Solar Electric Propulsion Nasa

The success of the NASA Deep Space I spacecraft has demonstrated that ion propulsion is a viable option for deep space science missions. More aggressive missions such as Comet Nuclear Sample Return and Europa lander will require higher power, higher propellant throughput and longer thruster lifetime than the NASA Solar Electric Propulsion Technology Application Readiness (NSTAR) engine. Presented here are thruster plume and discharge plasma measurements of an NSTAR-type thruster operated from 0.5 kW to 5 kW. From Faraday plume sweeps, beam divergence was determined. From Langmuir probe plume measurements on centerline, low energy ion production on axis due to charge-exchange and direct ionization was assessed. Additionally, plume plasma potential measurements made on axis were used to determine the upper energy limits at which ions created on centerline could be radially accelerated. Wall probes flush-mounted to the thruster discharge chamber anode were used to assess plasma conditions. Langmuir probe measurements at the wall indicated significant differences in the electron temperature in the cylindrical and conical sections of the discharge chamber. Foster, John E and Soulas, George C. and Patterson, Michael J. Glenn Research Center NASA/TM-2000-210382, E-12438, AIAA Paper 2000-3812, NAS 1.15:210382

Current technology for solar-electric propulsion is used to assess the potential performance advantages of low-thrust propulsion for an out-of-the-ecliptic mission. Simple normal-to-the-orbit thrust steering is assumed with coast subarcs permitted. The electric spacecraft is launched onto an Earth escape trajectory by an Atlas (SLV3C)-Centaur or a Titan IIIC. Comparisons with a similarly launched uprated Burner II stage reveal that significant performance gains are possible using the electric stage with 250- to 475-day flight times.

A preliminary investigation of a lunar-comet rendezvous mission using a solar electric propulsion (SEP) spacecraft was performed in two phases.The first phase involved exploration of the moon and the second involved rendezvous with a comet. The initial phase began with a chemical propulsion translunar injection and chemical insertion into a lunar orbit, followed by a low thrust SEP transfer to a circular, polar, low-lunar orbit. After collecting scientific data at the moon, the SEP spacecraft performed a spiral lunar escape maneuver to begin the interplanetary leg of the mission. After escape from the Earth-moon system, the SEP spacecraft maneuvered in interplanetary space and performed a rendezvous with a comet.The immediate goal of this study was to demonstrate the feasibility of using a low-thrust SEP spacecraft for orbit transfer to both the moon and a comet. Another primary goal was to develop a computer optimization code which would be robust enough to obtain minimum-fuel rendezvous trajectories for a wide range of comets. Kluever, Craig A. Unspecified Center NAG3-1581...

After the completion of the National Research Council (NRC) report, Maintaining U.S. Leadership in Aeronautics: Scenario-Based Strategic Planning for NASA's Aeronautics Enterprise (1997), the National Aeronautics and Space Administration (NASA) Office of Aeronautics and Space Transportation Technology requested that the NRC remain involved in its strategic planning process by conducting a study to identify a short list of revolutionary or breakthrough technologies that could be critical to the 20 to 25 year future of aeronautics and space transportation. These technologies were to address the areas of need and opportunity identified in the above mentioned NRC report, which have been characterized by NASA's 10 goals (see Box ES-1) in "Aeronautics & Space Transportation Technology: Three Pillars for Success" (NASA, 1997). The present study would also examine the 10 goals to determine if they are likely to be achievable, either through evolutionary steps in technology or through the identification and application of breakthrough ideas, concepts, and technologies.

This presentation reviews Solar Electric Propulsion (SEP) Mission Architectures with a slant towards power system technologies and challenges. The low-mass, high-performance attributes of SEP systems have attracted spacecraft designers and mission planners alike and have led to a myriad of proposed Earth orbiting and planetary exploration missions. These SEP missions are discussed from the earliest missions in the 1960's, to first demonstrate electric thrusters, to the multi-megawatt missions envisioned many decades hence. The technical challenges and benefits of applying high-voltage arrays, thin film and low-intensity, low-temperature (LILT) photovoltaics, gossamer structure solar arrays, thruster articulating systems and microsat systems to SEP spacecraft power system designs are addressed. The overarching conclusion from this review is that SEP systems enhance, and many times enable, a wide class of space missions. Kerslake, Thomas W. Glenn Research Center NASA/TM-2003-212456, NAS 1.15:212456, E-13995

