IAC-04-R.1.06

A HIGH-ENERGY TECHNOLOGY DEMONSTRATION PLATFORM: THE FIRST STEP IN A STEPPING STONES APPROACH TO ENERGY-RICH SPACE INFRASTRUCTURES

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ABSTRACT

The sun provides an abundant source of energy in space, which can be used to power exploration vehicles and infrastructures that support exploration. A first step in developing and demonstrating the necessary technologies to support solar-powered exploration could be a 100-kWe-class solar-powered platform in Earth orbit. This platform would utilize advanced technologies in solar power collection and generation, power management and distribution, thermal management, and electric propulsion. It would also provide a power-rich free-flying platform to demonstrate in space a portfolio of technology flight experiments. This paper presents a preliminary design concept for a 100-kWe solar-powered satellite with the capability to use high-powered electric propulsion, and to flight-demonstrate a variety of payload experiments.

BACKGROUND

Space solar power (SSP) studies by NASA have taken a stepping stones approach to develop the gigawatt systems necessary for cost-effective delivery of power from space.[1] These steps start with a 100 kW-class satellite, leading to a 500 kW and then a 1 MW-class platform. Later steps develop a 100 MW bus that could eventually lead to a 1-2 GW pilot plant for SSP. Studies have shown that a modular approach is cost effective.[2] Modular designs include individual laser-power-beaming satellites that fly in constellations, or that are autonomously assembled into larger structures at geosynchronous orbit (GEO). A cost-effective approach to launching these modular units is to use existing Earth-to-orbit (ETO) launch systems, in which the modules are dropped into low Earth orbit (LEO) and then the modules perform their own orbit transfer to GEO using expendable solar arrays to power solar electric thrusters. At GEO, the modules either rendezvous and are assembled robotically into larger platforms, or are deployed into constellations of identical laser power-beaming satellites.

Exploration scenarios can also employ modular solar-powered units, which self-assemble into either solar electric propulsion (SEP) transport vehicles or energy-rich infrastructure platforms. Both SSP and solar-powered exploration vehicles need similar technology developments in solar power generation (SPG), power management and distribution (PMAD), modular deployable structures, and lightweight thermal management systems.

ENABLING TECHNOLOGIES FOR SPACE EXPLORATION

In expressing his vision for U.S. space exploration, President Bush emphasized affordable human and robotic programs, and the need to develop and demonstrate innovative technologies, including power generation and propulsion. Mr. Sean O'Keefe, NASA administrator, re-iterated the need to develop new capabilities in power and propulsion, and to demonstrate these technologies both on the ground and in Earth orbit. Finally, the Packard Commission, directed by the Department of Defense to recommend methods for reducing program cost overruns, stated that technology development needs to focus on lowering the cost of the system, and not just increasing performance, and that technologies should be sufficiently matured prior to implementation in systems. One of the key threads from these various sources is that "advanced power and propulsion technologies should be developed and demonstrated."

Solar electric propulsion (SEP) offers multiple responses to these national calls for power and propulsion technology development. Solar power collection utilizes freely available solar energy, provides the capability to operate in low Earth orbits (<1,000 km) where nuclear power would not be desirable, and has the flexibility to power any desired electric thruster concept. Although they have low thrust-to-weight ratios, electric thrusters couple a high specific impulse (an order of magnitude higher than chemical engines) with the capability to operate for lengthy periods of time, resulting in significant total impulse using substantially lower propellant quantities than current chemical engines. SEP provides the capability to support unmanned near-Earth and lunar transportation missions (for both equipment and propellant), exploration missions to the inner planets, and provide the capability to act as a test platform for nuclear electric propulsion technologies. The primary disadvantages of SEP are a low thrust-to-weight ratio and a fairly large solar array area, which results in increased atmospheric drag at the lower altitudes, potential sensitivity to acceleration, and launch vehicle packaging issues. Also, the reduced solar constant at the outer planets impacts the spacecraft power collection capability.

