An Overview of NASA's Electric Propulsion Programs, 2010-11

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James E. Polk,[*] Hani Kamhawi,[†] Kurt Polzin,[‡] John Brophy,[§]
John Ziemer,[¶] Timothy D. Smith,[‖] and Mike LaPointe[**]

A resurgence of NASA investment in electric propulsion technology has occurred over the last two years. Recent accomplishments include a record of 37415 hours demonstrated life and 640 kg of xenon processed in an ongoing lifetest of the NEXT ion thruster, further maturation of the HiVHAc Hall thruster, and delivery of the colloid micronewton thruster systems for the ST7/LISA Pathfinder technology demonstration mission. Advances have also been made in adapting commercial thruster systems for use in deep space. The fundamental understanding of thruster physics advanced significantly in efforts focused on modeling Hall thruster and hollow cathode wearout failure modes, yielding the promise of much longer thruster life. Development of high power Hall thrusters and high current cathodes for human exploration missions has resumed, along with research in other high power concepts such as pulsed inductive thrusters. Finally, the ion thruster-propelled Dawn spacecraft has rendezvoused with the main belt asteroid Vesta, reaping the benefits of earlier technology investments. The results of these activities over the last two years are reviewed in this paper.

I. Introduction

NASA's electric propulsion programs span a broad range of activities, from basic research to science mission applications. Technology development efforts are currently focused on developing products to meet the needs of NASA planetary, astrophysics, and human exploration missions. The products of these programs include propulsion system components (thrusters, power processing units, propellant management systems), mission design tools, physics-based modeling and simulation tools for thruster design and life qualification, and new mission concepts. Examples of new technologies include NASA's Evolutionary Xenon Thruster (NEXT), a high performance ion propulsion system for deep space missions, colloid micronewton thrusters for disturbance reduction on astrophysics missions, and 20 to 50 kW Hall thrusters under development for very high power vehicles to support human exploration missions.

Advanced development programs combined with technology demonstration missions have produced several important technology infusion successes, such as the use of ion propulsion on the Dawn mission, which just arrived at the main belt asteroid Vesta, and colloid microthruster technology, which is being demonstrated for future use on the Laser Interferometer Space Antenna (LISA) gravity wave detection mission. The

[*]Principal Engineer, Propulsion and Materials Engineering Section, Jet Propulsion Laboratory, California Institute of Technology, M/S 125-109, 4800 Oak Grove Drive, Pasadena, CA 91109

[†]Senior Aerospace Engineer, Propulsion and Propellants Branch, Glenn Research Center, 21000 Brookpark Rd, Cleveland, OH

[‡]Propulsion Research Engineer, Propulsion Research and Technology Applications Branch, Propulsion Systems Department. Marshall Space Flight Center, Huntsville, AL 35812.

[§]Principal Engineer, Propulsion and Materials Engineering Section, Jet Propulsion Laboratory, California Institute of Technology, M/S 125-109, 4800 Oak Grove Drive, Pasadena, CA 91109

[¶]Senior Engineer, Propulsion and Materials Engineering Section, Jet Propulsion Laboratory, California Institute of Technology, M/S 125-109, 4800 Oak Grove Drive, Pasadena, CA 91109

[‖]Advanced In-Space Propulsion Project Manager, Glenn Research Center, 21000 Brookpark Rd, Cleveland, OH

[**]Deputy Manager, Science and Exploration Research Office, Marshall Space Flight Center, Huntsville, AL 35812.

earliest success, hydrazine arcjet technology for use on communications satellites, opened the door for the commercial development of other thruster technologies. Some of these commercial systems are now mature, and may provide low cost alternatives for deep space missions.

This paper provides a review of the accomplishments in each of three application areas over the last two years. To provide context, we first provide an overview of how electric propulsion programs at NASA are structured, discuss how they have evolved over the last 30 years, and present current plans for technology development and application.

II. Programmatic Overview

II.A. General Structure of NASA's Electric Propulsion Programs

NASA-funded technology programs generally involve a combination of in-house research and development, primarily at the NASA Glenn Research Center (GRC), the Jet Propulsion Laboratory (JPL), and the Marshall Spaceflight Center (MSFC), research activities at universities, and contracted work with industry to exploit their expertise and to develop commercial sources for subsequent mission applications. The target applications include competed science missions under programs like Discovery and New Frontiers, which are focused on lower cost classes of planetary missions, more ambitious, directed Flagship missions, Earth science and astrophysics missions, and in-space transportation infrastructure to support human missions.