Human exploration beyond low Earth orbit will require the use of enabling technologies that are efficient, affordable, and reliable. Solar electric propulsion (SEP) has been proposed by NASA s Human Exploration Framework Team as an option to achieve human exploration missions to near Earth objects (NEOs) because of its favorable mass efficiency as compared to traditional chemical systems. This paper describes the unique challenges and technology hurdles associated with developing a large high-power SEP vehicle. A subsystem level breakdown of factors contributing to the feasibility of SEP as a platform for future exploration missions to NEOs is presented including overall mission feasibility, trip time variables, propellant management issues, solar array power generation, array structure issues, and other areas that warrant investment in additional technology or engineering development.

This paper presents study results quantifying the benefits of higher voltage, electric power system designs for a typical solar electric propulsion spacecraft Earth orbiting mission. A conceptual power system architecture was defined and design points were generated for system voltages of 28-V, 50-V, 120-V, and 300-V using state-of-the-art or advanced technologies. A 300-V 'direct-drive' architecture was also analyzed to assess the benefits of directly powering the electric thruster from the photovoltaic array without up-conversion. Fortran and spreadsheet computational models were exercised to predict the performance and size power system components to meet spacecraft mission requirements. Pertinent space environments, such as electron and proton radiation, were calculated along the spiral trajectory. In addition, a simplified electron current collection model was developed to estimate photovoltaic array losses for the orbital plasma environment and that created by the thruster plume. The secondary benefits of power system mass savings for spacecraft propulsion and attitude control systems were also quantified. Results indicate that considerable spacecraft wet mass savings were achieved by the 300-V and 300-V direct-drive architectures. Kerslake, Thomas W. Glenn Research Center NASA/TM-2003-212304, E-13876, NAS 1.15:212304

In this study, the Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) team completed a design for a multi-asteroid (Nereus and 1996 FG3) sample return capable spacecraft for the NASA In-Space Propulsion Office. The objective of the study was to support technology development and assess the relative benefits of different electric propulsion systems on asteroid sample return design. The design uses a single, heritage Orion solar array (SA) (approx.6.5 kW at 1 AU) to power a single NASA Evolutionary Xenon Thruster ((NEXT) a spare NEXT is carried) to propel a lander to two near Earth asteroids. After landing and gathering science samples, the Solar Electric Propulsion (SEP) vehicle spirals back to Earth where it drops off the first sample s return capsule and performs an Earth flyby to assist the craft in rendezvousing with a second asteroid, which is then sampled. The second sample is returned in a similar fashion. The vehicle, dubbed Near Earth Asteroids Rendezvous and Sample Earth Returns (NEARER), easily fits in an Atlas 401 launcher and its cost estimates put the mission in the New Frontier s (NF's) class mission. Oleson, Steven R. and McGuire, Melissa L. Glenn Research Center ASTEROIDS; SOLAR ELECTRIC PROPULSION; SAMPLE RETURN MISSIONS; PROPULSION SYSTEM CONFIGURATIONS; SYSTEMS ENGINEERING; FLYBY MISSIONS; PROPULSION SYSTEM PERFORMANCE; ELECTRIC PROPULSION; COST ESTIMATES; SUPPORT SYSTEMS; SOLAR ARRAYS

This paper presents several mission designs for heliospheric boundary exploration using spacecraft with low-thrust ion engines as the primary mode of propulsion The mission design goal is to transfer a 200-kg spacecraft to the heliospheric boundary in minimum time. The mission design is a combined trajectory and propulsion system optimization problem. Trajectory design variables include launch date, launch energy, burn and coast arc switch times, thrust steering direction, and planetary flyby conditions. Propulsion system design parameters include input power and specific impulse. Both SEP and NEP spacecraft arc considered and a wide range of launch vehicle options are investigated. Numerical results are presented and comparisons with the all chemical heliospheric missions from Ref 9 are made. Kluever, Craig A. Glenn Research Center NAG3-1731...

