Ion Propulsion Technology for Fast Missions to Pluto

Dave Rodgers* and John Brophy**
Jet Propulsion laboratory
California Institute of Technology
Pasadena, California
David.H.Rodgers@jpl.nasa.gov,
John.R.Brophy@jpl.nasa.gov

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Ion propulsion, based on the technology validated on NASA's Deep Space 1 (DS1) spacecraft, provides substantial technical and programmatic advantages for a fast Pluto flyby mission. With an Atlas IIIB launch vehicle and a Venus gravity assist trajectory, a five-engine ion propulsion system providing a total ΔV of 11 km/s, can deliver a 430 kg spacecraft to Pluto in 11.5 years. The use of ion propulsion enables the use of a smaller launch vehicle, enables yearly launch opportunities and eliminates the requirement for a Jupiter gravity assist. Substantial programmatic flexibility is provided by the ion propulsion system that enables flight time to be traded for spacecraft mass at the rate of approximately 135 kg per year of additional flight time. Advanced versions of the DS1 ion propulsion technology, characterized by higher specific impulse and greater engine life, can reduce the solar array power required from 18 kW to 13 kW while simultaneously reducing the flight time to 11 years.

Introduction and Rationale

A mission to Pluto requires a very high ΔV to arrive in a reasonable time. This can be accomplished by using chemical, gravity assist, or electric propulsion systems. The low exhaust velocities of chemical rockets imply massive propulsion systems. Jupiter gravity assists (JGA) mitigate this but impose additional onerous constraints. The optimum JGA is only available every 12 years and carries a significant radiation penalty (~100 krad). There are other JGA opportunities but they are much less helpful than when launching at the optimum time. This newly recognized accident scenario raises safety questions regarding the use of a solid rocket motor with a Radioisotope Thermoelectric Generator (RTG), would impose additional implementation risk and cost. Solar Electric Propulsion (SEP) with its high exhaust velocity (~35 km/sec), offers an alternate approach that was validated by Deep Space 1 (DS1) [1-3].

A joint JPL/TRW team proposed just such a mission in the spring of 2001. The Principal Investigator (PI) was Dr. Lawrence Soderblom and the mission was called TEMPO (Tombaugh Explorer Mission: Pluto-Outbound). TEMPO uses SEP combined with a Venus Gravity Assist (VGA) to provide an 11.5-year flight time to Pluto with yearly launch opportunities. Table 1 compares TEMPO with the conventional chemical/JGA, ballistic mission design developed for the Pluto Kuiper Express (PKE) Mission. Major benefits arise from the SEP approach. The TEMPO spacecraft is shown in Fig. 1 in the powered cruise configuration.

The launch dates occur on roughly one year centers and have relatively wide launch windows since the SEP propulsion allows effective post launch trajectory shaping and reoptimization. Programmatically, yearly launch opportunities can be a major benefit, allowing funding to be much more flexible than when the launch is wholly constrained by planetary positions and ballistic trajectories. Flight time can also be traded for mass providing additional programmatic flexibility to deal with unforeseen

*Deputy Director, Center for In Situ Science and Sample Return
**Section Staff, Thermal and Propulsion Engineering, Member AIAA

Fig. 1 TEMPO spacecraft in the powered cruise configuration.

development or fiscal issues. This flexibility is shown in Fig. 2 below. These data indicate that for flight times in the range of 11 to 13 years the delivered mass capability increases at a rate of approximately 135 kg per year of added flight time.

The Ion Propulsion Module (IPM) supplies a ΔV of ~11 km/s. This enormous ΔV capability enables the required launch vehicle capability for an SEP mission to be significantly less than for a chemical/ballistic approach even with the assistance of a JGA. The SEP based system uses a trajectory which keeps the spacecraft in the inner solar system to build up velocity for approximately three years. The trajectory is then shaped and assisted by a Venus Gravity Assist (VGA) as shown in Fig. 3. The SEP thrust continues until roughly the orbit of Jupiter at which point the power available from the solar arrays is insufficient to operate the thrusters effectively.

Table 1. TEMPO versus PKE.

Factor Compared	**TEMPO (SEP)**	**PKE - chemical plus JGA**
Launch Dates	2006-9	2004
Launch Approval	Reasonable	Very difficult
Existing LV	Yes	No
Solid Kick Stage	No	Yes
Pointing control	1.5 mrad	5 mrad
Pointing stability	7 μrad/s	100 μrad/s
Avionics	TRW T310	X2000
Telecom X-band	20 W	5 W
Telecom Ka-band	27 W	none
HGA diameter	3.4 m	2.0 m
Radio Science Approach	Traditional downlink	New uplink technique
Playback period	1 month	~ 9 months

Earth Gravity Assist (EGA) trajectories also exist and are marginally more effective than VGA. They have the drawback that for a nuclear (RTG) powered