The timeline in Fig. 1 shows the history and structure of NASA electric propulsion programs over the last 30 years and projections into the next 30 years based on current plans. Technology programs are listed at the top in blue. Successful products of the technology programs flow into either technology demonstration missions (shown in red) or into science missions (shown in green). Potential future science or exploration missions are also shown in green. Commercial programs which have benefited from NASA technology investments or which may supply propulsion systems for NASA missions are shown in orange. This timeline reveals several trends in technology development and application.

1. A shift in primary focus from supporting industry to developing products for NASA-unique missions. NASA has always fostered technologies it needed, but in the late 1980s and early '90s particular emphasis was placed on technology transfer to support commercial space development.[1] An example of this is the in-house and contracted development of arcjet technology under the Office of Aeronautics and Space Technology (OAST) Base Research and Technology Program,[2,3] which ultimately resulted in commercial hydrazine arcjet systems used for North-South stationkeeping on communications satellites.[4] This was of tremendous benefit to US industry and arguably opened the door for the use of electric propulsion on commercial satellites, but did not directly benefit any NASA missions. In contrast, current technology programs are almost exclusively focused on developing products for NASA applications. Commercial applications are encouraged, but the overwhelming emphasis is on meeting NASA's needs. Even the SBIR program, which is specifically designed to foster new technology development in small businesses, now releases solicitations with a clear focus on addressing problems associated with NASA missions. Demonstration of commercial potential is required in the proposals, but often in practice it is sufficient to identify a future NASA customer to satisfy this requirement.

2. Tremendous growth in commercial use of electric propulsion. Despite the trend in NASA away from a primary goal of supporting commercial applications, industrial use of electric propulsion has blossomed. In addition to the arcjet systems which are now standard on the Lockheed Martin A2100 buses, three major product lines have emerged in the last two decades. Boeing started using the Hughes Xenon Ion Propulsion Systems (XIPS) on their 601HP spacecraft in 1997 and the 702 spacecraft in 1999 for North-South Stationkeeping (NSSK) and orbit-raising.[5] Although XIPS was not funded by NASA, it benefited from earlier investments in ion engine technology. XIPS product development, in turn, contributed to the development of the NSTAR ion engine that NASA subsequently flew on the Deep Space 1 and Dawn missions. SPT-100 Hall thrusters developed by Fakel were adapted by Space Systems Loral for use on their spacecraft.[6] More recently, Hall thrusters have been developed domestically by Busek[7–9] and Aerojet.[10] The BPT-4000 Hall thruster manufactured by Aerojet is being used on the Advanced EHF spacecraft.[11,12] The development of Hall thrusters in the US has also benefited from Department of Defense and NASA technology programs.

