30-kW SEP Spacecraft as Secondary Payloads for Low-Cost Deep Space Science Missions

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The Solar Array System contracts awarded by NASA's Space Technology Mission Directorate are developing solar arrays in the 30 kW to 50 kW power range (beginning of life at 1 AU) that have significantly higher specific powers (W/kg) and much smaller stowed volumes than conventional rigid-panel arrays. The successful development of these solar array technologies has the potential to enable new types of solar electric propulsion (SEP) vehicles and missions. This paper describes a 30-kW electric propulsion vehicle built into an EELV Secondary Payload Adapter (ESPA) ring. The system uses an ESPA ring as the primary structure and packages two 15-kW Megaflex solar array wings, two 14-kW Hall thrusters, a hydrazine Reaction Control Subsystem (RCS), 220 kg of xenon, 26 kg of hydrazine, and an avionics module that contains all of the rest of the spacecraft bus functions and the instrument suite. Direct-drive is used to maximize the propulsion subsystem efficiency and minimize the resulting waste heat and required radiator area. This is critical for packaging a high-power spacecraft into a very small volume. The fully-margined system dry mass would be approximately 1120 kg. This is not a small dry mass for a Discovery-class spacecraft, for example, the Dawn spacecraft dry mass was only about 750 kg. But the Dawn electric propulsion subsystem could process a maximum input power of 2.5 kW, and this spacecraft would process 28 kW, an increase of more than a factor of ten. With direct-drive the specific impulse would be limited to about 2,000 s assuming a nominal solar array output voltage of 300 V. The resulting spacecraft would have a beginning of life acceleration that is more than an order of magnitude greater than the Dawn spacecraft. Since the spacecraft would be built into an ESPA ring it could be launched as a secondary payload to a geosynchronous transfer orbit significantly reducing the launch costs for a planetary spacecraft. The SEP system would perform the escape from Earth and then the heliocentric transfer to the science target.

I. Introduction

Solar electric propulsion systems, with power levels of hundreds of kilowatts, have the promise of providing substantial reductions in the mass required to be launched to low-Earth orbit (LEO) for a variety of future human exploration missions (see for example Refs.1-10). The highest power SEP spacecraft flying today in deep space is the Dawn spacecraft which has a 10-kW solar array (beginning of life at 1 AU) and can process a maximum input power to the electric propulsion system of 2.5 kW. The largest commercial communication satellites using solar electric propulsion have 24-kW solar arrays (BOL) with existing electric propulsion systems that can process up to 9 kW. Using existing rigid-panel solar arrays and electric propulsion technologies it is likely that multiple organizations could develop a 30-kW-class solar electric propulsion vehicle without the need for a technology demonstration mission, and such a vehicle could potentially provide attractive performance for certain types of cargo missions in

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cislunar space. However, it would be difficult to scale this approach to significantly higher power levels envisioned to be necessary to support ambitious human missions beyond low-Earth orbit.

For this reason NASA's Space Technology Mission Directorate (STMD) is sponsoring the Solar Array System (SAS) development of lightweight solar arrays in the 30- to 50-kW range (BOL at 1 AU) that are extensible to power levels of order 300 kW. A key feature of these solar array developments is the goal to enable packaging the arrays for launch in a significantly smaller volume than conventional rigid-panel arrays. STMD has awarded two contracts for solar array development, one to ATK for the development of the MegaFlex concept, and the other to Deployable Space Systems (DSS) for development of the Mega Roll-Out Solar Array (Mega-ROSA) concept.[11]

To reduce the risk for the future development of a multi-hundred kilowatt SEP vehicle and to produce a high-performance near-term SEP capability, NASA has recognized the need for a high-power SEP Technology Demonstration Mission (TDM). Such a mission would address the advanced features a high-power SEP system that can't be adequately tested on the ground. Such features could include deployment of the large SAS solar arrays, operation of high-voltage (~300-V) solar arrays in the plasma environment produced by the electric propulsion system, and demonstration of direct-drive with multiple Hall thrusters operating in parallel. In-house and industry cost estimates for a high-power SEP TDM suggest that a major partner would likely be needed to improve the affordability of the TDM. One potential partnership would be to combine the SEP TDM with the Asteroid Redirect Mission[12] currently being studied by NASA.