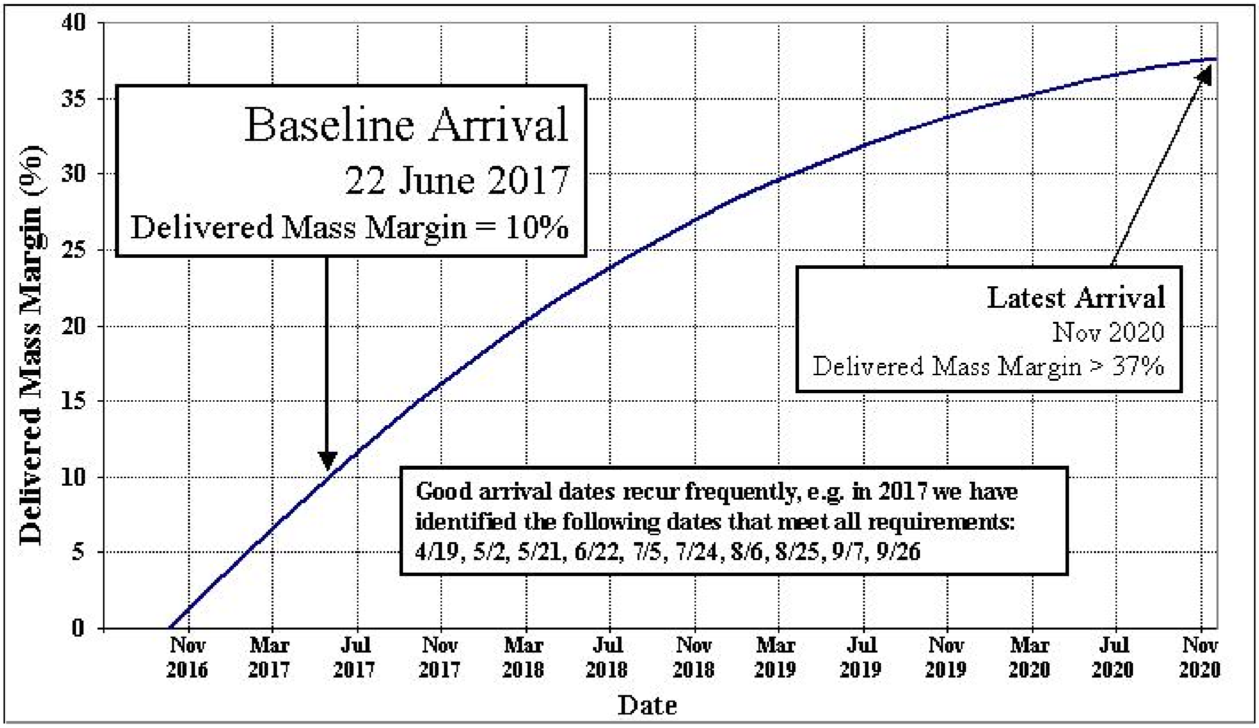

Fig. 2 Delivered Mass Margin increases rapidly with flight time

spacecraft they raise the issue of nuclear materials in the Earth's biosphere by accident or failure of the spacecraft. This issue has been successfully addressed by missions such as Cassini. However, EGA maneuvers will result in programmatic complications and additional costs to the mission related to meeting environmental impact regulations. VGA trajectories avoid the issues of post launch hazard to the Earth's biosphere and hence we selected such a trajectory.

Analysis has been performed that shows that the flight time of an SEP system is strongly influenced by the electric power available from the solar arrays. This is, in essence, a programmatic cost/schedule trade since larger arrays are more expensive and impose additional mechanical/thermal constraints on the spacecraft to provide articulation, mass and accommodation. The variation in net delivered mass with end-of-life solar array power (referenced to 1 AU) is given in Fig. 4 for a Delta IIIB launch vehicle, a Venus gravity assist trajectory, and an 11-year trip time. For the TEMPO mission a nominal 18kW at 1AU array was selected.

SEP can be integrated into the spacecraft design in a straight forward manner and all

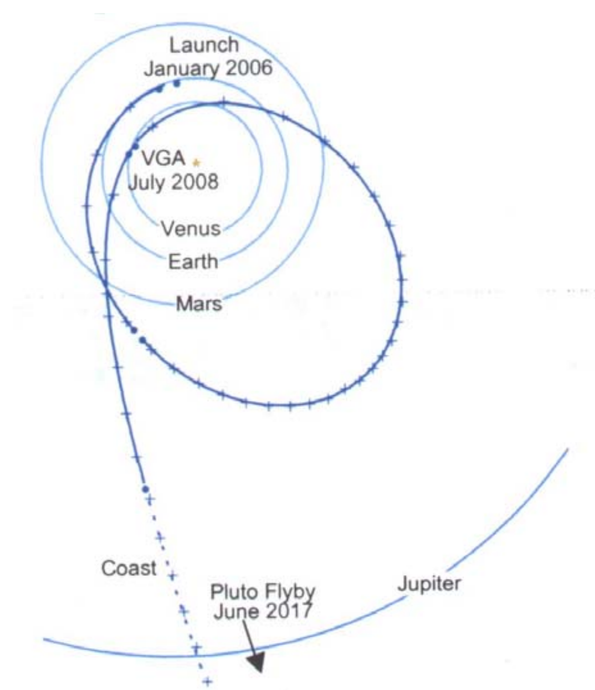

Fig. 3 Venus gravity assist trajectory for an ion propulsion mission to Pluto.

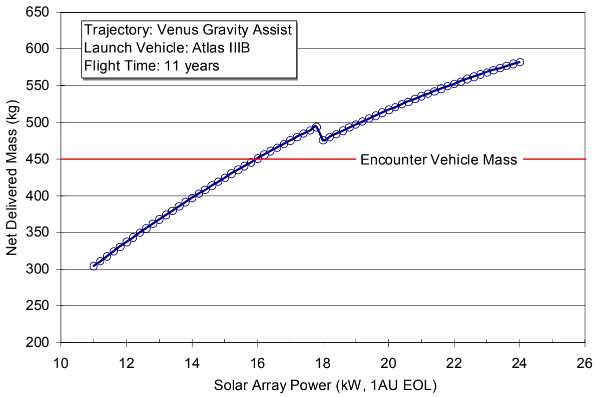

Fig. 4 Increasing solar array power increases the net delivered mass capability at a fixed flight time. The net delivered mass is everything that is not the ion propulsion system or solar array.

elements have been demonstrated either on DS1 (engines, thrusting, navigation, attitude control, and charging issues) or on large commercial satellites such as the Boeing 702 commercial spacecraft bus [4] (multi-engine operation, orbit transfer and attitude control with very large solar arrays). One highly advantageous use of SEP systems, demonstrated by DS1 is the use of the gimbaled electric engines to provide attitude control via thrust vector control. This minimizes the usage of attitude control propellant at least as far out as Jupiter. Either the control is used directly for spacecraft ACS or it can dump the accumulated momentum of the reaction wheels on spacecraft so equipped.