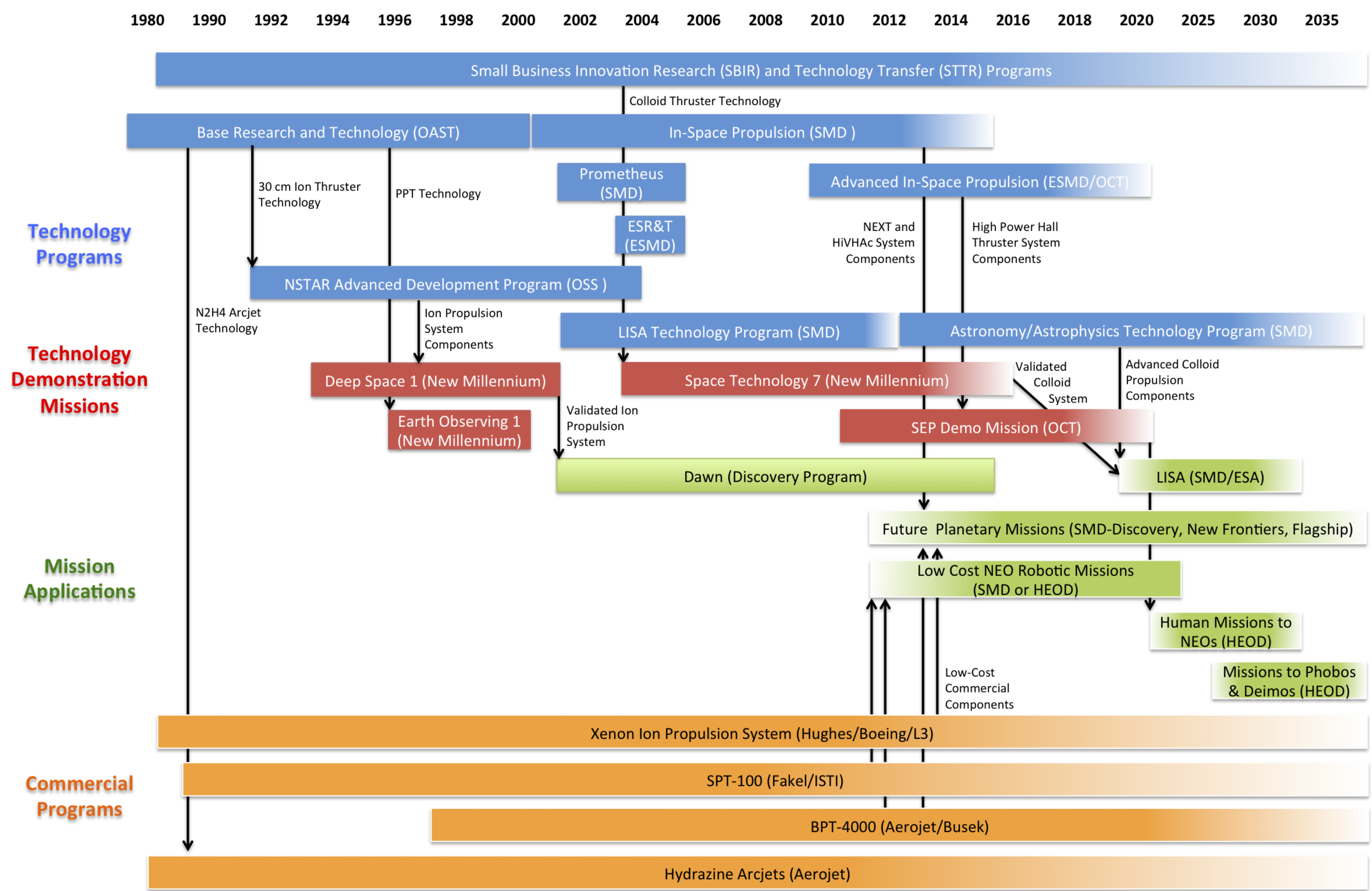

Figure 1. Timeline showing the recent history and structure of NASA electric propulsion programs.

The flight rate of commercial electric propulsion systems is now much higher than NASA's. The requirements of some planetary missions are sufficiently similar to those for which the commercial products were developed that they can be used as is or with minor modifications. The active commercial product lines offer the potential for low cost propulsion systems on cost-constrained competed missions in the coming decades.[13]

3. Volatility in technology funding. The last decade has been marked by extreme volatility in technology funding. Changes in NASA priorities have resulted in the cancellation of relatively short-lived programs such as Project Prometheus,[14] which was focused on developing nuclear electric propulsion systems for robotic outer planet missions, and the Exploration Systems Research and Technology (ESR&T) program, which included the development of high power electric propulsion technologies for human missions.[15] The demise of these programs in 2006 and a subsequent redirection of funding at NASA away from technology development resulted in a 4-5 year gap in technology funding. The In-Space Propulsion Technology program, under which solar electric propulsion (SEP) systems for planetary exploration are being developed, and the LISA technology program, which includes colloid thruster technologies for the LISA mission, provided some continuity.

New programs have been initiated in 2010, in part to address concerns that NASA was not investing enough in technology. The Advanced In-Space Propulsion (AISP) Project was initiated in the Exploration Systems Mission Directorate (ESMD, subsequently reorganized into the Human Exploration and Operations Directorate, HEOD) to support technologies for human exploration. In 2012 it is being transferred to the Office of the Chief Technologist (OCT), as part of a new, large technology development effort. This program is leveraging work on high power Hall thrusters that was started under the Prometheus Project and the ESR&T program.

The NASA centers in general have weathered the recent gap in funding, but volatility like this is also difficult for universities, where stability over at least 4-5 years (the typical duration of a PhD program) is required. Some stability has been afforded by the Graduate Student Research Program (GSRP), started in 1997. This program provides three year grants to support student research, including some work at a NASA center. GSRP fellowships have been exploited by NASA's electric propulsion community to maintain some university involvement during the recent dip in funding. The OCT technology program hopes to provide greater stability in university funding, in part with the new NASA Space Technology Research Fellowship (NSTRF) program which will provide up to four years of funding for graduate students. Six of the 80 awards made in the inaugural year of this program were for research topics in electric propulsion.