Current satellite power requirements range from less than a kilowatt up to 20 kilowatts. Looking into the future, space-based power needs are estimated to reach the 100-kilowatt range in the 2010 to 2012 timeframe. Fig. 1 shows that commercial satellite power requirements may exceed 40 kilowatts by 2010. In addition, space-based radars and lasers may require 40 to 125 kilowatts as soon as 2010. For exploration, electric propulsion, electromagnetic accelerators, and wireless power transfer technologies will require power levels of at least 100 kilowatts. A high-power technology demonstration spacecraft can be used to verify the non-nuclear technologies required to support these increasing satellite power requirements.

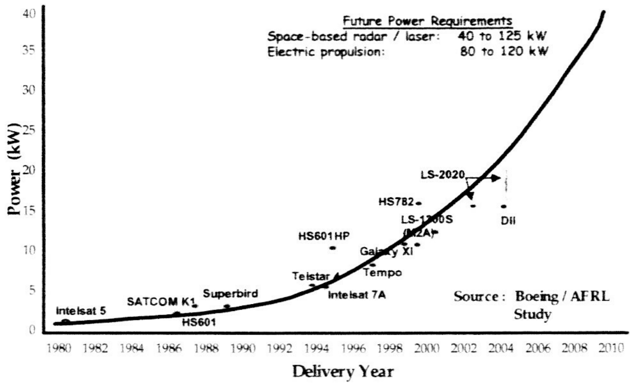

Fig.1: Spacecraft power requirements are accelerating.

PRELIMINARY DESIGN STUDY ASSUMPTIONS

A primary groundrule was the use of solar energy to generate 100 kilowatts (kWs) of electrical power under end-of-life conditions. Other groundrules included expendable launch vehicle (ELV) launch, a three-year lifetime, and a mission that orbits in low-Earth-orbit (LEO) for two years, followed by self-transport to mid-Earth-orbit (MEO) for a third year of experiment demonstration. Near-term technologies that could support a manufacturing start in the 2008 timeframe were also groundruled, in addition to a 2000kg payload capability for experiments. To avoid significant battery weight and solar array sizing penalties, high-power payload operation was limited to periods of insolation. A primary design driver was a high-voltage SPG and PMAD system that would enable "direct drive" of the high-voltage electric thrusters, and that would minimize heavy power converters at the arrays or the bus. The mission profile included a spacecraft transit of the Van Allen belts for verification of the high-voltage system in a high radiation environment. An existing spacecraft bus was selected to minimize development risk, and modular concepts were implemented where feasible.

SPACECRAFT TECHNOLOGIES

The 100 kW-class technology demonstrator will utilize advanced solar power collection and generation technologies, power management and distribution, advanced thermal management, and solar electric propulsion. State-of-the-art solar concentrators, highly efficient multi-junction solar cells, integrated thermal management on the arrays, and innovative deployable structure design and packaging make the 100 kW satellite feasible for launch on one existing launch vehicle. Early SSP studies showed that a major percentage of the on-orbit mass for power-beaming satellites was from massive power converters at the solar arrays, at the bus, at the power transmitter, or at combinations of these locations.[2]

Solar Power Generation

Based on a study completed by the US Department of Energy in early 2002, multi-junction solar cells not only have the highest efficiencies, but they also exhibit the highest levels of increased efficiencies in recent years. Some of the most promising are triple-junction cells that can be expected to exceed 30% efficiency by 2008. Quadruple-junction technology, if available in large quantities by 2008, will produce even higher efficiencies, in the mid-thirty percent range.

Solar concentrator technologies enable smaller solar arrays, which lower system masses and stowed volume requirements, and simplify deployment. In addition, solar concentrators increase performance and lower costs due to a reduction in the number of solar cells. This study has selected the ENTECH Stretched Lens Array (SLA), which has the advantage of passive thermal management and a solar concentration of approximately eight suns. A key characteristic of these solar concentrators is a solar cell efficiency increase of over 10%.

