEPS delivered masses are shown in Fig. 8. As shown in Fig. 8, the percent delivered mass increase between the Delta-IV 4450 and Atlas-V 421 vehicles was approximately 10%. The increase provided by the Atlas-V 431 over the Atlas-V 421 was about 12%.

SEPS Delivered Mass Sensitivity to Electric Propulsion I_{sp}

A thruster I_{sp} sensitivity study was performed to determine the variation in SEPS delivered mass with I_{sp}. As shown in Fig. 9, for the power levels investigated, and using the Atlas-V 431 launch vehicle, only small differences in SEPS delivered mass were found for thruster I_{sp} values of 3600 and 4000 s. The 4000-s thruster was chosen for the remaining analyses in this paper due to this demonstrated thruster insensitivity to I_{sp} and because greater thruster lifetimes will, in general, be technically easier to achieve for the higher I_{sp} thruster.

SEPS Delivered Mass Sensitivity to Total Number of Active Thrusters

Further analyses were performed to determine whether an increase in the number of SEPS thrusters along with associated increase in array power would dramatically increase the SEPS delivered mass to Saturn vicinity. Results, shown in Fig. 10, indicated that, by increasing the number of operational thrusters from four to five, the SEPS delivered mass to Saturn vicinity increased by about 3%. Note, however, that in going from four to five operational thrusters, there will likely be a significantly higher cost associated with increased power and one additional thruster, PPU, and propellant management string, whereas the actual increase in Titan payload will be relatively small.

Figure 10 also indicates the reduction in trip time facilitated by five thrusters for an example SEPS delivered mass of 2500 kg. (Note that this example SEPS delivered mass represents a conservative estimate of the required mass for the CPS spacecraft and total Titan payload.) The approximately 8-year trip time that is required to deliver the 2500 kg to the Saturn vicinity for the four-thruster SEPS can be reduced to about 5.7 years by the five-thruster system. Note that both Saturn and Titan arrival V_{hp} are also shown in Fig. 10. In this study, Saturn V_{hp} defines the relative velocity (sometime called the excess velocity) of the CPS spacecraft in the heliocentric coordinate system with respect to Saturn. Likewise, Titan V_{hp} defines the relative velocity in Saturn-centered coordinates of the CPS spacecraft with respect to Titan. As can be gleaned from Fig. 10, the V_{hp} at Saturn arrival increases dramatically as the transfer time falls below the 7.5-year transfer time. This sharp increase in Saturn V_{hp} with decrease in transfer time, coupled with a Titan payload that falls off rapidly with increasing Saturn V_{hp}, implies that the SEPS delivered mass benefit with five thrusters is quickly lost with decreasing transfer time. Note that Titan payload sensitivity with Saturn and Titan V_{hp} will be more thoroughly covered in an upcoming section that addresses Titan payload sensitivities.

For completeness, a three-operational-thruster SEPS configuration was investigated as well. This configuration was marginally successful in placing the reference payload at Titan. The average thruster throughput for the three-engine case, however, was significantly higher than the baseline NEXT program design value of 270 kg per thruster.

Titan Payload Delivery Sensitivity Analyses Results

Titan Payload Sensitivity to Launch Vehicle and Transfer Time

A shift is now made from considering SEPS delivered mass to considering actual Titan payload. Titan payload has been defined as the direct entry lander and the previously mentioned orbiter science instruments. The transfer vehicle is the earlier baselined combined four-thruster SEPS (4000 s I_{sp}) stage and the NTO/N_2H_4-based CPS spacecraft. The SEPS stage provides the primary interplanetary propulsion that sends the CPS with total science payload to the Saturn vicinity. As the CPS spacecraft arrives in Saturn vicinity, the direct entry lander is seprated from the spacecraft and proceeds to Titan for direct entry and landing. Subsequent to lander release, the remaining CPS-based spacecraft with the orbiter science instruments proceeds to Titan for a chemical-based capture maneuver. The total science payload (Titan payload) delivered to Titan as a function of transfer time, launch vehicle type, and arrival V_{hp} is shown in Fig. 11. Figure 11 shows that Titan payload maxima are reached at approximately 8.5 years. The NASA reference Titan payload value[2] is shown in Fig. 11 as a bold, horizontal line. Actual estimated payload delivery capability is shown in Fig. 11 as a series of curves, one curve for each of the three different launch vehicles. The baseline

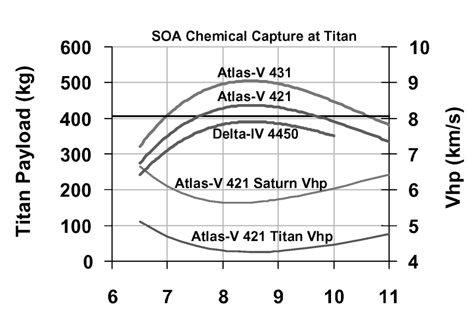

Fig. 11 Titan payload sensitivity to transfer time and launch vehicle.