4. A trend from cross-cutting programs to mission-specific programs (and back again). In the 1980s and '90s most of the technology development was funded by OAST, a Directorate which was separate from those responsible for implementing planetary, astronomy and astrophysics, and human exploration missions. This was a broad-based program with multiple technology elements and efforts ranging from basic research to low- and mid-technology readiness level (TRL) development to support a number of potential future customers. Advanced development prior to flight applications was conducted under separate programs such as the NASA SEP Technology Applications Readiness (NSTAR) program[16,17] and Earth Observer 1 (EO1).[18] OAST was subsequently dissolved in a NASA reorganization. Technology development responsibility was transferred to the individual Mission Directorates, resulting, for example, in the In-Space Propulsion Technology program in the Science Mission Directorate (SMD) and the Exploration Systems Research and Technology program in the Exploration Systems Mission Directorate. The astrophysics program has until recently funded technology development as part of individual missions. These programs are generally structured to bring specific technologies to TRL 6, at which point they are expected to be sufficiently mature to transition to flight projects. This trend has generally resulted in less diversity in the portfolio, more focused research and technology development, and greater resources for a given technology. The new technology program in the Office of the Chief Technologist may signal a return to cross-cutting research and technology. It is a separate organization from the Mission Directorates and has a requirement for multiple users of the technologies.

5. A trend from technology-push to mission-pull. This trend is related to the one noted above. Technology programs in the individual Mission Directorates tend to follow a model of mission-pull, in which the needs of relatively near-term missions determine the requirements for technology development. The LISA Technology Program is an extreme example of this, where the program exists solely to develop those technologies required for a single mission. Guidance for mission-pull programs comes from decadal surveys, science roadmaps, the

Science Mission Directorate science plan, the NRC technology roadmapping exercise, and ongoing mission and architecture studies.

Cross-cutting programs often follow a technology-push model, in which there is explicit direction to not be responsive to a single, currently planned mission. These programs often have multiple potential future customers, or perhaps no existing advocates. The objective is to develop technologies that are truly transformational, which will enable entirely new mission concepts. This generates a rich environment for the cultivation of new ideas and often has a longer-term focus, launching new technology products into the pipeline for future applications. Technology-push programs, however, have to clearly define what concepts are considered "transformational," accept higher failure rates, be prepared for questions of relevance during periods of tight budgets, and recognize that concepts that are truly new and innovative may require long development times to realize.

6. Pathways to technology infusion. The timeline shows schematically the successful pathways taken by some technologies from development into application. As noted above, the first was low power hydrazine arcjet technology. The transition to flight was driven by strong market forces. An increase in payload life up to 15 years and a trend toward more transponders per spacecraft, and therefore larger spacecraft with more on-board power, led to much larger propellant loads for NSSK. These changes necessitated the use of larger, more expensive launch vehicles. Hydrazine arcjets offered a low risk way to reduce the propellant load, enabling a drop in launch vehicle class with savings of tens of millions of dollars per launch. The choice of hydrazine propellant minimized the risk of adopting a new technology, because the new thrusters used flight-proven feed systems and existing infrastructure for propellant handling. The common hydrazine propellant also allowed architectures with conventional monopropellant systems as a backup on the first few flights. The market climate was therefore favorable, and successes in the OAST and industry technology programs led to rapid infusion.[1] The OAST arcjet technology program began in 1983, arcjets were first used on the Telstar 401 spacecraft in 1993. Hydrazine arcjet systems are now used on the majority of Lockheed Martin commercial communications spacecraft.

At that time, however, NASA was not subject to the same market forces. Launch vehicles were purchased with a separate part of the NASA budget and their cost was not included in the mission cost. There was therefore no incentive on the part of mission managers to take any risks to enable use of smaller launch vehicles. This changed in 1995, when missions were required to include the cost of the launch vehicle in the total mission cost. Despite the potential for significant savings in launch costs with electric propulsion, conservative science missions were still reluctant to adopt new technologies because of perceived risk. NASA started the New Millennium Program (NMP) in 1993 to demonstrate key new technologies in flight to reduce the risk for future users. Deep Space 1 (DS1), the first of the NMP missions focused on planetary exploration technologies, exploited the ion propulsion technologies developed under the OAST Base R&T program and the NSTAR advanced development effort to fly by the asteroid Braille in 1999 and the comet Borrelly in 2001.[19] The success of Deep Space 1 resulted in the selection of the Dawn mission, which is using the same technologies to rendezvous with the main belt asteroids Vesta and Ceres, in late 2001.[20] Earth Observer 1 (EO 1), the first NMP mission supporting technologies for earth-orbiting spacecraft, included a pulsed plasma thruster (PPT) based on technologies developed under the OAST Base R&T program for precision pointing.[18] EO 1 was also very successful, but PPTs have not yet been used in a NASA science application.