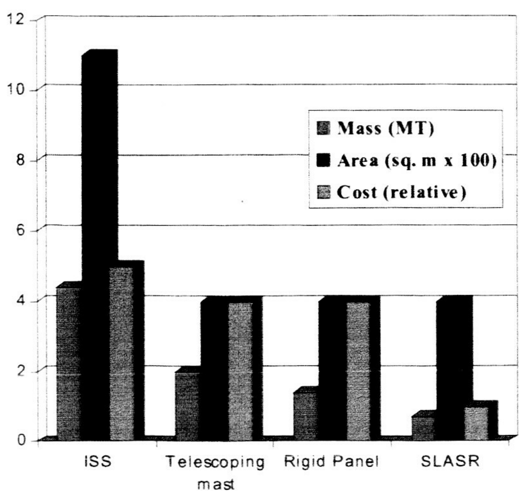

Fig. 2: Solar Panel Architecture Comparisons

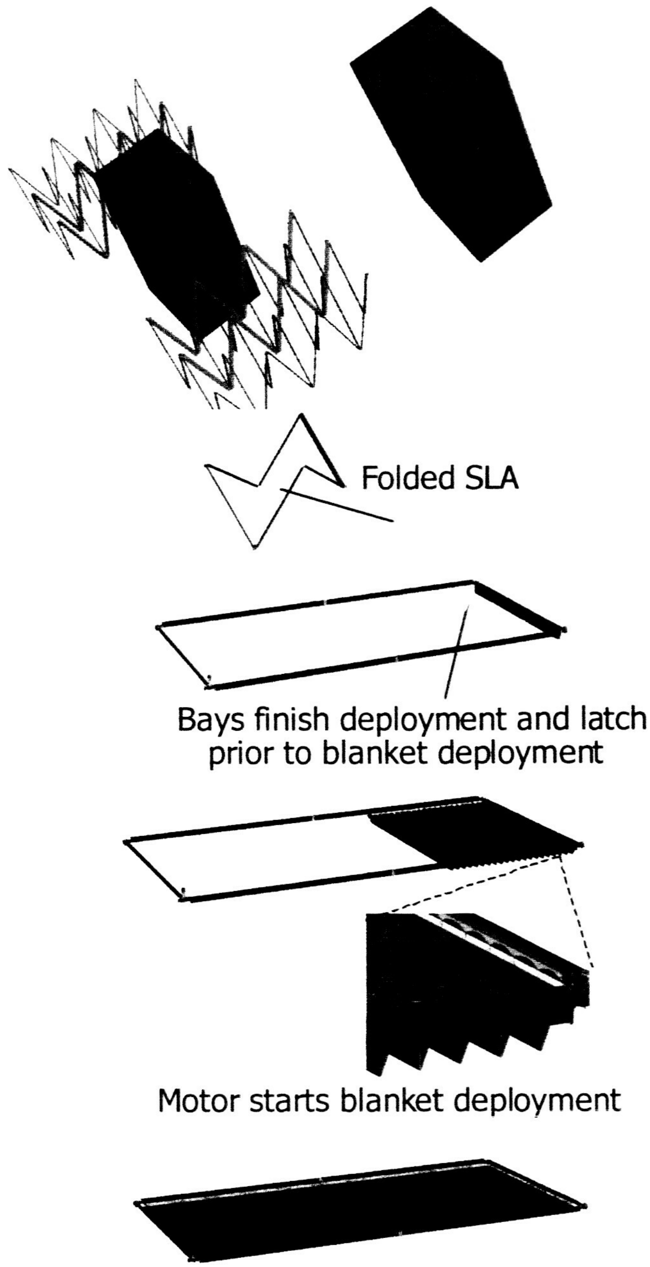

Fig. 3: SLASR Deployment

This study traded various solar panel architectures: rigid panels, concentrating photovoltaic systems, telescoping mast designs, and a square-rigger (SR) flexible blanket design. Fig. 2 provides a comparison of mass, area, and cost estimates for these solar array architectures. For reference, ISS solar array performance metrics are shown in the figures in the left bars. The telescoping mast is an ISS design with minor improvements. Cell efficiency estimates for the telescoping mast, rigid panel, and square-rigger are all based on a mid-thirty-percent, quadruple-junction cells. Mass estimates include PV assemblies, harnesses, structure and substrate, mechanisms, and tie downs. Area estimates are based on a 110 kW end-of-life design at LEO; the areas for the telescoping mast, rigid panel, and SLASR are similar due to the same cell efficiency and overall area required to capture the solar energy. The SLA with the square-rigger (SLASR) structure provides the lowest mass and lowest cost estimates, primarily because its surface is sparsely populated by solar cells. The square rigger system is based on a standard 2.5m x 5m bay, providing a modular unit that can be tailored to match satellite power requirements and solar array aspect ratios. One of the primary benefits of the square-rigger is the extremely efficient stowed packaging capability. Deployment is automated, with the bay structure being deployed first, followed by the solar-cell-and-stretched-lens assembly, as shown in Fig. 3. The SLASR team has plans to fabricate and test a full-scale array in the next few years.