The work described herein, however, assumes that no major partnerships are available and presents an approach to keeping the total SEP TDM costs below an assumed maximum value of approximately $250M. Derivatives of the resulting SEP TMD vehicle may be useful in their own right as an affordable approach for planetary science missions.

II. Study Approach

The study was conducted by the representatives from JPL and GRC listed in the Acknowledgements. The general approach followed a design-to-cost methodology in which costs were managed by requiring the system to be compatible with a secondary ride into space and by limiting the total vehicle power level as necessary. The mission objective was assumed to be to demonstrate only the high-tech features that require a flight test to convince skeptics. The study team generated the draft Level 1 and Level 2 requirements listed in Tables 1 and 2. Based on these requirements the mission and flight system were defined, a master equipment list (MEL) was developed and multiple cost estimates were generated.

Table. 1 Level 1 Requirements.

#	Requirement
1	Mitigate the risks associated with the development of a multi-hundred kilowatt SEP vehicle that can only be addressed through a flight test – assume there will only be one flight test before the development of the multi-hundred kilowatt SEP vehicle.
2	Demonstrate in-space operation of SEP technologies that are essential to a multi-hundred kilowatt SEP vehicle and that are not amenable to ground-based testing.
3	Launch before the end of the decade.
4	The cost for the SEP TDM shall not exceed $250M including the launch vehicle.

Table 2. Level 2 Requirements

#	Requirement
1	The total mission cost to NASA OCT for the SEP TDM shall not exceed $250M with a goal of $200M (FY13) including the launch costs
2	The SEP TDM shall be compatible with a launch date no later than December 31, 2019
3	The SEP TDM shall be compatible with the results from the Solar Array System (SAS) NRA
4	The SEP TDM shall demonstrate operation of a high-voltage (> 200 V) solar array in the plasma environment experienced by a spacecraft using electric propulsion
5	The SEP TDM shall demonstrate operation of a high-voltage (> 200 V) solar array in the natural environment associated with low-Earth orbit
6	The SEP TDM shall demonstrate operation of a high-voltage (> 200 V) solar array in the natural environment associated with the Earth's radiation belts
7	The SEP TDM shall demonstrate in-space deployment of a large-area, flexible-blanket solar array structure
8	The SEP TDM shall demonstrate in-space direct-drive operation of an electric propulsion subsystem
9	The SEP TDM shall demonstrate in-space direct-drive operation of at least two electric thrusters in parallel

One of the keys for an affordable TDM is how to manage the launch costs. To meet the cost objectives only shared rides were considered in this study including: the EELV Secondary Payload Adapter (ESPA);[13] Geosynchronous communication satellite secondary payloads (see for example Ref. 14), and the Space X Dragon Trunk.[15] Because it was perceived to provide the most mission flexibility and also provided well defined interfaces the ESPA ring approach was selected for this study. With this selection, the study then addressed the following three questions:

1. What can we physically fit on an ESPA ring?
2. What can we afford to put on an ESPA ring?
3. Is the resulting flight system worth flying, i.e., does it meet enough of the Level 1 and 2 requirements?

III. Results

To meet the Level 1 and 2 requirements the following mission was defined. The launch date was assumed to be in 2019 with a total mission duration of 90 days. The spacecraft would be launched as a secondary payload on an ESPA ring to either low-Earth orbit (LEO) or a geosynchronous transfer orbit (GTO) and then spiral to a higher Earth orbit. The electric propulsion system would be turned off during eclipses. The maximum propellant usage was limited to 200 kg and once the propellant is exhausted the mission was assumed to be over. The flight system was assumed to be Class D, single string. No science instruments were included since this was assumed to be a technology demonstration mission. However, it was assumed that there would be on-board cameras to observe the deployment of the solar arrays.