System Description

For TEMPO the ion propulsion system is configured as a propulsion module that is separable from the Encounter Vehicle (EV). It is the EV that performs the science measurements at Pluto. The Ion Propulsion Module (IPM) includes the Ion Propulsion System (IPS) hardware as well as the solar array, power management and distribution system, thermal control, structure, and mechanisms for separation from the launch vehicle and the encounter vehicle.

The driving requirements for the IPM are that it be single fault tolerant, be capable of providing the required ΔV of 11 km/s in as low a wet mass as possible, and that the perception of risk for the

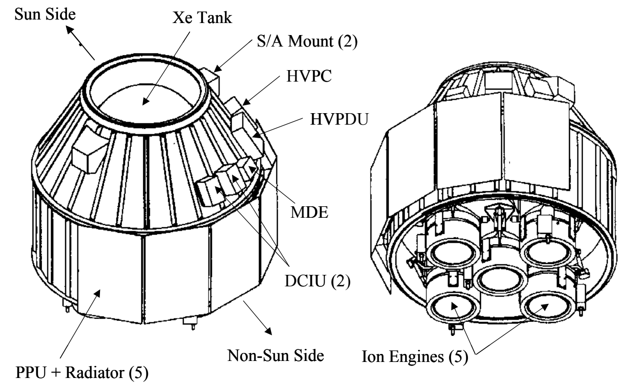

Fig. 5 Ion Propulsion Modoule (IPM) Configuration (without the solar arrays).

selected approach be a low as possible. This last requirement may seem somewhat non-technical, however, in an environment in which missions and their implementation are selected in a competitive process, the perception of risk becomes an important design driver.

The approach selected makes maximum use of heritage for the IPM, building on the success of the Deep Space 1 (DS1) and the adaptation of the Ultraflex solar arrays flight qualified for the Mars '01 Lander. The configuration of the IPM is given in Fig. 5. A mass list for the IPM is given in Table 2.

Ion Propulsion System (IPS): The IPS in the Ion Propulsion Module includes five NSTAR/DS1 ion engines and Power Processor Units (PPUs) [5,6]. Four of the five engines are attached to 2-axis gimbals and the center engine is fixed. The Xenon Feed System (XFS) provides storage capability for a maximum of 600 kg of xenon and controls the flow rate of xenon to each engine. A Digital Control and Interface Unit (DCIU) controls the XFS and PPUs based on high-level commands from the Encounter Vehicle. A second DCIU is included as a cold spare. The PPUs are cross-strapped to the thrusters as shown in the Fig. 6 where each PPU is connected to two thrusters. During the mission a maximum of four ion engines are operated simultaneously. With four engines operating, the IPM will process a maximum input power of 10 kW. Furthermore, no fewer than two engines are operated simultaneously to minimize the required gimbal angles and to enable 3-axis control of the spacecraft with the IPS during powered cruise. Simultaneous multiple-engine operation is now routine for orbit transfer of GEO comsats [4].

Table 2 IPM Mass List

		QTY	Unit Mass (kg)	Mass (kg)	Contingency	Total Mass (kg)
IPS				91.6		97.4
	DS-1 flight ion thruster	4	8.33	33.32	5%	34.99
	DS-1 PPU	4	13.33	53.32	5%	55.99
	DCIU	2	2.47	4.94	30%	6.42
Xenon feed system				22.8		29.1
	Xe Tank	1	13.00	13.00	30%	16.90
	VARIABLE REGULATOR	4	0.35	1.40	50%	2.10
	LATCH VALVES	11	0.10	1.10	20%	1.32
	PRESSURE TRANSDUCERS	7	0.25	1.75	10%	1.93
	FCD (WITH FILTER)	12	0.10	1.20	20%	1.44
	SYSTEM FILTER	1	0.25	0.25	20%	0.30
	SERVICE VALVE	11	0.12	1.32	10%	1.45
	TUBING	1	2.50	2.50	30%	3.25
	FITTINGS	1	0.30	0.30	50%	0.45
Power				110.0		143.0
	High Voltage Power Distribution Unit (HVPDU)	**1**	**4.00**	**4.00**	**30%**	**5.2**
	High Voltage Power Converter (HVPC)	**1**	**6.50**	**6.50**	**30%**	**8.5**
	Motor Drive Electronics	**1**	**4.00**	**4.00**	**30%**	**5.2**
	Transient Suppression Box	**1**	**0.65**	**0.65**	**30%**	**0.8**
	Solar Array BOL Power and Specific Mass	**16.5**	**kW**	**174**	**W/kg**	
	Solar Arrays (ultraflex)	1	94.83	94.83	30%	123.3
Structures/mechanisms				161.3		207.1
	Upper Equipment Module	1	28.00	28.00	30%	36.4
	Lower Equipment Module	1	38.00	38.00	30%	49.4
	Engine Assembly Plate	1	32.00	32.00	30%	41.6
	EAP Support Structure	1	13.00	13.00	30%	16.9
	Marmon Clamp Hardware	1	13.20	13.20	10%	14.5
	SEP thruster gimbals+actuators	**4**	**3.00**	**12.00**	**30%**	**15.6**
	S/A latch/release mechanism	2	3.20	6.40	30%	8.3
	S/A Gimbal Drive Assembly	2	5.00	10.00	30%	13.0
	System Assembly Hardware	0.02	437.13	8.74	30%	11.4
Cabling				24.4		31.8
	HVPDU Input Power Harness	1	0.60	0.60	30%	0.8
	HVPDU to HVPC Harness	1	0.41	0.41	30%	0.5
	HVPC to MDE Harness	**1**	**0.19**	**0.19**	**30%**	**0.2**
	DCIU Input Power Harness	**2**	**0.19**	**0.38**	**30%**	**0.5**
	PPU Input Power Harness	4	0.41	1.64	30%	2.1
	Engine Gimbal Actuator Harness	4	1.15	4.60	30%	6.0
	DCIU to PPU Harness	8	0.19	1.52	30%	2.0
	DCIU to XFS Harness	2	0.56	1.12	30%	1.5
	PPU to thruster cable	8	0.35	2.80	30%	3.6
	S/C to LV pass thorugh cables	**1**	**8.00**	**8.00**	**30%**	**10.4**
	S/C to DCIU (1553B)	**2**	**0.50**	**1.00**	**30%**	**1.3**
	S/C to HVPDU (control)	1	0.19	0.19	30%	0.2
	S/C to HVPC (control)	1	0.19	0.19	30%	0.2
	S/C to MDE (control)	1	0.19	0.19	30%	0.2
	S/C to S/A (power for deployment)	4	0.40	1.60	30%	2.1
	Other Misc. Cabling	0	2.00	0.00	30%	0.0
Thermal				35.7		45.7
	Thermal control (per PPU)	4	5.00	20.00	30%	26.0
	Thermal Blankets	1	10.60	10.60	30%	13.8
	Heaters and PRT's	1	2.00	2.00	30%	2.6
	16 blade Louvers	**4**	**0.8**	**3.1**	**5%**	**3.3**
	TOTAL DRY:			**446**		**554**
	Xenon deterministic prop.	1	**469**			
	Xenon Contingency for Nav. and Trajectory	**2%**	**9.4**			
	Xenon Contingency for Other Errors	**5%**	**23.5**			
	Total Xenon Stored		502			502
	TOTAL WET:					***1056***