Electric Propulsion

Electric thrusters are an ideal match for this solar-powered spacecraft design concept, since they provide fairly low thrust levels, which minimize solar array disturbances. Various near-term electric propulsion options have been examined, and the two primary candidates are Hall-effect thrusters and ion engines, based primarily on their relatively high technology readiness levels.

Table 1 provides a relative comparison of these two candidate engines. The Hall-effect thrusters have been selected for the technology demonstrator concept, based on their higher thrust levels, higher power levels, and lower costs. Two Hall thrusters can be mounted on the rear surface of the spacecraft to provide 100 kW propulsion capabilities. NASA Glenn Research Center testing has shown stable performance with power levels ranging from 9 to 72 kW, and with no evidence of thermal limitations.[3]

Power Management and Distribution (PMAD)

To achieve higher thrust and specific impulse performance from the Hall thrusters, voltage levels in the 500-600V ranges are needed. Two PMAD architectures have been evaluated. The first architecture baselines a 300V spacecraft bus, which would require power conditioning at the Hall thrusters to elevate the voltage. A second concept uses a 600V bus, which would enable "direct drive" capability for the Hall thrusters.

Both systems have been found to be technically feasible, although both use significantly higher voltages than that used on current spacecraft. Some effort will be required in the development of corona detection and mitigation, and the qualification of new hardware for the higher voltage levels. However, high-voltage electronics is currently at the TRL 5 level. The 600V bus was selected for this study concept; the 300V system could be an alternative should the high voltage system have problems during development.

	HALL-EFFECT THRUSTER	ION ENGINE
Candidate	NASA 457M	NASA Evolutionary Xenon Thruster (NEXT)
TRL (low power / high power)	9 / 5	9 / 6
Thrust Range	Up to 3,300 mN	Up to 210 mN
Unit Power Capability	Up to 73 kW	Up to 6 kW
Specific Impulse	3,200 sec	4,050 sec
Thruster Efficiency	62%	68%
Integration	2 thrusters for 100kW	16 thrusters for 100kW
Complexity	Simple with few moving parts	Sophisticated design with many components
Relative Cost	Low	High
Other	Shorter trip times; 4:1 throttle capability	More extensive US flight experience

Table 1: Near-Term Electric Propulsion Candidates.

Thermal Management

Large amounts of power on satellites imply that large amounts of waste heat will need to be managed. In order to shunt unused electric power from the solar arrays, a parasitic load radiator, capable of accommodating over 100kW, will be needed. However, the radiator can be designed to reject heat at significantly higher temperatures than typical spacecraft radiators. An additional thermal management system will handle the typical temperature ranges from PMAD components and other spacecraft electronics. Integrated heat rejection techniques using loop heat pipes and advanced radiator designs are planned.

DESIGN REFERENCE MISSION

A design reference mission was defined to help size certain subsystems, determine propellant needs, and define subsystem requirements. The spacecraft launches from Kennedy Space Center (KSC) on an ELV into a 407km, 51.6° inclination orbit. The spacecraft maintains this low-Earth orbit for two years to enable ISS interactions, baseline spacecraft performance capabilities, and allow primary payload operation. After two years, the spacecraft would use its Hall thrusters to raise its orbit to approximately 15,000km. The slow transition through the inner Van Allen belt would verify high-power, high-voltage spacecraft operation in this high radiation environment. The spacecraft would maintain this mid-Earth orbit for approximately six months for payload operation and to compare spacecraft performance against the LEO baseline performance. The spacecraft would then spiral out to Earth escape to complete the mission, transitioning through the outer Van Allen belt.

SPACECRAFT BUS

An existing flight-proven satellite bus has been selected for this concept study, to leverage existing structure and subsystems.