Solar Electric Propulsion (SEP) when used for station keeping and final orbit insertion has been shown to increase a geostationary satellite's payload when launched by existing expendable launch vehicles. In the case of reusable launch vehicles or expendable launch vehicles where an upper stage is an expensive option, this methodology can be modified by using the existing on-board apogee chemical system to perform a perigee burn and then letting the electric propulsion system complete the transfer to geostationary orbit. The elimination of upper stages using on-board chemical and electric propulsion systems was thus examined for GEO spacecraft. Launch vehicle step-down from an Atlas IIAR to a Delta 7920 (no upper stage) was achieved using expanded on-board chemical tanks, 40 kW payload power for electric propulsion, and a 60 day elliptical to GEO SEP orbit insertion. Optimal combined chemical and electric trajectories were found using SEPSPOT. While Hall and ion thrusters provided launch vehicle step-down and even more payload for longer insertion times, NH3 arcjets had insufficient performance to allow launch vehicle step-down. Degradation levels were only 5% to 7% for launch step-down cases using advanced solar arrays. Results were parameterized to allow comparisons for future reusable launch vehicles. Results showed that for an 8 W/kg initial power/launch mass power density spacecraft, 50% to 100% more payload can be launched using this method. Oleson, Steven Glenn Research Center NASA/TM-1999-209646, NAS 1.15:209646, IEPC-99-185, E-11994

The sun tower concept of collecting solar energy in space and beaming it down for commercial use will require very affordable in-space as well as earth-to-orbit transportation. Advanced electric propulsion using a 200 kW power and propulsion system added to the sun tower nodes can provide a factor of two reduction in the required number of launch vehicles when compared to in-space cryogenic chemical systems. In addition, the total time required to launch and deliver the complete sun tower system is of the same order of magnitude using high power electric propulsion or cryogenic chemical propulsion: around one year. Advanced electric propulsion can also be used to minimize the stationkeeping propulsion system mass for this unique space platform. 50 to 100 kW class Hall, ion, magnetoplasmadynamic, and pulsed inductive thrusters are compared. High power Hall thruster technology provides the best mix of launches saved and shortest ground to Geosynchronous Earth Orbital Environment (GEO) delivery time of all the systems, including chemical. More detailed studies comparing launch vehicle costs, transfer operations costs, and propulsion system costs and complexities must be made to down-select a technology. The concept of adding electric propulsion to the sun tower nodes was compared to a concept using re-useable electric propulsion tugs for Low Earth Orbital Environment (LEO) to GEO transfer. While the tug concept would reduce the total number of required propulsion systems, more launchers and notably longer LEO to GEO and complete sun tower ground to GEO times would be required. The tugs would also need more complex, longer life propulsion systems and the ability to dock with sun tower nodes. Oleson, Steve Glenn Research Center NASA/TM-1999-209307, E-11833, NAS 1.15:209307, AIAA Paper 99-2872

The NASA Office of the Chief Technologist Game Changing Division is sponsoring the development and testing of enabling technologies to achieve efficient and reliable human space exploration. High-power solar electric propulsion has been proposed by NASA's Human Exploration Framework Team as an option to achieve these ambitious missions to near Earth objects. NASA Glenn Research Center (NASA Glenn) is leading the development of mission concepts for a solar electric propulsion Technical Demonstration Mission. The mission concepts are highlighted in this paper but are detailed in a companion paper. There are also multiple projects that are developing technologies to support a demonstration mission and are also extensible to NASA's goals of human space exploration. Specifically, the In-Space Propulsion technology development project at NASA Glenn has a number of tasks related to high-power Hall thrusters including performance evaluation of existing Hall thrusters; performing detailed internal discharge chamber, near-field, and far-field plasma measurements; performing detailed physics-based modeling with the NASA Jet Propulsion Laboratory's Hall2De code; performing thermal and structural modeling; and developing high-power efficient discharge modules for power processing. This paper summarizes the various technology development tasks and progress made to date Kamhawi, Hani and Manzella, David H. and Smith, Timothy D. and Schmidt, George R. Glenn Research Center WBS 182603.01.04.02