Three flight system options, listed in Table 3, were identified based on the two SAS solar array concepts. The first option assumes the use of two 7.5-kW ROSA wings along with two 5-kW Hall thrusters operated direct-drive. The second option would have two 15-kW MegaFlex solar array wings and two 14-kW Hall thrusters operated direct-drive. The third option would only partially populate the 15-kW-sized MegaFlex wings with solar cells to produce 7.5-kW per wing. It would then use the same propulsion system as Option 1. The reason for depopulating the wings in Option 3 was simply to reduce cost consistent with the design-to-cost approach adopted for the study.

Table 3. Flight System Options.

Parameter	Option 1	Option 2	Option 3
Solar Array Type	ROSA	MegaFlex	MegaFlex
Solar Array Wing Dimensions	3.5 m x 12 m	8.3-m diameter	8.3-m diameter
Power per Wing (BOL)	7.5 kW	15 kW	7.5 kW
Input Power per Hall Thruster	5 kW	14 kW	5 kW
Number of Hall Thrusters	2	2	2
Direct-Drive	Yes	Yes	Yes
Xenon Propellant	220 kg	220 kg	220 kg
Hydrazine Propellant	26 kg	26 kg	26 kg
Dry Mass (fully margined)	744	1122	990

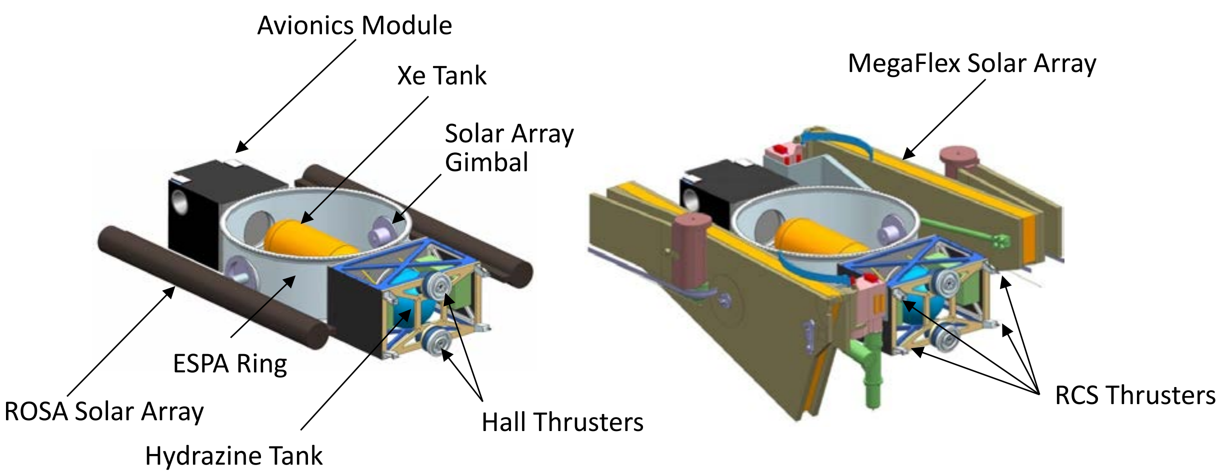

Figure 1. Stowed configurations for Option 1 (left) and Options 2 & 3 (right) with a standard ESPA ring.