transfer vehicle can deliver the design reference mission (DRM) payload to Titan in 7.0 years if launched by an Atlas-V 431 and in 7.5 years if launched by an Atlas-V 421. Titan payload delivered with the Delta-IV 4450 (the smallest launcher considered) in 8.5 years is 390 kg, just under the 406-kg DRM payload. At 8.5 years, the Atlas-V 431 and 421 vehicles delivered 505- and 437-kg payloads, respectively. The maximization of Titan payload at 8.5 years can be understood by noting that the minimum Titan V_{hp} is reached at 8.5 years. Minimum V_{hp} at Titan translates into minimum propulsive ΔV for the CPS capture and, thus, maximum Titan payload.

Orbiter Science Mass Sensitivity to Chemical Propellant Combinations

A detailed deliniation of subsystem mass is provided in this section to clarify the relationship that orbiter science instruments mass has to chemical propellant combination. Three potential CPS systems were considered, and Table 4 shows the corresponding SEPS stage and CPS spacecraft mass data for the three chemical propellant combinations. For this study, the baseline assumptions include the Atlas-V 431 launch vehcile, a 8.5-year trip time, and the baseline four SEPS configuration. The SOA NTO/N_2H_4 case was shown in Fig. 11. For each of the propellant combinations shown in Table 4, the SEPS stages are identical, with the dry mass of 848 kg and a total propellant load of 898 kg. The total SEPS stage mass is 1839 kg. For a total solar array power level of 30-kWe EOL, the SEPS stage specific mass is nearly 61 kg/kWe.

The total ΔV for the Titan capture (including all losses) was estimated at 3455 m/s, and the chemical stage I_{sp} used in this analysis were 330, 351, and 380 s for the SOA chemical, LOX/N_2H_4 system, and F2/N_2H_4 system, respectively. The corresponding dry mass for the chemical systems are 496, 533, and 515 kg, respectively. The SOA CPS has the lowest dry mass due to the compact nature of the baseline Earth-storable propellant system. The space-storable propellant combinations, LOX/N_2H_4, and F2/N_2H_4, required greater dry mass systems due in part to increased subsystem masses for the cryocoolers and added power to operate the cryocooler. Both of the space-storable systems provide an orbiter science instruments delivery advantage over the SOA CPS. The orbiter science instruments increase was 12 kg for LOX/N_2H_4 and 99 kg for F2/N_2H_4. These orbiter paylaod increases over SOA chemical can be attributed to the larger I_{sp} that the space-storable propulsion systems provide.

The relatively small 12-kg orbiter science instruments mass increase over SOA for the LOX/N_2H_4 is considered a marginal increase in total payload delivery capability. A relatively high Titan payload of approximately 600 kg can be achieved with the F_2 propellant[16,17] on an Atlas-V 431 launch. This turns out to be an almost 100-kg greater orbiter science instruments mass than the SOA chemical system can deliverer to Titan orbit. In addition, further analyses for F_2 propulsion showed that the Atlas-V 421 and the Delta-IV 4450 deliver 520- and 460-kg Titan payloads, repectively. Thus, with F_2 propulsion, the DRM payload can be successfully delivered to Titan for the smallest launch vehicle case investigated. F_2 engines were tested successfully by NASA and the U.S. Air Force during engine development programs of the 1960s and 1970s; however, work on F_2 was not done in the past decade. Thus, an appreciable technology development program would be necessary to raise F_2 propulsion technology to an appropriate technology readiness level before to in-space application.

In addition to the aforementioned propellant combinations, this study investigated advanced monopropellants to determine the efficacy of an advanced monopropellant propulsion system application for this mission class. Analysis indicated that a monopropellant CPS capture stage operating at 275-s I_{sp} combined with the baseline SEPS stage could not achieve the 406-kg reference payload for any of the three launch vehicles investigated. The analysis further showed that an I_{sp} of 320 with an Atlas-V 431 launch vehicle would be required for a monopropellant system to deliver the DRM Titan payload.

Orbiter Science Mass Sensitivity to Tank Technology Improvements

A study was performed to determine the potential Titan payload increase that could be realized from CPS main propellant tank technology improvements. Analyses were performed to determine Titan payload sensitivity to propellant tank liner thickness and composite overwrap thickness. A modest increase in Titan payload of about 15 kg was found by decreasing tank liner thickness from 30 to 5 mil. Recent manufacturing improvements in tank liner technology have achieved this sixfold decrease in liner thickness. Reducing composite tank overwrap weight was also evaluated. An increase in tank composite overwrap strength by 30% resulted in an increase of approximately 3% in Titan orbiter science instruments mass. Tank overwrap strength technology improvements of 30% over SOA are within the reach of current tank manufacturing processes.