The results of the NASA Glenn Research Center (GRC) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) internal Solar Electric Propulsion (SEP) stage design are documented in this report (Figure 1.1). The SEP Stage was designed to deliver a science probe to Saturn (the probe design was performed separately by the NASA Goddard Space Flight Center s (GSFC) Integrated Mission Design Center (IMDC)). The SEP Stage delivers the 2444 kg probe on a Saturn trajectory with a hyperbolic arrival velocity of 5.4 km/s. The design carried 30 percent mass, 10 percent power, and 6 percent propellant margins. The SEP Stage relies on the probe for substantial guidance, navigation and control (GN&C), command and data handling (C&DH), and Communications functions. The stage is configured to carry the probe and to minimize the packaging interference between the probe and the stage. The propulsion system consisted of a 1+1 (one active, one spare) configuration of gimbaled 7 kW NASA Evolutionary Xenon Thruster (NEXT) ion propulsion thrusters with a throughput of 309 kg Xe propellant. Two 9350 W GaAs triple junction (at 1 Astronomical Unit (AU), includes 10 percent margin) ultra-flex solar arrays provided power to the stage, with Li-ion batteries for launch and contingency operations power. The base structure was an Al-Li hexagonal skin-stringer frame built to withstand launch loads. A passive thermal control system consisted of heat pipes to north and south radiator panels, multilayer insulation (MLI) and heaters for the Xe tank. All systems except tanks and solar arrays were designed to be single fault tolerant.

The NASA Solar Electric Propulsion Technology Applications Readiness Program (NSTAR) will provide a single-string primary propulsion system to NASA's New Millennium Deep Space 1 Mission which will perform comet and asteroid flybys in the years 1999 and 2000. The propulsion system includes a 30-cm diameter ion thruster, a xenon feed system, a power processing unit, and a digital control and interface unit. A total of four engineering model ion thrusters, three breadboard power processors, and a controller have been built, integrated, and tested. An extensive set of development tests has been completed along with thruster design verification tests of 2000 h and 1000 h. An 8000 h Life Demonstration Test is ongoing and has successfully demonstrated more than 6000 h of operation. In situ measurements of accelerator grid wear are consistent with grid lifetimes well in excess of the 12,000 h qualification test requirement. Flight hardware is now being assembled in preparation for integration, functional, and acceptance tests. Sovey, James S. and Hamley, John A. and Haag, Thomas W. and Patterson, Michael J. and Pencil, Eric J. and Peterson, Todd T. and Pinero, Luis R. and Power, John L. and Rawlin, Vincent K. and Sarmiento, Charles J. and Anderson, John R. and Bond, Thomas A. and Cardwell, G. I. and Christensen, Jon A. Glenn Research Center; Jet Propulsion Laboratory RTOP 242-70-01...

Human exploration beyond low Earth orbit will require the use of enabling technologies that are efficient, affordable, and reliable. Solar electric propulsion (SEP) has been proposed by NASA s Human Exploration Framework Team as an option to achieve human exploration missions to near Earth objects (NEOs) because of its favorable mass efficiency as compared to traditional chemical systems. This paper describes the unique challenges and technology hurdles associated with developing a large high-power SEP vehicle. A subsystem level breakdown of factors contributing to the feasibility of SEP as a platform for future exploration missions to NEOs is presented including overall mission feasibility, trip time variables, propellant management issues, solar array power generation, array structure issues, and other areas that warrant investment in additional technology or engineering development. Capadona, Lynn A. and Woytach, Jeffrey M. and Kerslake, Thomas W. and Manzella, David H. and Christie, Robert J. and Hickman, Tyler A. and Schneidegger, Robert J. and Hoffman, David J. and Klem, Mark D. Glenn Research Center NASA/TM-2012-217275, AIAA Paper-2011-7251, E-18030