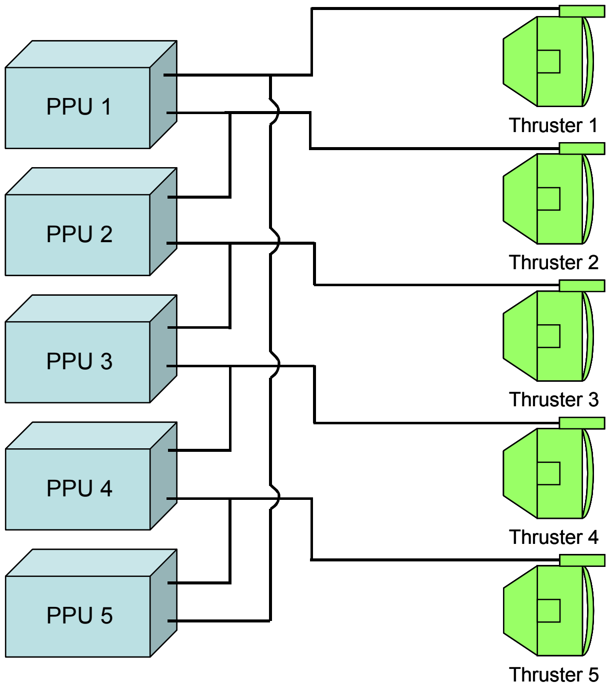

Fig. 6 PPU to engine cross strapping.

The ion engines are identical to those flown on DS1. Thermal modeling was performed by Ray Becker at JPL over a range of solar ranges from 0.6 to 1.2 AU for the worst-case illumination conditions with four engines operating. The thermal model is shown in Fig. 7 indicating the extent to which the thrusters are buried inside the launch vehicle adapter. The results of this modeling are given in Table 3 for the worse case combination of solar range and sun angel. These are for the center engine and were calculated assuming that all five engines were operating at full power. The mission design assumes that no more than four engines are operated simultaneously so these results are expected to be conservative. The calculated temperatures indicate that there are no significant thermal issues associated with the simultaneous operation of five engines at full power at the worst-case solar illumination conditions expected during the mission.

The guideline for the use of the NSTAR/DS1 ion engine is that each engine can process 130 kg of Xe with a low risk of engine wear-out failure. This guideline has been adopted by NASA's Discovery Program [7]. It should noted that since this throughput limit was established in the spring of 2000, the DS1 flight spare ion engine has processed more than 155 kg of xenon (as of the end of September 2001) in an ongoing life test at JPL [8].

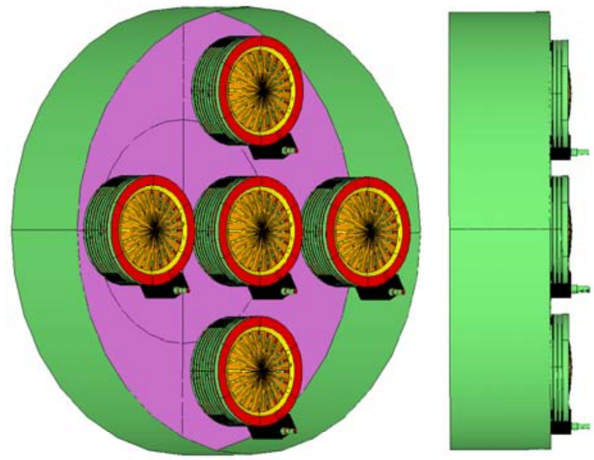

Fig. 7 Ion engine thermal model configuration showing engines partially buried in the launch vehicle adapter.