Solar Array Sizing

To size the solar arrays for radiation degradation, available data for triple-junction cells was used to estimate the temperature-based efficiencies of quadruple-junction cells in the MEO environment. For the triple-junction cells, over 340 W/m^2 will be available under end-of-life conditions. In order to provide ~100 kW at EOL, approximately 300 square meters of solar array area is required.

This mission requires additional solar cell shielding due to the high-voltage application and the radiation environment of the Van Allen belts. The mass of this additional insulation is partially offset by the low-mass SLA concentrator technology and its decreased solar cell count. For a 100kW delivery requirement, the mass of a planar array would be approximately triple the mass of an SLA system.

Two 150 m^2 solar arrays, each with twelve bays, is shown on the technology demonstrator concept in Fig. 4.

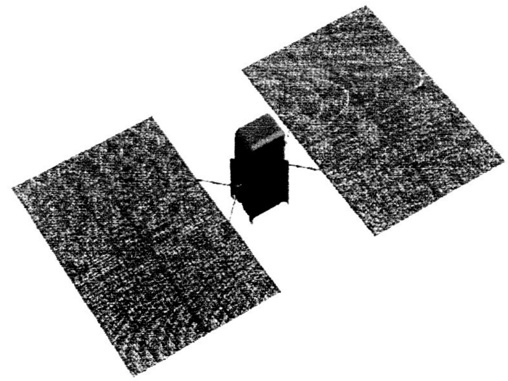

Fig. 4: Technology Demonstrator Concept with Deployed Solar Arrays

Electric Propulsion System Accommodation

The spacecraft bus originally included a chemical main engine and bi-propellant tanks, which have been removed in the concept and replaced by four 50kW Hall thrusters and pressurized Xenon tanks. After trades, thruster gimbaling has been selected for thrust vector control. The single mounting plate of the aft end of the spacecraft would provide two-axis gimbal capability.

Attitude Control System Assessment

In addition to existing reaction wheels and star trackers, the spacecraft bus attitude control system concept will add small gimbaled electric thrusters on the spacecraft sides for drag make-up and momentum management. Two-axis steering will be required for the large solar panels, and solar array pointing on the order of one degree will be required during both normal operations and orbit adjust maneuvers.

Figs. 5 and 6 show the spacecraft orbit attitudes. Fig. 5 shows the operational attitude, in which the payload is Earth-oriented in an local vertical, local horizontal attitude. Fig. 6 is the orbit adjust attitude, in which the spacecraft has been yawed ninety degrees to align the Hall thrusters with the velocity vector. The spacecraft is steered using thrust vectoring, while the solar arrays remain inertially fixed with respect to the sun.

Due to the addition of larger solar arrays, the modified spacecraft moments and products of inertia have been estimated, using masses from the original bus. Analysis has shown that the inertias and flexible body modes for the modified spacecraft are close enough to the heritage spacecraft to provide confidence in the use of existing control bandwidths. Transient responses to Hall-thruster impulses have been modeled, using a total thrust at 33% higher than anticipated thrusts, and a one-inch center-of-gravity offset. Maximum responses of the solar array to thruster impulses are significantly less than the one-degree requirement. Based on these top-level assessments, the modified attitude control system meets the requirements for the technology demonstrator with the SLASR arrays.