Power limited, low-thrust trajectories were assessed for missions to Jupiter, Saturn, and Neptune utilizing a single Venus Gravity Assist (VGA) and a primary propulsion system based on either a 3-kW high voltage Hall thruster, of the type being developed by the NASA In-Space Propulsion Technology Program, or an 8-kW variant of this thruster. These Hall thrusters operate with specific impulses below 3,000 seconds. A trade study was conducted to examine mission parameters that include: net delivered mass (NDM), beginning-of-life (BOL) solar array power, heliocentric transfer time, required launch vehicle, number of operating thrusters, and throttle profile. The top performing spacecraft configuration was defined to be the one that delivered the highest mass for a range of transfer times. In order to evaluate the potential future benefit of using next generation Hall thrusters as the primary propulsion system, comparisons were made with the advanced state-of-the-art (ASOA), 7-kW, 4,100 second NASA's Evolutionary Xenon Thruster (NEXT) for the same mission scenarios. For the BOL array powers considered in this study (less than 30 kW), the results show that the performance of the Hall thrusters, relative to NEXT, is largely dependant on the performance capability of the launch vehicle, and that at least a 10 percent performance gain, equating to at least an additional 200 kg dry mass at each target planet, is achieved over the higher specific impulse NEXT when launched on an Atlas 551.Witzberger, Kevin E. and Manzella, DavidGlenn Research CenterHALL THRUSTERS; SOLAR ELECTRIC PROPULSION; DEEP SPACE 1 MISSION; SPACECRAFT CONFIGURATIONS; NASA SPACE PROGRAMS; SOLAR ARRAYS; POWER CONDITIONING; MATHEMATICAL MODELS; SPECIFIC IMPULSE; HIGH VOLTAGES; NEPTUNE (PLANET); SATURN (PLANET); LAUNCH VEHICLES

Throughout most of the twentieth century, electric propulsion was considered the technology of the future. Now, the future has arrived. This important new book explains the fundamentals of electric propulsion for spacecraft and describes in detail the physics and characteristics of the two major electric thrusters in use today, ion and Hall thrusters. The authors provide an introduction to plasma physics in order to allow readers to understand the models and derivations used in determining electric thruster performance. They then go on to present detailed explanations of: Thruster principles Ion thruster plasma generators and accelerator grids Hollow cathodes Hall thrusters Ion and Hall thruster plumes Flight ion and Hall thrusters Based largely on research and development performed at the Jet Propulsion Laboratory (JPL) and complemented with scores of tables, figures, homework problems, and references, Fundamentals of Electric Propulsion: Ion and Hall Thrusters is an indispensable textbook for advanced undergraduate and graduate students who are preparing to enter the aerospace industry. It also serves as an equally valuable resource for professional engineers already at work in the field.

Power limited, low-thrust trajectories were assessed for missions to Jupiter, Saturn, and Neptune utilizing a single Venus Gravity Assist (VGA) and a primary propulsion system based on either a 3-kW high voltage Hall thruster, of the type being developed by the NASA In-Space Propulsion Technology Program, or an 8-kW variant of this thruster. These Hall thrusters operate with specific impulses below 3,000 seconds. A trade study was conducted to examine mission parameters that include: net delivered mass (NDM), beginning-of-life (BOL) solar array power, heliocentric transfer time, required launch vehicle, number of operating thrusters, and throttle profile. The top performing spacecraft configuration was defined to be the one that delivered the highest mass for a range of transfer times. In order to evaluate the potential future benefit of using next generation Hall thrusters as the primary propulsion system, comparisons were made with the advanced state-of-the-art (ASOA), 7-kW, 4,100 second NASA's Evolutionary Xenon Thruster (NEXT) for the same mission scenarios. For the BOL array powers considered in this study (less than 30 kW), the results show that the performance of the Hall thrusters, relative to NEXT, is largely dependant on the performance capability of the launch vehicle, and that at least a 10 percent performance gain, equating to at least an additional 200 kg dry mass at each target planet, is achieved over the higher specific impulse NEXT when launched on an Atlas 551.

A design study for a cargo transfer vehicle using solar electric propulsion was performed for NASA's Revolutionary Aerospace Systems Concepts program. Targeted for 2016, the solar electric propulsion (SEP) transfer vehicle is required to deliver a propellant supply module with a mass of approximately 36 metric tons from Low Earth Orbit to the first Earth-Moon libration point (LL1) within 270 days. Following an examination of propulsion and power technology options, a SEP transfer vehicle design was selected that incorporated large-area (approx. 2700 sq m) thin film solar arrays and a clustered engine configuration of eight 50 kW gridded ion thrusters mounted on an articulated boom. Refinement of the SEP vehicle design was performed iteratively to properly estimate the required xenon propellant load for the out-bound orbit transfer. The SEP vehicle performance, including the xenon propellant estimation, was verified via the SNAP trajectory code. Further efforts are underway to extend this system model to other orbit transfer missions. Sarver-Verhey, Timothy R. and Kerslake, Thomas W. and Rawlin, Vincent K. and Falck, Robert D. and Dudzinski, Leonard J. and Oleson, Steven R. Glenn Research Center NASA/TM-2002-211970, E-13613, NAS 1.15:211970, AIAA Paper 2002-3971