The flight configurations for corresponding to Option 1 and Options 2 & 3 are shown in Fig. 1. Each of these options uses the standard ESPA ring as the primary spacecraft structure. Separation mechanisms at both the top and bottom of the EPSA ring would be needed to separate the spacecraft from the primary payload and the launch vehicle. The following features are in common for all three options. Each system would include two Hall thrusters each attached to a two-axis mechanical gimbal, and both thrusters would be operated simultaneously during the nominal mission. The spacecraft would be 3-Axis stabilized. When thrusting with the electric propulsion (EP) system the Hall thrusters would provide pitch and yaw control and a single reaction wheel would be used for roll control. Monopropellant hydrazine Reaction Control System (RCS) thrusters would provide 2-axis control when not thrusting with the EP system, and the reaction wheel would still be used for roll control. Either set of thrusters could be used to unload the roll-control reaction wheel. The RCS includes four thrusters and 26 kg of propellant. The reaction wheel is used to roll the vehicle in conjunction with a single axis of rotation solar array gimbal to provide the two degrees of freedom necessary to maintain the solar arrays normal to the sun during spiral out in Earth orbit. Attitude determination is provided by a star tracker and gyro measurements blended in a filter (e.g., Kalman filter). Coarse sun sensors and gyros provide attitude information for initial deployment and safe mode. The spacecraft avionics is assumed to be a commercial system based on the RAD 750 microprocessor. The telecom system is assumed to use an S-band transponder with two fixed S-band low-gain antennas.

The use of direct-drive greatly reduces the amount of waste heat that the thermal control system must reject resulting in a simpler, lower-cost system and enabling the system to be packaged into a relatively small volume. Direct-drive systems with single and multiple Hall thrusters have been successfully operated on the ground in a series of tests designed to demonstrate feasibility and retire risks for future high-power direct-drive systems.[16,17]

The MegaFlex configurations (Options 2 & 3) result in violations of both the upper and lower separation planes (defined by the top and bottom of the ESPA ring). However, it is expected that the locations and extent of these violations would not be a problem in practice and could be negotiated with the primary payload and the launch vehicle. The ROSA configuration does not violate either separation plane. The 15-kW ROSA configuration (Option 1) is shown inside a 4-m launch vehicle fairing in Fig. 2. The 30-kW MegaFlex configuration is shown inside a 5-m fairing in this figure.