SEPS/Chemical Transfer System Comparison to an All-Propulsive NASA DRM

Mission analyses results from an earlier and separate NASA inhouse Titan all-chemical reference mission is included in this section for comparison purposes. Figure 12 shows three columns of data; the first column on the left of Fig. 12 shows total stack mass data for the baseline SEPS/chemical system, the middle column shows the total stack mass data for the F_2-based SEPS/chemical system, and the column on the right shows the total stack mass for the NASA all-chemical DRM. In this NASA DRM, no SEPS stage was used, and a Delta-IV heavy launch vehicle was required to inject the DRM payload (406 kg) into a 12-year Venus–Earth–Earth Gravity Assist (VEEGA) transfer trajectory. Aside from several midcourse

Table 4 Total mass statement for chemical propellant combination[a]

Parameter	Capture stage bipropellant NTO/N_2H_4	O_2/N_2H_4	F_2/N_2H_4
I_{sp}, s	330	351	380
Launch vehicle adapter, kg	54	54	54
SEPS transfer stage ($\Delta V = 8018$ m/s)			
Xe propellant mass, kg	821	821	821
Xe propellant reserves, residuals, kg	82	82	82
RCS propellant, N_2H_4, kg	5	5	5
Dry mass, kg	848	848	848
Orbiter adapter, kg	29	29	29
CPS capture stage ($\Delta V = 3455$ m/s)			
Main propellant mass, kg	1331	1286	1224
Main propellant reserves, residuals, kg	149	145	138
RCS propellant, N_2H_4, kg	13	13	13
Dry mass, kg	496	533	515
Orbiter science instruments mass, kg	141	153	240
Lander (Titan direct entry, not captured), kg	364	364	364
Total initial mass (launch C3 = 12.3 km^2/s^2), kg	4333	4333	4333

[a]All cases are for an 8.5-year transfer time and an Atlas-V 431 launch vehicle.

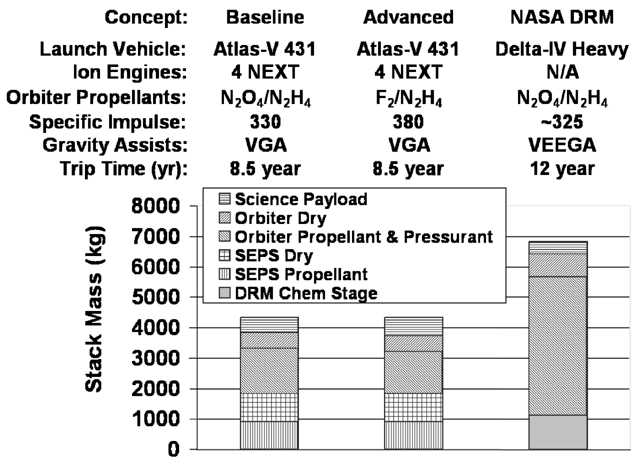

Fig. 12 SEPS transfer system total mass compared to DRM.

correction burns, the all-propulsive Titan CPS coasts for the duration of the mission, gaining much of the required momentum to reach Saturn via the three gravity assists. Because of the VEEGA trajectory chosen, a relatively high arrivial velocity is encountered at Saturn for this DRM as compared to the SEPS mission analysed herein, with its single VGA and 1000+ day ion thrusting duration. This study's SEPS trajectory provides a more moderate Saturn arrival velocity, even though its trip time of 8.5 years is significantly less. The higher Saturn arrival velocity of the VEEGA trajectory results in a significantly higher Titan CPS capture stage propellant load as compared to that needed for the CPS of the SEPS/VGA mission baselined in this study. In Fig. 12, total stack mass is listed for two SEP/VGA missions (columns 1 and 2), described earlier in Table 4, in addition to the aforementioned all-propulsive DRM. The primary difference between columns 1 and 2 is the propellant choice for the capture stage. As compared to the all-chemical DRM, the SEPS configuration enables significantly lower total missions mass for a given payload, allowing for the use of a medium rather than a heavy lift launch vehicle. The stack masses of the two SEP configurations presented here are about 35% less than the all-chemical DRM system. Also, the baseline SEPS configuration of this study can deliver a modestly higher payload to Titan in a 3.5-year shorter trip time than the all-propulsive DRM.