Colloid thruster technology, which is taking a similar infusion path, is a success story for the SBIR program. Colloid microthruster development at Busek took place initially under Phase I and II SBIR contracts,[21] and was then chosen for the Space Technology 7 (ST7)/LISA Pathfinder spacecraft, which will demonstrate key technologies for LISA.

The successful demonstrations of ion engines on DS1 and Dawn, as well as technologies such as XIPS, the SPT-100, and the BPT-4000 on commercial spacecraft are thought to have addressed most of the perceived risks for technologies in this power class (1-10 kW). Therefore, there are no plans or need for dedicated technology demonstration missions for new systems such as the NEXT ion thruster or commercial electric propulsion options, and direct infusion into Discovery, New Frontiers, and Flagship missions is being actively pursued. Current plans for the development and application of higher power systems are following the successful models for infusion described above. For example, an ongoing technology maturation program for Hall thrusters in the 20-50 kW power range[22] would be followed by a subscale demonstration of key thruster, power system, and solar array technologies. This demonstration mission, which is in the initial planning phase, would retire the risks for subsequent application to multi-hundred kW vehicles for human exploration missions.[23]

7. Increases in available power. Another key factor in the success of electric propulsion over this time period is the dramatic increase in available solar power on spacecraft. Plotting the power level of the highest power spacecraft launched each year as a function of its launch year (Fig. 2) reveals that the maximum available power has doubled approximately every four years for the last 5 decades.[23] SERT II, the Space Electric Rocket Test II launched in 1970 with 1kW of solar array power is well above the curve, reflecting the high power demands of electric propulsion. Significantly, the two deep-space missions with electric propulsion launched by NASA, Deep Space 1 in 1998, and Dawn in 2007, are well below the curve. By the late 1990's, solar array capability, driven primarily by communications satellite payloads and the International Space Station, had surpassed the levels required for near-term electric propulsion systems. This trend suggests that within the next decade, solar arrays capable of supporting electric propulsion systems with hundreds of kW should be available.

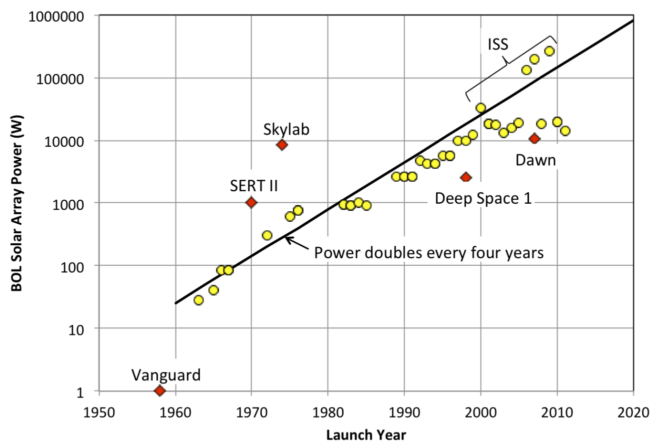

Figure 2. Historical trends indicate that available space solar array power doubles approximately every four years.

In general, interest in electric propulsion at NASA has grown over the last three decades. Electric propulsion options are more often considered by mission planners. A significant indicator of success in gaining acceptance of the technology is that some of the strongest advocates are now found in advanced mission design groups, not just amongst technologists.

II.B. Currently Active Programs

The In-Space Propulsion Technology Program. NASA's Science Mission Directorate initiated the In-Space Propulsion Technology (ISPT) Program in 2001 to develop new, enabling propulsion technologies that could not otherwise be achieved within the cost or schedule constraints of flight projects. The focus is on technologies in the mid-TRL range that can be matured in 4-6 years and will benefit near- and mid-term robotic science missions by reducing cost, mass and/or trip times. The Program goal is to mature technologies to TRL 6 and reduce the risk sufficiently for infusion into science missions. The electric propulsion technologies currently under development by the ISPT Program include the NEXT ion thruster and the High Voltage Hall Accelerator (HiVHAc). In 2009, the ISPT program was tasked with developing propulsion technologies for sample return missions.

Astrophysics Programs. The ST7 payload was initially the seventh selection in the New Millennium technology demonstration series and is currently sponsored by NASA's Astrophysics Division. It will demonstrate precision formation flying technologies for future missions such as the LISA mission, including a colloid microthrust propulsion system. The ST7 disturbance reduction system (DRS) is a payload on the ESA LISA Pathfinder spacecraft along with the European gravitational reference sensor (GRS) as part of the ESA LISA Technology Package (LTP). The LISA Technology Program is a separately funded activity under the

LISA pre-project which is focused on enhancements to the ST7 colloid thruster system to meet the lifetime requirements of LISA. Starting in FY12 it will be part of a larger technology development effort in the Astrophysics Division.