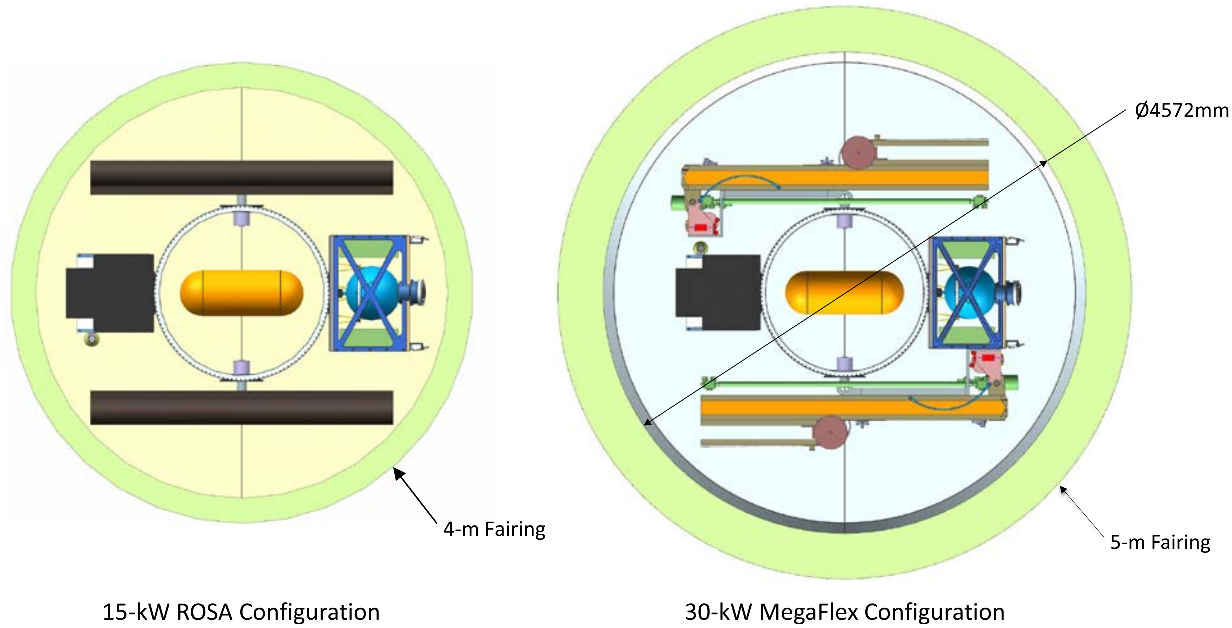

Figure 2. Option 1 (left) shown inside a 4-m launch vehicle fairing and Options 2 & 3 shown inside a 5-m fairing.

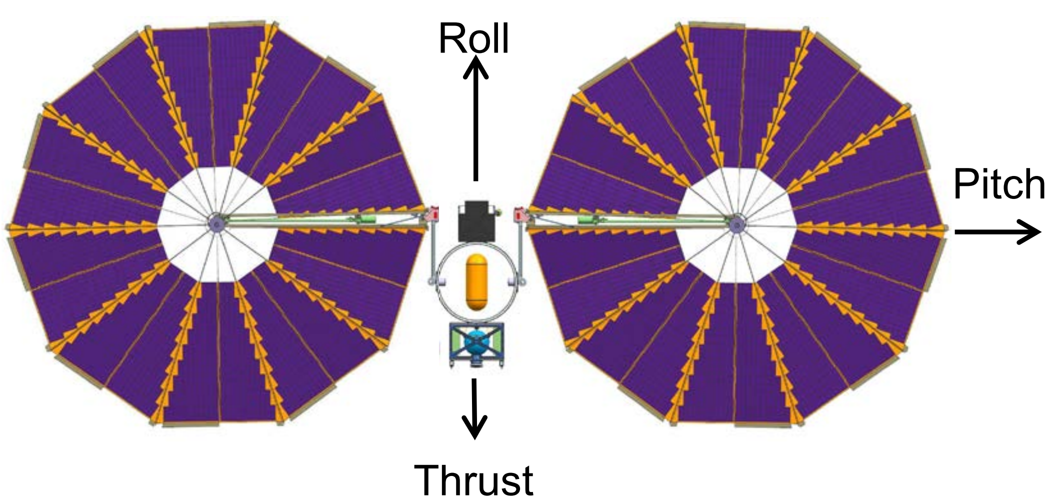

Figure 3. Flight configuration of the 30-kW MegaFlex option (Option 2).

The flight configuration for the 30-kW MegaFlex option is given in Fig. 3. The large solar arrays drive the design to rotate the spacecraft about velocity (roll) while rotating arrays about the pitch axis. A trade study indicated that the lowest mass approach was to rotate the vehicle about roll using a single reaction wheel. The wheel mass is more than offset by the resulting propellant mass savings. Minor modifications to heritage control algorithms would be required to use the single reaction wheel in a hybrid control mode along with the electric or RCS thrusters.

Mass and Performance Estimates

Roll-ups of detailed MELs are shown in Table 4 for each of the three options. All mass estimates meet the JPL Design Principles for margin. Each of these options assumes a total xenon mass of 220 kg stored in a single tank as indicated in Fig. 1. The total wet masses range from 990 kg for Option 1 with a 15-kW ROSA array to 1370 kg for Option 2 with a 30-kW fully-populated MegaFlex array. For reference, the fully-fueled Dawn spacecraft had an initial wet mass of 1218 kg including 425 kg of xenon.[18]