Table 3. Calculated Center Engine Temperatures for Worst-Case Solar Range and Sun Angle

Heliocentric Distance	0.7 AU	0.7 AU
Sun Angle with respect to Thrust Axis	70 deg.	70 deg.
Thruster Throttle Level (All 5 thrusters)	TH15	Off
Engine Component	**Temperature (^{o}C)**	
Front Mask	190	117
Aft Magnet Ring	228	86
Middle Magnet Ring	243	93
Forward Magnet Ring	313	127

The TEMPO mission requires the processing of 510 kg of Xe, this is equivalent to 102 kg of Xe per engine or 128 kg Xe per engine in the case of one failure. The actual propellant loaded is 10% greater than 510 kg (for a total xenon load of 560 kg) to account for ullage, filling errors, flow rate uncertainties, leakage, start/stop losses, navigation uncertainties, and trajectory modeling errors. This allocation of "extra" xenon is consistent with mission design used on DS1. The xenon tank is sized for 600 kg to accommodate changes in the required xenon load without requiring redesign of the tank.

The PPUs are identical to those flown on DS1 [6]. They accept unregulated power from the solar array and convert it to the currents and voltages necessary to start and run the ion engines. The PPUs

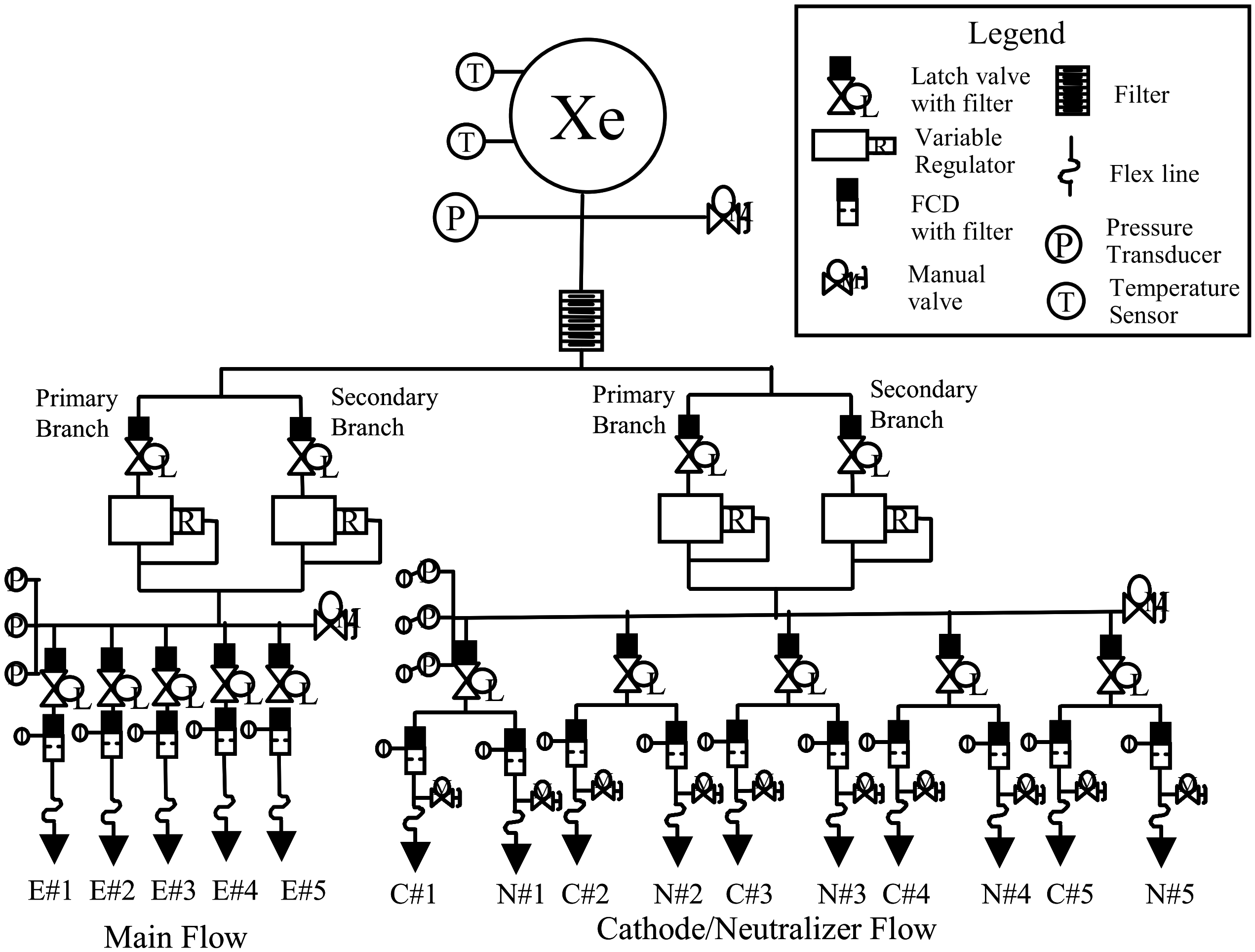

Fig. 8 Xenon feed system for the TEMPO ion propulsion system.

operate with input voltages of 80 V to 160 V to accommodate the variation in array voltage with solar range.

The XFS includes the xenon storage tank, the Xenon Control Assembly (XCA), and the propellant lines. A block diagram of the XFS is given in Fig. 8. The xenon tank has a thin-walled aluminum liner and a composite over wrap. The tankage fraction for this tank technology is only 0.025. This lightweight tank technology is being flight qualified for the Mar Exploration Rover (MER) program. The tank for IPM will be a new size, but will use the same technology. A qualification program for this new tank size, and for operation with xenon, is required.

The XCA is similar to that used on DS1 except that the bang-bang pressure regulation system is replaced with a more modern variable pressure regulator developed since the launch of DS1. The use of this new component provides a significant mass and volume reduction in the XCA, but requires modifications to the DCIU hardware and software.