The Advanced In-Space Propulsion Project. In FY11, NASA established the Enabling Technology Development and Demonstration (ETDD) Program to increase the capabilities and reduce the cost of future exploration activities. ETDD established a set of demonstration projects and foundational domain projects, the former to mature technologies toward ground or flight demonstrations, and the latter to provide the framework of innovation required to advance NASA technical capabilities. The Advanced In-Space Propulsion (AISP) Project was set up as a foundational domain project under ETDD to mature low to mid-level TRL in-space propulsion technologies for transition to ETDD demonstration and Agency flagship missions. The principal areas investigated under the FY11 AISP Project include both nuclear thermal propulsion and high power electric propulsion, with an emphasis in electric propulsion on developing high power (50-kW class) Hall thrusters and PPUs, high current cathodes, advanced numerical simulation capabilities, and advanced concepts such as the MPD thruster and Pulsed Inductive Thruster. These areas and related accomplishments are further discussed in the sections below. In FY12, the ETDD AISP Project will transition from the Exploration Systems Mission Directorate to the NASA Office of the Chief Technologist, under a new Advanced In-Space Propulsion Project. The focus of the new project will be to downselect and sequentially mature specific high power electric propulsion technologies in three year increments, with the intent to more rapidly infuse high power electric propulsion systems into future NASA missions.

Technology Programs in the Office of the Chief Technologist. The new technology development effort in OCT includes three programs that span a range of technology readiness levels. The Early Stage Innovation Division funds advanced concepts that are generally at a low TRL. This includes a newly formulated NASA Innovative Advanced Concepts (NIAC) Program which makes awards for initial feasibility studies for visionary concepts. Of thirty initial contracts awarded this year, 8 were for propulsion concepts and two of these involved electric propulsion. Early Stage Innovation also includes the new NASA Space Technology Research Fellowships mentioned above. The Game Changing Technology Division focuses on advanced space technologies in the mid-TRL range that may lead to entirely new approaches for future missions. The goal of the Game Changing Technology Division is to mature technologies through ground testing to the point where they can potentially be transitioned to a system level flight demonstration. As noted above, the Advanced In-Space Propulsion Project is being incorporated into the OCT program as part of the Game Changing Division. The Crosscutting Capability Demonstrations Division aims to bridge the mid-TRL gap and infuse new technologies that are of benefit to multiple customers into science or exploration missions by demonstrating them in flight. A high power (30 kW-class) solar electric propulsion demonstration mission that could lead to multi-hundred kW vehicles is one candidate that is being considered.

Small Business Innovation Research (SBIR) and Technology Transfer (STTR) Program. The SBIR Program makes competitive awards to small businesses (with less than 500 employees) for innovative ideas that address the research and development needs of NASA and other government agencies. An initial Phase 1 award for a relatively small proof-of-concept effort lasts 6 months and, if successful, may lead to a two year Phase 2. The objective is to develop technologies to the point that subsequent commercial development will be funded by other sponsors. The STTR Program funds teams consisting of a small business concern and a non-profit research institution such as a university for cooperative research and development. A list of the current SBIR and STTR contracts related to electric propulsion is included in Appendix A.

Discovery and New Frontiers Programs. The Discovery and New Frontiers Programs fund small- and medium-class competed planetary missions to address NASA's Science Strategy. The Discovery program includes smaller, focused science missions that fit within a $425M cost cap (not including the launch vehicle) and a development time of 36 months or less, with a goal of launching once every 1-2 years. The Dawn mission described below is a part of the Discovery Program. The New Frontiers Program focuses on larger missions with a cost cap of $800M and a launch rate goal of once every 3 years. Many small body missions of interest to these programs are enabled or enhanced by the use of electric propulsion.