The potential performance of these vehicles is impressive. For Option 2 operating at 28 kW input to the electric propulsion system at a specific impulse of 2000 s, the total thrust is about 1.54 N, resulting in an initial vehicle acceleration of 1.1 mm/s. This is more than an order of magnitude greater than the initial acceleration of the Dawn spacecraft. Assuming the spacecraft can use 200 kg of the 220-kg initial xenon load, it could provide a total delta-V of 3.1 km/s in about a month. The lighter, lower-power spacecraft represented by Option 1 (with a maximum input power to the electric propulsion system of 10 kW) could provide a total delta-V of about 4.4 km/s with the same xenon load, but would take nearly three times as long. If we double the propellant load by adding another xenon tank as indicated in Fig. 4, then Option 1 could provide a total delta-V of nearly 8.5 km/s in about six months. Incredibly, Option 2 loaded with 440 kg of xenon could provide a delta-V of about 6.3 km/s in approximately two months. With such accelerations, radiation was found to not be a significant effect for planetary missions starting from GTO due to the short transit times through the Earth's radiation belts.

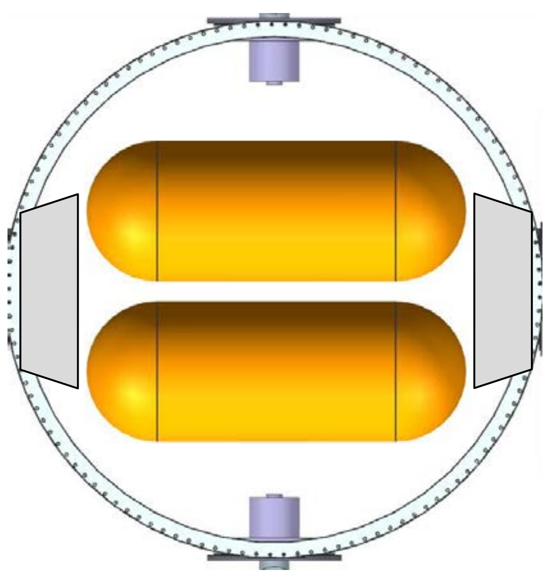

Fig. 4. Two-tank configuration for increasing the xenon storage capability from 220 kg to 440 kg inside a standard ESPA ring.

Table 4a. Option 1: 15-kW ROSA

	CBE Mass (kg)	Contin gency (%)	Total Mass (kg)	Heritage/Comments
Payload	7.0	2%	7.1	Cameras
C&DH	5.6	16%	6.5	BRE Integrated Avionics
Power	108.8	28%	139	15 kW ROSA Solar Arrays
Telecom	3.2	15%	3.6	L3 OTS S-band
Structures	264.4	19%	314	Includes ESPA ring,
Thermal	23.6	24%	29.3	Heaters, MLI, thermostats
Propulsion	87.4	21%	106	EP and RCS
GN&C	20.0	4%	20.8	LN200, AASTR, GDE
Spacecraft Total	**520.1**	**21%**	**627**	
System Margin			116.7	
Dry Mass Total		**43%**	**744**	43% on all but ESPA (8%)
Xenon Propellant			220.0	
RCS Propellant			26.0	
Wet Mass Total			**990**	

Table 4b. Option 2: 30-kW MegaFlex

	CBE Mass (kg)	Contin gency (%)	Total Mass (kg)	Heritage/Comments
Payload	7.0	0.0	7.1	Cameras
C&DH	5.6	16%	6.5	BRE Integrated Avionics
Power	215.8	29%	278	Solar Arrays + ABSL battery
Telecom	3.2	15%	3.6	L3 OTS S-band
Structures	337.1	18%	399	Includes ESPA ring,
Thermal	28.9	24%	35.8	Heaters, MLI, thermostats
Propulsion	166.8	25%	208	EP and RCS
GN&C	20.0	4%	20.8	LN200, AASTR, GDE
Spacecraft Total	**784.5**	**22%**	**960**	
System Margin			161.9	
Dry Mass Total		**43%**	**1122**	
Xenon Propellant			220.0	
RCS Propellant			26.0	
Wet Mass Total			**1368**	