The DCIU accepts high level commands from the EV over a MIL-STD-1553B data bus and controls the operation of the PPUs and the XFS. It also digitizes and formats data from the XFS, and PPUs and inputs this information to the EV.

A simple, lightweight ion engine gimbal design is used to provide +/-5^{o} gimbal capability in two axes. This gimbal design provides a substantial mass and volume advantage over that used on DS1. During powered cruise, the encounter vehicle avionics controls the engine gimbals to provide three-axis control of the entire vehicle.

IPM Power System: The power for the IPM is provided by two 9-kW (beginning of life (BOL) at 1 AU) solar array wings based on the Ultraflex design (from Able Engineering), each with a deployed diameter of 6.2 m. The array design uses 26.5% efficient triple junction (TJ) solar cells wired to provide a nominal array output voltage of 100 V. A flight Ultraflex solar array design was successfully developed for a 900-W wing (BOL at 1 AU) for the Mars '01 lander. The two flight solar arrays wings fabricated for Mars '01 were designed to be deployable in 1-g for ground testing and Mars deployment. Nevertheless they have an outstanding specific power of 103 W/kg.

The GaInP2/GaAs/Ge triple junction solar cells planned for TEMPO are a derivative of the current dual junction cells now flying on commercial GEO comsats. Over 100 kW of dual junction solar cells are currently flying. Triple junction cells at 26.5% efficiency have been flight qualified and are currently in production.

Commercial communications satellites are now flying with BOL power levels up to 20 kW, which is more than the BOL solar array power of 18.5 kW for the TEMPO IPM.

Several risk mitigation activities in the development of the solar array are required including: survivability of the panel design to hot and cold temperature extremes; validation of the panel's high voltage design; Low Intensity Low Temperature (LILT) characterization of the TJ cells; and radiation characterization of the TJ cells. Finally, a full-scale structural model of one 9-kW Ultraflex wing should be made to validate the wing mechanical design and deployment.

The 18-kW BOL power at 1AU from the Ultraflex array is much greater than the 10-kW maximum power that the ion propulsion system can process because the solar array size is determined by the power required at solar ranges greater than 1 AU. When the available power is greater than that required by the IPM, the excess power is left on the array. The thermal impact of leaving the excess power on the solar array has been evaluated and is not a problem even at 0.6 AU.

The solar array wings are articulated in one-axis using a Solar Array Drive Assembly (SADA). The power is transferred across the gimbal interface using a cable wrap since this is the lowest risk approach. During normal operation at solar ranges less than 1.0 AU the array is feathered for thermal control. Flight qualified high temperature cell adhesives and welded interconnects will be used to assure that the solar array will have adequate temperature margin for operation at a solar range of 0.6 AU even if the array is not feathered.

During powered cruise 300 W of solar array power is allocated to the IPM to operate the DCIU and PPU control circuitry, the engine and solar array gimbal actuators, and to power heaters for thermal control (including replacement heat for non-operating PPUs).

The power and data interfaces for the IPM are shown in the block diagram Fig. 9. The IPM contains no battery. All power for the IPM is provided by the solar array with two exceptions: the encounter vehicle provides power for the initial deployment of the solar array, and for articulation of the array under some fault conditions. The IPM executes a Power-On-Reset whenever power is restored to the module following a fault in which the sun is off the solar arrays.

The High Voltage Power Distribution Unit (HVPDU) distributes the solar array power to the PPU's, the High Voltage Power Converter (HVPC) and the thermostatically controlled heaters on the IPM. The HVPDU provides spacecraft-commandable switches that can remove the high voltage input to each PPU in the event of a PPU fault.

The internally redundant HVPC converts the solar array output voltage to 28 V to power the DCIU, PPU logic circuits, and the gimbal actuators. The HVPC provides power to one of the two DCIUs as directed by the encounter vehicle. The Motor Drive Electronics (MDE) controls the ion engine and solar array gimbal actuators based on spacecraft input through a 1553B interface.

IPM Structure: The IPM structure consists of three major pieces: the upper epipment module (UEM), the lower equipment module (LEM), and the engine assembly plate (EAS).

Primary loads are carried through the UEM and LEM. The Xenon tank is internally mounted to the UEM via a composite skirt. The HVPDU, HVPC, DCIU, MDE, and solar arrays are externally attached to the UEM with the avionics boxes on the non-sun side. The top of the UEM provides a marmon clamp band that mates the IPM to the T310. All five PPU's are attached to the LEM on the anti-Sun side. The base of the LEM provides a standard marmon clamp interface to the launch vehicle. The UEM and LEM are designed to separate for easy integration of the EAS and XFS.