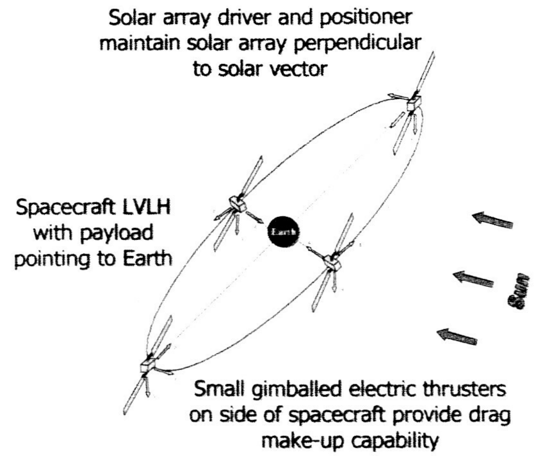

Fig. 5: On-Orbit Attitude

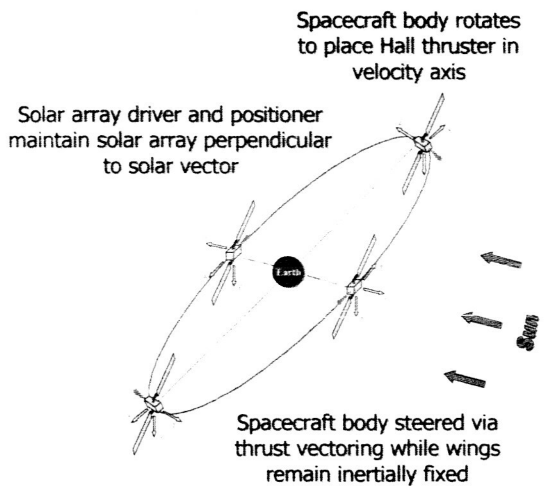

Fig. 6: Orbit-Raising Attitude

PMAD Assessment

The spacecraft concept will use high-voltage SLASR arrays and a 600V main power system to provide "direct drive" capability to the Hall thrusters. However, the spacecraft subsystems require 100Vdc. Existing batteries will be replaced with lithium ion batteries. A top-level sizing analysis of the PMAD system has been done, resulting in approximately 1000 kg mass increase and a volume increase of 0.9 m^3. In addition, significant electrical losses associated with this high-power, high-voltage architecture provide large heat loads that must be accommodated. Depending on the primary payload needs, a series boost converter could be eliminated from this architecture, reducing the mass by approximately 60 kg, and reducing the heat load by 1.7kW.

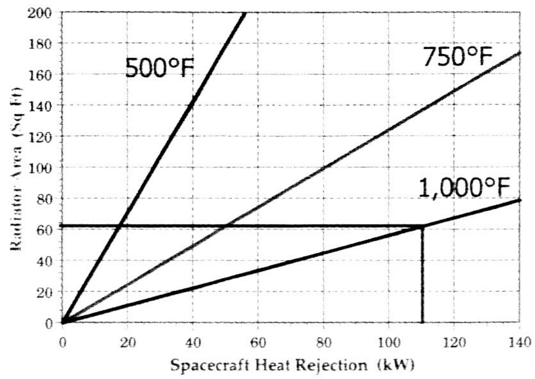

Fig. 7: Parasitic Load Radiator Sizing

Thermal Radiator Sizing

The current spacecraft design has extensive external exposed area for larger radiators. One of the primary impacts of the high-power architecture is the addition of a high-temperature parasitic load radiator, to shunt unused solar array power that could be as high as 110kW. As shown in Fig. 7, significant area improvements are obtained by operating the radiator at 1000°F. An additional benefit of such a high-temperature radiator is that this panel can be located on the sun-facing side of the spacecraft. The parasitic load radiator area has been estimated at 63 ft^2; an illustration of this radiator on the bus is shown in Fig. 8.

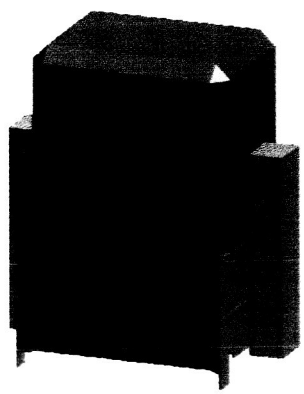

Fig. 8: Parasitic Load Radiator on Spacecraft Bus

In addition to the parasitic load radiator, additional radiators will be necessary to accommodate the increased thermal load from the high-power

PMAD components. These estimates range from 5 to almost 7kW; the elimination of the series boost converter could save 1.7kW. A fixed radiator on the spacecraft face opposite the parasitic load radiator could accommodate 1.5kW, requiring the addition of deployable radiators mounted at the spacecraft corners. These double-sided deployable radiators have been sized at 60 ft^2 each.

Trajectory Analysis and Propellant Needs

Trajectory analyses have determined propellant quantities and trip times required to complete the planned mission profile. The burn from the initial low-Earth orbit at ISS altitudes to mid-Earth orbit (MEO) at 15,000 km requires 85 days and 900 kg of propellant. The final orbit-adjust from MEO to Earth escape (906,378 km) requires 50 days and 750 kg of propellant.