III. Planetary Exploration

Recent mission and systems studies continue to confirm the benefits of electric propulsion for robotic planetary missions[24,25] and exciting new concepts, such as using electric propulsion to return an entire near-Earth asteroid to the space station for analysis,[26] have been developed. The arrival of the Dawn spacecraft at Vesta marks the first use of ion propulsion for a planetary mission, reaping the benefits of earlier technology demonstration missions. The NEXT project has completed much of the planned subsystem and integrated system development activities and development in FY 2011 has focused on thruster life validation and completing PPU technology validation. Work on HiVHAc has focused on demonstrating the target performance and resolving issues with the EM thruster. A combination of modeling and experiments has revealed a method of potentially eliminating Hall thruster channel erosion, and other experiments and simulations have shown that barium is very effectively recycled in state of the art hollow cathodes. Both of these results offer the promise of Hall thrusters with very long lifetimes. Finally, several recent developments have helped address the differences in requirements between communications satellite and deep space applications of commercial thrusters, improving the potential for use of these relatively inexpensive systems on competed missions.

III.A. Arrival of the Dawn Spacecraft at Vesta

Dawn, the ninth mission in the Discovery program is led by the principal investigator, Dr. Christopher Russell of the University of California, Los Angeles. JPL was responsible for project systems engineering, mission design, the development of the spacecraft, science payload, the ion propulsion system (IPS), and the navigation plan, and is currently conducting the mission operations. Orbital Sciences Corporation was responsible for spacecraft bus development, flight system integration and testing, and launch operations. The scientific objective of the Dawn mission is the characterization of Vesta and Ceres, the two largest main belt asteroids.[27] Vesta appears to be a dry, rocky body, while water appears to have played a role in the formation of the surface of Ceres, which may still have subsurface water ice. Comparison of the physical and chemical properties of these two very different bodies will help answer important questions about the origin and evolution of our solar system.

Dawn will be the first spacecraft to rendezvous with and orbit two different bodies, a capability that would not have been possible without electric propulsion. The Dawn spacecraft, shown in Fig. 3, employs an IPS based on the system flown on DS1.[16,19,28] The IPS consists of three ion thrusters and thruster gimbal assemblies (TGAs), two power processing units (PPUs), two digital control interface units (DCIUs), a xenon control assembly (XCA), and a xenon propellant tank. The ion thrusters, based on the 30 cm diameter ring cusp design developed under the OAST and NSTAR programs,[29] are capable of operating over a power range of 500 W to 2.3 kW, producing thrust ranging from 19 to 92 mN at specific impulses of 1900 to 3100 s. The system is designed to operate only one thruster at a time, but requires a second engine to process the entire 395 kg xenon propellant load and a third as a spare.

Dawn was launched in September 2007 and entered orbit around Vesta on July 16, 2011, where it will remain for one year before continuing on to rendezvous with Ceres in February 2015. The IPS is responsible for providing all of the post-launch ΔV, a total of 11 km/s, which is almost equal to the total ΔV provided by the Delta-II Heavy launch vehicle. The IPS has already provided a ΔV of 6.8 km/s, breaking the previous record for an in-space propulsion system set by the IPS on DS1. The IPS has accumulated a total of 23450 hours of operation and consumed only 252 kg of xenon.

The Dawn system operation has been nearly flawless.[30] Both PPUs have performed as expected, demonstrating a total efficiency of greater than 92% at full power. The xenon flow system operation has also been perfect, with the exception of a solenoid valve cycle rate slightly higher than expected. The total number of cycles is not expected to exceed the flight allowable value, so this behavior is not a threat to valve cycle life. The TGAs have also performed nominally, with a duty cycle and total number of actuator revolutions less than or equal to the expected values. All three ion thrusters have proved to be extremely reliable. The IPS was operated at a planned weekly duty cycle of approximately 95% during Vesta cruise. There have been no significant missed thrust events. Thruster components appear to be healthy.[30] Changes in thruster operating characteristics that are indicative of wear are within levels expected for the amount of operating