Conclusions

This analysis showed that an advanced-ion-engine-based SEPS transfer stage can deliver a NASA DRM payload of 406 kg to Titan in missions using medium-class launch vehicles between the years 2007 and 2015. These missions can be accomplished with approximately 30 kWe of solar power driving four, 4000-s I_{sp}, NEXT-class thrusters with single VGA interplanetary transfers and SOA chemical technology for Titan capture. Payloads somewhat greater than the DRM can be delivered at transfer times of approximately 8.5 years using Atlas-V 421 or 431 launch vehicles. Smaller launch vehicles such as the Delta-IV 4450 could not deliver the reference Titan payload using the four ion engine/SOA chemical stage combination. For all of the launch vehicle cases investigated, increased Titan payload delivery was shown using a five-SEPS thruster system, but the improvement was marginal. A three-engine case required an ion system capability beyond the design throughput limits of current development programs.

Results of chemical capture stage analyses indicated that the baseline, off-the-shelf, storable NTO/N_2H_4 CPS can perform the mission. The advanced fluorine/N_2H_4 CPS provided a significant (25%) Titan payload increase over the baseline NTO/N_2H_4 stage, but would require a fluorine engine technology development program. The LOX/N_2H_4 CPS provided marginal performance improvement over NTO/N_2H_4 and probably does not warrant a new engine development program for this class of engine. Monopropellant-based CPS provides no improvements over SOA chemical unless the I_{sp} is greater than 320 s. Relatively easily obtainable tank technology improvements provide modest Titan payload delivery enhancement capability of the SEPS/chemical propulsion transportation system.

Acknowledgments

This work was performed by Science Applications International Corporation In-Space Technology Assessment program team under the support and leadership of the In-Space Propulsion Technology Project at NASA Marshall Space Flight Center. The efforts of William Hartmann, University of Illinois Urbana–Champaign, for Solar Electric Propulsion Trajectory Optimization Program trajectory generation, are gratefully acknowledged.

References

[1]Noca, M., Frisbee, R., Johnson, L., Kos, L., Gefert, L., and Dudzinski, L., "Evaluating Advanced Propulsion Systems for the Titan Explorer Mission," International Electric Propulsion Conference, Rept. IEPC-01-175, Oct. 2001.

[2]Herrmann, M. C., James, B. G., and Lockwood, M. K., "Aerocapture at Titan: Systems Analysis Review," NASA TM-2003-212746, Oct. 2003.

[3]Jankovsky, R. S., "HAN-Based Monopropellant Assessment for Spacecraft," AIAA Paper 96-2863, July 1996.

[4]"DELTA IV Payload Planners Guide," The Boeing Co., Rept. MDC 00H0043, Huntington Beach, CA, Oct. 2000.

[5]"Atlas Launch System Mission Planner's Guide," Lockheed Martin, Rept. CLSB-0105-0546, Rev. 9, McLean, VA, Sept. 2001.

[6]Ace Able Engineering, Inc., "Ultraflex Solar Arrays," URL: http://www.aec-able.com [cited 20 October 2003].

[7]Kerslake, T., "Photovoltaic Array Performance During an Earth-to-Jupiter Heliocentric Transfer," NASA John H. Glenn Research Center at Lewis Field, Power and Propulsion Office, Rept. IBR PS-496, Aug. 2000.

[8]*NASA Research Announcement Proposal Information Package, Next Generation Ion Engine Technology*, NASA, Nov. 2001, Sec. A.9.2.

[9]Patterson, M., Haag, T. W., Foster, J. E., Rawlin, V. K., Roman, R. F., and Soulas, G. C., "Development Status of a 5/10-kW Class Ion Engine," AIAA Paper 2001-3489, July 2001.

[10]"Advanced X-ray Astrophysics Facility Mass Properties," NASA Rept. DPD692 SE09, March 1999.

[11]Dickerson, R. E., and Gray, H. B., *Chemical Principles*, W. A. Benjamin, Menlo Park, CA, 1970 (1972 corrected printing).

[12]Sauer, C., Jr., "Optimization of Multiple Target Electric Propulsion Trajectories," AIAA Paper 73-205, Jan. 1973.

[13]Sauer, C., Jr., "Solar Electric Propulsion Performance For Medlite and Delta Class Planetary Missions," American Astronautical Society, Paper 97-726, Aug. 1997.

[14]Coverstone, V., Hartmann, J., and Woo, B., "Outer-Planet Mission Analysis Using Solar-Electric Ion Propulsion," American Astronautical Society, Paper 03-242, Feb. 2003.

[15]Brown, C., *Spacecraft Mission Design*, 2nd ed., AIAA Education Series, AIAA, Reston, VA, 1998, pp. 115–117.

[16]Appel, M. A., Kaplan, R. B., and Tuffias, R. H., *Liquid Fluorine/Hydrazine Rhenium Thruster Update*, Vol. 1, Marquardt Co., Van Nuys, CA, 1980, pp. 85–90.

[17]Bond, D. L., "Technology Status of a Fluorine–Hydrazine Propulsion System for Planetary Spacecraft," AIAA Paper 79-0907, May 1979.

J. Martin
Associate Editor