Spacecraft Mass Summary

The spacecraft mass is estimated in Table 2. The total liftoff mass for the technology demonstrator, including payload, is 7000 kg.

PRIMARY PAYLOAD OPTIONS

Several high-power technology experiments would be suitable as a primary or secondary payload for the high-power technology demonstration spacecraft. These include advanced communications applications, wireless power transmission, space-based radar or laser applications, energy storage, and on-orbit propellant production and storage. Fig. 9 depicts some of these experiment payloads.

SUBSYSTEM	MASS (KG)
C&DH / ACS	120
Electric Propulsion	510
Power (Solar Panels)	400
Power (Electronics)	875
Power (Batteries)	100
Structures / Thermal	525
Wiring	135
Payload	2,150
Propellant	1,920
LV Adapter	115
Margin	150
Total	**7,000**

Table 2: Technology Demonstrator Mass Estimates

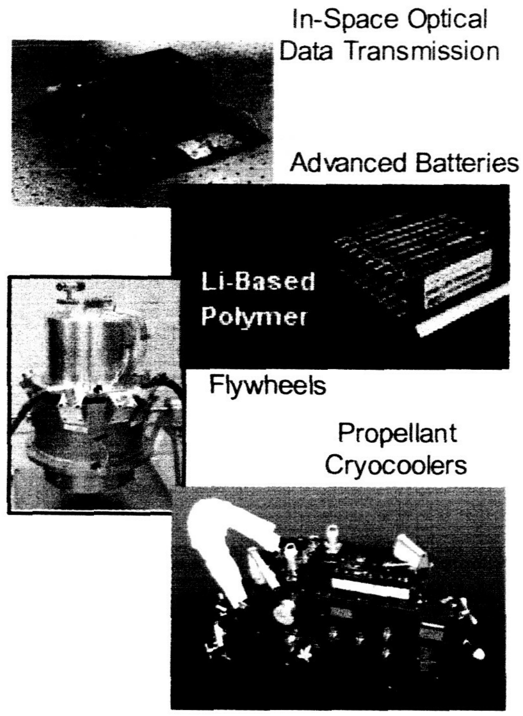

Fig. 9: Several technology experiment payload candidates

Advanced Communications

Both optical and radio-frequency (RF) communications experiments can be mounted as payloads on this satellite. Advanced communications transmitters can also be flown as secondary experiments to provide high bandwidth communications for the primary payload. There are several laser communications efforts in progress (NASA, ESA, and AFRL). ESA flew SILEX in 1998 on SPOT 4, which provided data rates of 50 Mbps. New systems being designed by JPL have significantly higher data rates and will require flight-demonstration in the 2007 timeframe. Although the input power requirements are fairly low, higher power transmitters would allow larger ground footprints, which would relax pointing requirements. However, high-power transmitters require development and qualification. On the RF side, again there are multiple advanced RF communications efforts in progress by NASA, ESA, and AFRL. Goddard Space Flight Center recently completed a spacecraft crosslink study that concluded Ka-band (22-27.5 Ghz) is the preferred band for intersatellite communications. In addition, the latest three TDRSS satellites incorporated Ka-band for high data rates in the 800 Mbps range. Power requirements for TV downlink are typically less than 10 kilowatts, but higher power transmitters would allow smaller and more compact ground-based receivers. As with higher-power laser transmitters, these RF transmitters will require development and qualification.

Wireless Power Transmission

Placing this high-power technology demonstration satellite in ISS orbit provides the opportunity to interact with ISS. A wireless power transmission (WPT) experiment would be ideally suited for this application. A laser unit could be mounted on the Exposed Facility of the Japanese Experiment Module (JEM-EF), and power can be optically transmitted to the technology demonstration spacecraft. Infrared sensors on the ISS-based laser module would image the heated edge of the satellite photovoltaic array and aid pointing to this target. An additional WPT experiment could be mounted on the spacecraft primary deck, and could demonstrate space-to-ground WPT capability. This technology experiment would provide key demonstrations of space-to-space and space-to-ground technologies for Space Solar Power.