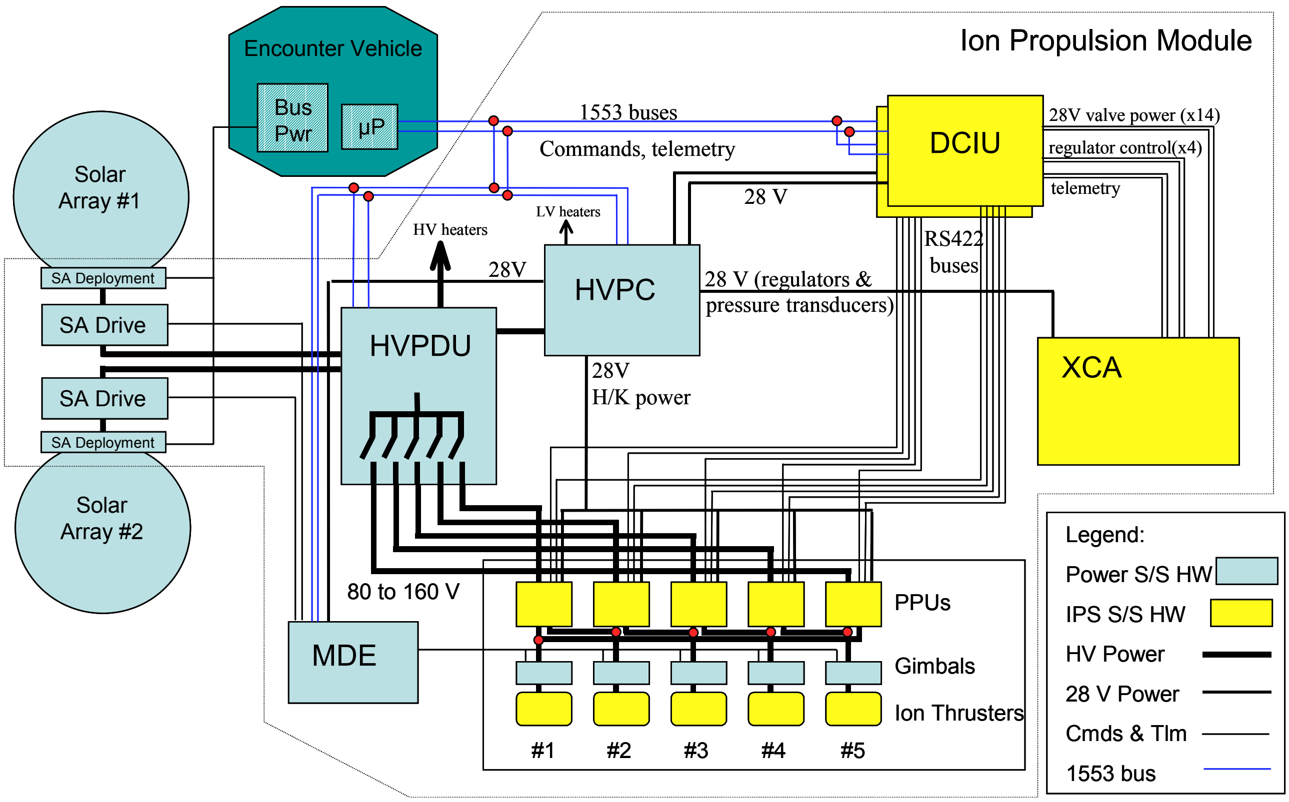

Fig. 9 Simplified block diagram of the power and data system for the TEMPO IPM.

The EAS integrates the engines, engine gimbals, and XCA components. When the EAS and UEM are integrated, the entire XFS is accessable for final assembly and inspection

IPM Thermal Design: The IPM thermal design utilizes a thermally coupled approach that maximizes power sharing and provides modulation of temperatures via louvers. All avionics boxes are thermally coupled to and placed on the anti-Sun side of the primary structure. The XFS is packaged in the interal volume of the UEM and LEM which is thermally controlled. Combining the use of louvers on the PPU radiators, thermal blankets, and the thermal heat dissipated by the electronics, the entire IPM system is coupled and regulated as one unit. This design minimizes the quantity of and total power required for electrical heaters. This also eliminates the risk and complexity of many line heaters.

IPM Capabilities

With a January 2006 launch with an Atlas IIIB, the ion propulsion module can deliver a 430-kg Encounter Vehicle to Pluto in 11.5 years. This includes 3.1 years for the solar electric Venus gravity assist trajectory followed by an 8-year ballistic trajectory to Pluto arriving June 2017. The ion propulsion system provides a ΔV of approximately 11 km/s.

The trajectory given in Fig. 3 assumed a SEP duty cycle of 92% and use end of life engine performance throughout the powered cruise phase of the mission. Before the failure of the start tracker on DS1 the demonstrated SEP duty cycle was 92%. After the star tracker failure the ion propulsion system was used to provide pitch and yaw control of the spacecraft in order to save hydrazine propellant needed for the comet encounter. Consequently, the ion propulsion system was operated almost all of the time and the demonstrated SEP duty cycle was greater than 99% for more than 14 months.

Development Time

The use of the NSTAR/DS1 ion engines and PPUs makes the development time for the IPM manageable. The long-lead item for the ion engine is the extensive life testing required to validate the engine service life. The NSTAR ion engine technology has been undergoing life testing since 1994 [9-16] and has been subjected to two very long-duration tests in order to identify unknown failure modes and to characterizer the long-term behavior of known failure modes. One of these tests operated the thrust or for 8200 hours at full power and demonstrated the full designed life capability of the engine [11]. The second test used the DS1 flights spare engine and demonstrated 150% of the engine design life [12]. This test was subsequently extended with the goal of running the engine until it failed. As of the end of September 2001 this engine has accumulated more than 18,800 hours of operation and has processed over 155 kg of Xe (or 187% of the original engine design life). In addition the NSTAR/DS1 ion engine has also logged over 14,200 hours of successful operation in deep space processing over 65 kilograms of Xe, and enabling the spectacular encounter with the comet Borrelly on September 22, 2001. This heritage makes the use of the NSTAR/DS1engine technology very attractive to flight projects.

The use of a new ion engine technology would require a similarly extensive life test program. The schedule implications of such a program are typically unacceptable to flight project. Therefore, it is almost certain that any new ion engine technology will have to have completed its service life validation prior to be beginning of phase C/D. Furthermore, since the phase B duration of a typical flight project is much shorter than the time required to validate the ion engine service life, this service life validation must be performed by a technology program funded separately from the flight project.