NASA's Office of the Chief Technologist (OCT) has begun to rebuild the advanced space technology program in the agency with plans laid out in 14 draft technology roadmaps. It has been years since NASA has had a vigorous, broad-based program in advanced space technology development and its technology base has been largely depleted. However, success in executing future NASA space missions will depend on advanced technology developments that should already be underway. Reaching out to involve the external technical community, the National Research Council (NRC) considered the 14 draft technology roadmaps prepared by OCT and ranked the top technical challenges and highest priority technologies that NASA should emphasize in the next 5 years. This report provides specific guidance and recommendations on how the effectiveness of the technology development program managed by OCT can be enhanced in the face of scarce resources.

The NASA's Evolutionary Xenon Thruster (NEXT) project is developing the next-generation solar electric propulsion ion propulsion system with significant enhancements beyond the state-of-the-art NASA Solar Electric Propulsion Technology Application Readiness (NSTAR) ion propulsion system in order to provide future NASA science missions with enhanced propulsion capabilities. As part of a comprehensive thruster service life assessment, the NEXT Long-Duration Test (LDT) was initiated in June 2005 to demonstrate throughput capability and validate thruster service life modeling. The NEXT LDT exceeded its original qualification throughput requirement of 450 kg in December 2009. To date, the NEXT LDT has set records for electric propulsion lifetime and has demonstrated 50,170 h of operation, processed 902 kg of propellant, and delivered 34.9 MN-s of total impulse. The NEXT thruster design mitigated several life-limiting mechanisms encountered in the NSTAR design, dramatically increasing service life capability. Various component erosion rates compare favorably to the pretest predictions based upon semi-empirical ion thruster models. The NEXT LDT either met or exceeded all of its original goals regarding lifetime demonstration, performance and wear characterization, and modeling validation. In light of recent budget constraints and to focus on development of other components of the NEXT ion propulsion system, a voluntary termination procedure for the NEXT LDT began in April 2013. As part of this termination procedure, a comprehensive post-test performance characterization was conducted across all operating conditions of the NEXT throttle table. These measurements were found to be consistent with prior data that show minimal degradation of performance over the thruster's 50 kh lifetime. Repair of various diagnostics within the test facility is presently planned while keeping the thruster under high vacuum conditions. These diagnostics will provide additional critical informat

NASA's Evolutionary Xenon Thruster (NEXT) is a next-generation high-power ion propulsion system under development by NASA as a part of the In-Space Propulsion Technology Program. NEXT is designed for use on robotic exploration missions of the solar system using solar electric power. Potential mission destinations that could benefit from a NEXT Solar Electric Propulsion (SEP) system include inner planets, small bodies, and outer planets and their moons. This range of robotic exploration missions generally calls for ion propulsion systems with deep throttling capability and system input power ranging from 0.6 to 25 kW, as referenced to solar array output at 1 Astronomical Unit (AU). Thermal development testing of the NEXT prototype model 1 (PM1) was conducted at JPL to assist in developing and validating a thruster thermal model and assessing the thermal design margins. NEXT PM1 performance prior to, during and subsequent to thermal testing are presented. Test results are compared to the predicted hot and cold environments expected missions and the functionality of the thruster for these missions is discussed. Anderson, John R. and Snyder, John S. and VanNoord, Jonathan L. and Soulas, George C. Glenn Research Center; Jet Propulsion Laboratory ION PROPULSION; SOLAR ELECTRIC PROPULSION; PROPULSION SYSTEM CONFIGURATIONS; PROPULSION SYSTEM PERFORMANCE; ROBOTICS; ION ENGINES; THROTTLING; SOLAR SYSTEM; SOLAR ARRAYS