Space-based Radar Applications

Space-based radars would be suitable as primary payloads for the high-power technology demonstration spacecraft. A 100-kilowatt radar system would significantly increase surveillance rates and target resolution. In addition, higher power permits a higher altitude orbit, which provides increased coverage, enhanced survivability, and lower constellation costs. Higher surveillance rates from this primary experiment would work well with the high bandwidth data links that could be provided by an advanced communication system experiment as a secondary payload.

Energy Storage

Advanced energy storage hardware can be demonstrated on this high-power demonstration spacecraft as a means of facilitating continuous payload operation. Significant levels of energy will need to be stored during periods of insolation to enable payload operation during the eclipse portion of the orbit. Two primary energy storage methods to be considered are batteries and flywheels.

Advanced Batteries

Advanced batteries offer increased energy densities (specific energy) and also permit greater depth of discharge. ISS is currently using nickel-hydrogen batteries, which have a specific energy of approximately 50 watts per kilogram and depths of discharge in the 35% range. Lithium ion and lithium polymer batteries have specific energies of 100 watts per kilogram and can accommodate a 90% depth of discharge. The proposed high-power spacecraft would provide an ideal technology platform for advanced energy storage hardware.

Flywheels

Advanced aerospace flywheels store energy more efficiently that rechargeable chemical batteries. In addition, they offer improvements in satellite attitude control over control moment gyros (CMGs) and reaction control wheels. Electricity powers a motor that spins a flywheel, and the flywheel stores energy mechanically, speeding up as it accumulates energy, and slowing down as it delivers energy to a load. The decrease in mechanical energy is converted back into electricity by a generator. The technology spacecraft would provide an ideal onboard energy storage testbed.

On-Orbit Propellant Production and Storage

This high-power technology testbed can provide a platform to produce and store cryogenic propellants. In addition, this platform can be used to demonstrate autonomous fluid transfer techniques. With the addition of energy, water can be broken down by electrolysis to its constituent elements, hydrogen and oxygen. The high-power platform provides an abundance of energy to accomplish electrolysis. Once the gaseous hydrogen and oxygen are formed, they can be cooled to cryogenic temperatures and stored as liquids. These liquids can then be maintained in cryogenic dewars demonstrating the latest advances in vapor-cooled shields. Long-term cryogen storage is an enabling technology for reusable in-space transportation systems. These demonstrations will allow assessments of the thermal leak paths and lead to improved dewar designs. In addition, critical demonstrations of autonomous cryogenic fluid transfer can be considered. Reusable in-space transportation systems will be heavily reliant on fuel depots to replenish their cryogenic propellants.

SUMMARY

A 100kW-class technology demonstrator concept has been developed in this study. Technical challenges include insulation on the high-voltage solar arrays, design and qualification of the high-voltage PMAD components, design of high-voltage power-conducting slip rings, and substantial radiators to accommodate large amounts of waste heat.

Preliminary design and top-level analyses have shown the feasibility of developing and building this spacecraft in the 2008 timeframe. Modular concepts have been presented for both power collection and PMAD to allow tailoring for specific mission requirements. A variety of experiment payload options that would benefit from this energy-rich technology demonstrator have been identified

ACKNOWLEGEMENTS

The results presented here are the compilation of multiple inputs from a variety of different organizations that supported the third author of this study.[4]. Robert Jankovsky and David Jacobson from NASA Glenn Research Center provided Hall thruster expertise. Steve Kusek from HiTek Services provided engineering support to solar collection analysis, and Mark O'Neill of ENTECH provided extensive support in solar power collection and concentration. Mike Eskenazi from Able Engineering provided support in the solar wing packaging solution. Paul Gill, Tom Lynch, Ken Metcalf, Jim Shoji, and Jim McClanahan from Boeing-Rocketdyne provided electric thruster information, PMAD architecture design, and trajectory analysis support. Denny Ertsman, Dr. Eric Jensen, Nigel Rowe, Mark Spiwak, Doug Bender, and Ketao Liu from Boeing Satellite Systems provided a wealth of satellite expertise in thermal design, attitude control, packaging, and payload integration. The Boeing Phantom Works in Huntsville tied the details together into a comprehensive final report.[4]

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