Table 4c. Option 3: 30-kW MegaFlex Sized Array Depopulated to 15 kW

	CBE Mass (kg)	Contin gency (%)	Total Mass (kg)	Heritage/Comments
Payload	7.0	2%	7.1	Cameras
C&DH	5.6	16%	6.5	BRE Integrated Avionics
Power	215.8	29%	278	Solar Arrays + ABSL battery
Telecom	3.2	15%	3.6	L3 OTS S-band
Structures	319.5	19%	380	Includes ESPA ring,
Thermal	26.2	24%	32.4	Heaters, MLI, thermostats
Propulsion	87.4	21%	106	EP and RCS
GN&C	20.0	4%	20.8	LN200, AASTR, GDE
Spacecraft Total	**684.7**	**22%**	**835**	
System Margin			144.4	
Dry Mass Total		**43%**	**979**	43% on all but ESPA (8%)
Xenon Propellant			220.0	
RCS Propellant			26.0	
Wet Mass Total			**1225**	

Cost Estimate Summary

Total costs for a class-D, single-string, high-power, SEP technology demonstration mission, based on the three options identified in Table 3, were estimated using the following four independent cost models:

1. JPL's Team-X cost models.
2. The NASA Air Force Cost Model (NAFCOM).
3. The Small Satellite Cost Model (SCCM) from The Aerospace Corporation.
4. An Independent Cost Estimate from NASA Glenn Research Center (GRC).

All cost estimates were done in FY13 dollars. The NAFCOM and SSCM estimates were derived using the specified cost models supplemented with historical wrap-factors. Since the NAFCOM and SSCM estimates encompass only the period from the beginning of Phase C/D through delivery of the flight unit, estimates for Phase A/B were assumed to be 15% of the NAFCOM and SSCM Phase C/D cost output. Cost reserves of 30% on Phase A-D, and 15% for Phase E were included in all estimates.

There was remarkably good agreement among the different cost models, with all estimates falling within +20%/-10% of the Team X estimate. The largest scatter was for Option 3. This is possibly the result of the uncertainty of how to cost a physically large, but partially-populated solar array. The resulting cost estimates are plotted in Fig. 5 against the maximum expected dry mass (current best estimate plus mass growth contingency) of the vehicle.