Perceived risk issues

There are several significant perceived risk areas associated with the ion propulsion module described here for the Pluto mission. The first of these is the large ultra-flex solar arrays. The ion propulsion module for Pluto requires an order of magnitude increase in the size of the ultra-flex solar array technology. While this appears to be largely an engineering task there are schedule and cost risks associative with this scale up. In addition, deployment and control of these large solar arrays is also perceived as risky. However, solar arrays of this size, but of a different design, are currently flooding on GEO comsats.

The second risk area is the simultaneous use of up to four ion engines with a maximum system power of 10 kW. However, simultaneous operation of two ion engines with a system power of 8.5 kW is now routine on GEO comsats. Modeling indicates that simultaneous operation of multiple engines at 0.6 AU is not a problem from an engine thermal standpoint.

A third risk area is that the ion propulsion module must operate over a large solar range from 0.68 AU to nearly 5 AU. This is a much larger solar range than was demonstrated on DS1. However, there are no known problems with the ion propulsion system provided the hardware is maintained within its temperature specifications. The solar cells must be tested for low temperature low intensity effects.

Impact of Improved Technologies

The introduction of improved ion engine and PPU technologies can provide significant benefits for missions to Pluto. For example, increasing the specific impulse and propellant throughput capabilities from the NSTAR values of 3100 seconds and 130 kg to 3700 seconds and 200 kg, respectively, enables the reduction of the number of client engines required in the IPS from the 5 to 4. This has significant cost, mass, and packaging benefits. In addition, these improvements also result in shorter trip times. The 3700-second specific impulse and 200-kg throughput capability represents a near-term derivative of the NSTAR/DS1 technology and is designated NSTAR-2. A more advanced derivative of the NSTAR Technology, designated NSTAR-3, seeks to increase the specific impulse to 5000 seconds while increasing the throughput capability to 400 kilograms of Xe. The performance characteristics of these three technology options are summarized in Table 4.

The performance capabilities of the NSTAR, NSTAR-2, and NSTAR-3 technologies are given in Fig. 10 as a function of flight time for an EOL solar array power of 18 kW. These data clearly indicate that increased delivered mass and charter flight times are enabled by improved technologies. However, the biggest impact in flight time is obtained by increasing the solar array power level.

The propellant tank and Xe feed system identified in the ion propulsion module already have outstanding mass characteristics and there is little to be gained through technology improvements of these components for application to spacecraft of this size.

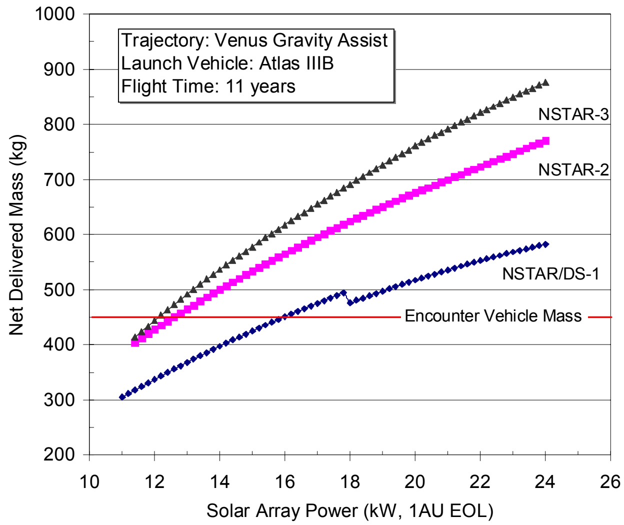

Fig. 10 Advanced NSTAR-derivative technologies provided significant benefits for the TEMPO mission.

The incremental cost for solar array power is on the order of $1M a kilowatt. In a cost constrained environment increasing the solar array power is not always an affordable option. The best ion propulsion systems are those that provide the required ΔV and flight times with the minimum beginning of life solar array power. Decreasing the solar array specific mass has a substantial impact on the overall system performance because the solar array mass is a substantial fraction of the total ion propulsion module mass.

Table 4. Performance Goals for NSTAR-Derivative Technologies

Parameter	NSTAR/ DS1	NSTAR-2	NSTAR-3
Max. Isp (s)	3100	3700	5000
Max. PPU Input Power (kW)	2.5	3.4	5.3
Max. Thrust (mN)	92	122	145
Xenon Propellant Throughput Capability (kg)	130	200	250

Conclusions

A fast flyby mission to Pluto requires a very high post-launch ΔV to achieve a reasonable flight time. The conventional approach has been to assume the use a Jupiter gravity assist to provide a large fraction of this ΔV. The optimum Jupiter gravity assist, however, is available only every 12 years and the last good opportunity requires a launch in 2004. On the other hand, the use of solar electric propulsion based on the NTAR ion propulsion system successfully validated on Deep Space 1 enables several significant benefits relative to the conventional approach. An SEP mission to Pluto and the Kuiper belt called TEMPO was proposed in the spring of 2001 and detailed these benefits. The use of SEP on TEMPO enables:

1. Yearly launch opportunities with an 11.5-year flight time resulting in substantial programmatic resiliency.
2. The use of a smaller, less expensive launch vehicle. TEMPO baselined the use of an Atlas IIIB launch vehicle
3. The elimination of the solid rocket motor upper stage. This feature is expected to significantly facilitate the launch approval process required for an RTG-powered spacecraft.
4. The flexibility to trade flight time for mass at the rate of 135 kg per year of added flight time.
5. The elimination of the radiation issues associated with a Jupiter gravity assist trajectory.

Advanced SEP technologies based on derivatives to the NSTAR/DS1 system provide additional benefits to this mission. These improvements to the NSTAR technology include operation at higher specific impulse levels and a greater propellant throughput capability per engine. The mission benefits provided by these improvements include either increased mass delivery capability or reduced solar array power. In addition, the advanced technology systems required fewer thrusters and PPUs resulting in reduced cost of the propulsion system.

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