In certain cases, Radioisotope Electric Propulsion (REP), used in conjunction with other propulsion systems, could be used to reduce the trip times for outer planetary orbiter spacecraft. It also has the potential to improve the maneuverability and power capabilities of the spacecraft when the target body is reached as compared with non-electric propulsion spacecraft. Current missions under study baseline aerocapture systems to capture into a science orbit after a Solar Electric Propulsion (SEP) stage is jettisoned. Other options under study would use all REP transfers with small payloads. Compared to the SEP stage/Aerocapture scenario, adding REP to the science spacecraft as well as a chemical capture system can replace the aerocapture system but with a trip time penalty. Eliminating both the SEP stage and the aerocapture system and utilizing a slightly larger launch vehicle, Star 48 upper stage, and a combined REP/Chemical capture system, the trip time can nearly be matched while providing over a kilowatt of science power reused from the REP maneuver. A Neptune Orbiter mission is examined utilizing single propulsion systems and combinations of SEP, REP, and chemical systems to compare concepts. Fiehler, Douglas I. and Oleson, Steven R. Glenn Research Center NASA/TM-2004-213220, E-14727, AIAA Paper 2004-3978

The results of the NASA Glenn Research Center (GRC) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) internal Solar Electric Propulsion (SEP) stage design are documented in this report (Figure 1.1). The SEP Stage was designed to deliver a science probe to Saturn (the probe design was performed separately by the NASA Goddard Space Flight Center s (GSFC) Integrated Mission Design Center (IMDC)). The SEP Stage delivers the 2444 kg probe on a Saturn trajectory with a hyperbolic arrival velocity of 5.4 km/s. The design carried 30 percent mass, 10 percent power, and 6 percent propellant margins. The SEP Stage relies on the probe for substantial guidance, navigation and control (GN&C), command and data handling (C&DH), and Communications functions. The stage is configured to carry the probe and to minimize the packaging interference between the probe and the stage. The propulsion system consisted of a 1+1 (one active, one spare) configuration of gimbaled 7 kW NASA Evolutionary Xenon Thruster (NEXT) ion propulsion thrusters with a throughput of 309 kg Xe propellant. Two 9350 W GaAs triple junction (at 1 Astronomical Unit (AU), includes 10 percent margin) ultra-flex solar arrays provided power to the stage, with Li-ion batteries for launch and contingency operations power. The base structure was an Al-Li hexagonal skin-stringer frame built to withstand launch loads. A passive thermal control system consisted of heat pipes to north and south radiator panels, multilayer insulation (MLI) and heaters for the Xe tank. All systems except tanks and solar arrays were designed to be single fault tolerant. Oleson, Steven R. and McGuire, Melissa L. Glenn Research Center SOLAR ARRAYS; PROPELLANTS; PACKAGING; MISSION PLANNING; SOLAR ELECTRIC PROPULSION; ELECTRIC BATTERIES; SPACECRAFT LAUNCHING; DATA TRANSMISSION; GALLIUM ARSENIDES; PANELS

NASA offers the full text of the June 15, 2000 article entitled "The Incredible Ions of Space Transportation," written by Lyn Lepre. Lepre discusses the history of NASA's research with ion propulsion engines. NASA started the NSTAR (NASA Solar Electric Propulsion Technology Application Readiness) program in 1992, which examines the challenges to using ion propulsion on deep space missions.

Advances in solar array and electric thruster technologies now offer the promise of new, very capable space transportation systems that will allow us to cost effectively explore the solar system. NASA has developed numerous solar electric propulsion spacecraft concepts with power levels ranging from tens to hundreds of kilowatts for robotic and piloted missions to asteroids and Mars. This paper describes nine electric and hybrid solar electric/chemical propulsion concepts developed over the last 5 years and discusses how they might be used for human exploration of the inner solar system.

The propulsion module comprises six to eight 30-cm thruster and power processing units, a mercury propellant storage and distribution system, a solar array ranging in power from 18 to 25 kW, and the thermal and structure systems required to support the thrust and power subsystems. Launch and on-orbit configurations are presented for both modular approaches. The propulsion module satisfies the thermal design requirements of a multimission set including: Mercury, Saturn, and Jupiter orbiters, a 1-AU solar observatory, and comet and asteroid rendezvous. A detailed mass breakdown and a mass equation relating the total mass to the number of thrusters and solar array power requirement is given for both approaches. Sharp, G. R. and Cake, J. E. and Oglebay, J. C. and Shaker, F. J. Glenn Research Center NASA-TM-X-3473, E-8800 RTOP 506-22...