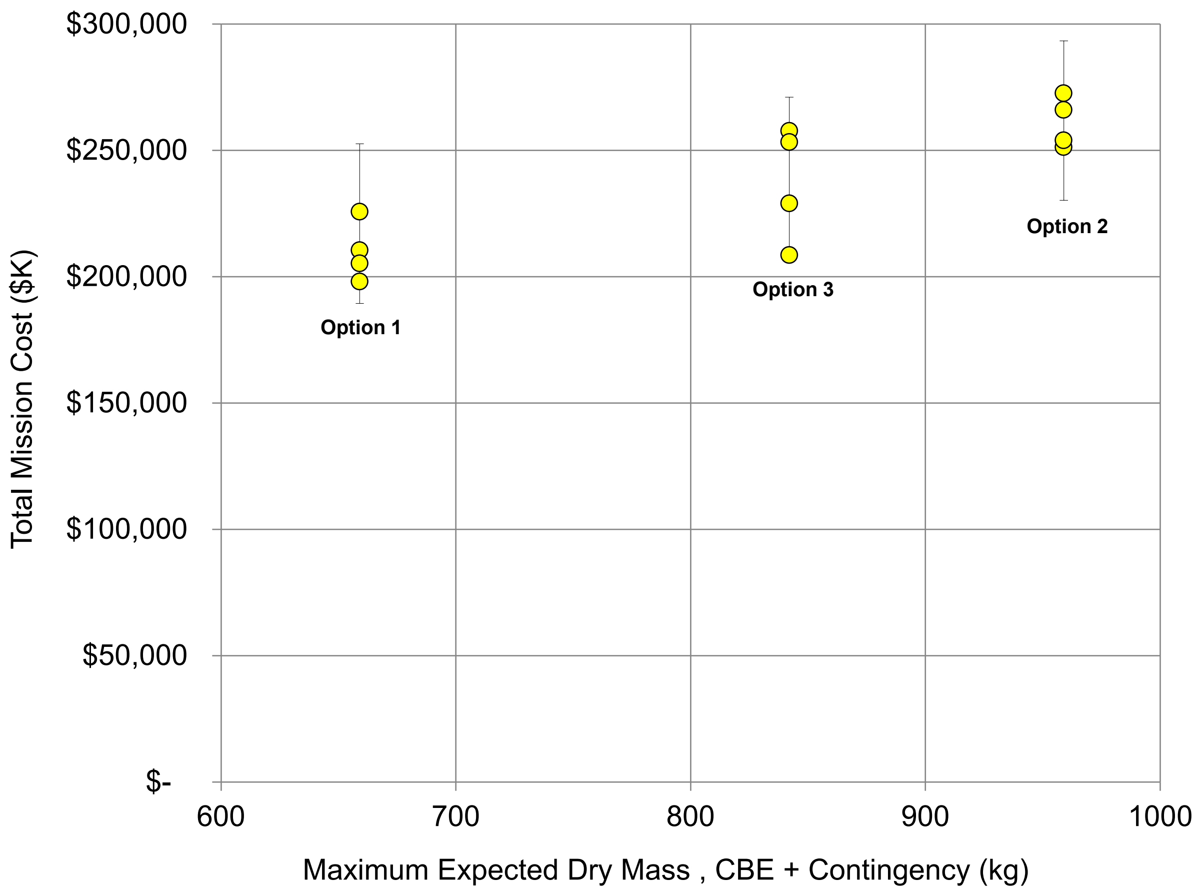

Figure. 5. Estimated total mission costs (Phases A-E) in FY'13$ for a class D, single-string, technology demonstration mission based on the options listed in Table 3.

IV. Conclusion

A meaningful, high-power SEP technology demonstration mission could be implemented for a total cost in the range $200M to $250M based on four independent cost estimates of an ESPA-ring-based high-power SEP vehicle with power levels up to 30 kW. The SEP TDM vehicle is designed around the ESPA ring as the primary structure enabling a low-cost ride into space as a Secondary Payload on an EELV. Incredibly, it was found to be possible to physically fit a 30-kW solar array in the form of two 15-kW MegaFlex wings onto a standard ESPA ring. This configuration, however, resulted in violations of the top and bottom separation planes, but this was not considered to be a significant problem. Configurations based on the MegaROSA solar array could easily package a 15-kW array (two 7.5-kW wings) on a standard ESPA ring in a 4-m launch vehicle fairing with no separation plane violations. A SEP TDM based on this approach would mitigate the key risks associated with the development of high power SEP vehicles that can only be satisfactorily demonstrated in space. It would demonstrate the deployment and operation of a large, light-weight, flexible blanket solar array, and is consistent with either SAS technology. It would demonstrate operation of a high-voltage (~300-V) solar array in the plasma environment produced by a Hall thruster-based electric propulsion system. Finally, it would demonstrate direct-drive operation of a multi-thruster electric propulsion subsystem.

A 30-kW, ESPA-based SEP vehicle with direct-drive has fantastic performance. It would have roughly 15x the initial acceleration of the Dawn spacecraft. It could provide approximately 3.1 km/s of ΔV in about one month, and ~6.3 km/s ΔV in about two months. The 15-kW ROSA version with 440 kg of xenon could provide up to about 8.5 km/s ΔV in approximately six months. The rapid spiral out from a GTO to Earth escape enabled by this system would minimize the effect of radiation damage on the system from the Earth's radiation belts. Higher ΔVs could be provided with configurations that enable the storage of more xenon (through the use of the ESPA Grande ring for example). Deep space missions using this approach would have to replace the S-band telecommunication system with an appropriate deep-space system.

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