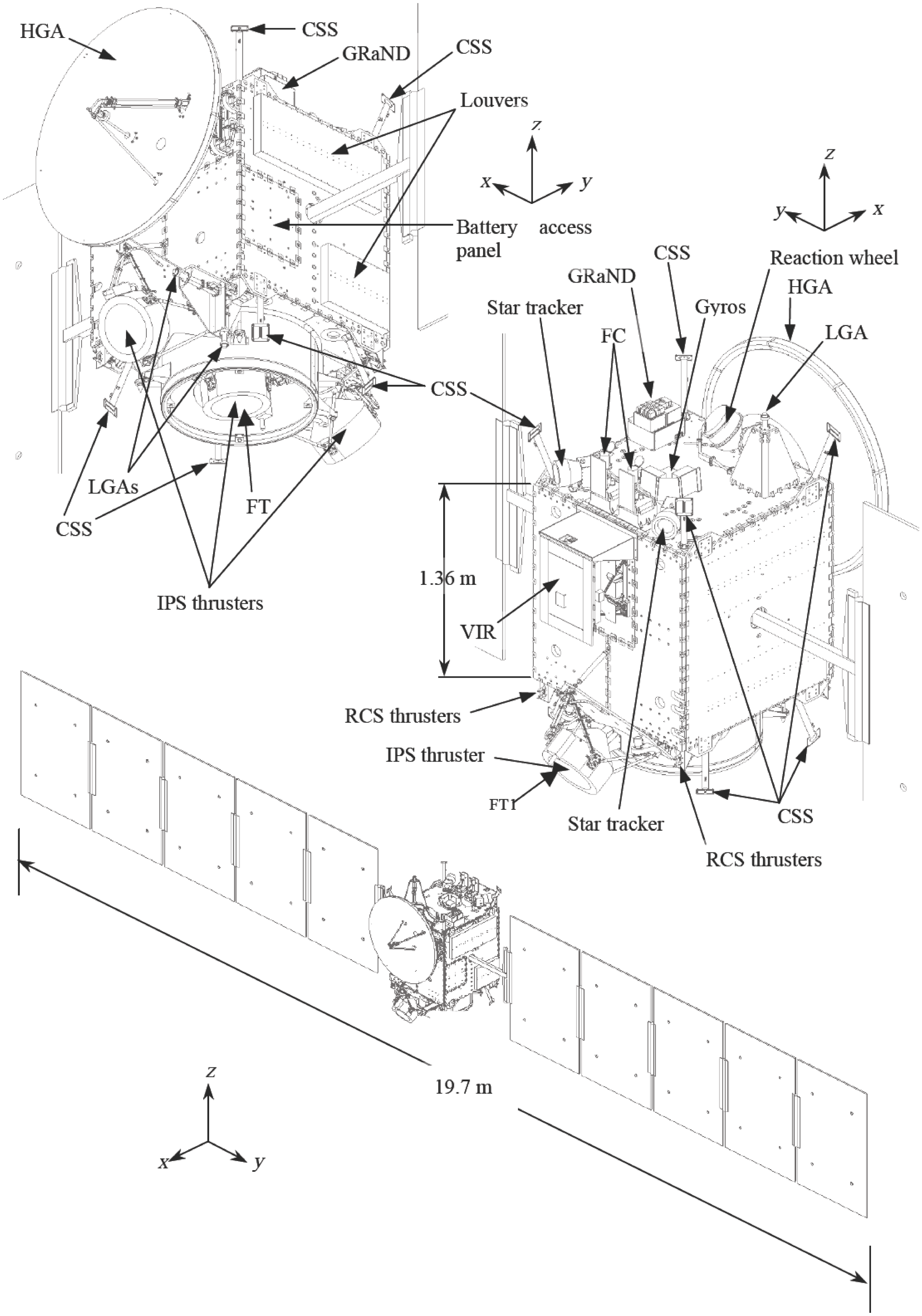

Figure 3. Schematic diagram of the Dawn flight system.

time and propellant throughput accumulated so far. The only issue encountered so far was the loss of a 5V valve control signal in one DCIU. This led to a spacecraft safing event, but operation was quickly resumed on the spare DCIU. Subsequent analysis showed that the loss of the 5V signal was likely due to a single event upset transient, and the DCIU was found to be fully functional again after power cycling.[30]

In general,the behavior of the engines in space has been very similar to that observed in ground tests. The thrust was measured with very high accuracy based on changes in the Doppler shift from the spacecraft radio signal in initial checkout tests[31] and was found to agree quite well with the thrust calculated from engine electrical parameters using corrections for double ion content and beam divergence based on ground tests. Initial thrust values reconstructed with a 1σ uncertainty of < 0.5 mN from navigation measurements during the cruise phase lie within the uncertainty of the calculated thrust, but differ from the calculated values at later times, as shown in Fig. 4. The measured thrust has been as much as 1.2, 1, and 0.5 mN lower than the calculated thrust for FT1, FT2, and FT3, respectively. This behavior is not well-understood, but may be due to changes in double ion content or beam divergence, or errors in thruster telemetry or spacecraft mass estimates based on propellant usage. Trajectory planning is currently based on a conservative thrust estimate of 98% of the calculated thrust. The accelerator grid current for all three thrusters increased by 0.5 to 1.0 mA up to a steady state value at full power in the first 1700 hours. This initial trend is opposite that observed in ground tests, and is not yet understood. The magnitude of the accelerator grid current is similar to the value observed in the Extended Life Test (ELT) of the NSTAR thruster,[32] however, and does not indicate any problems with the ion optics.

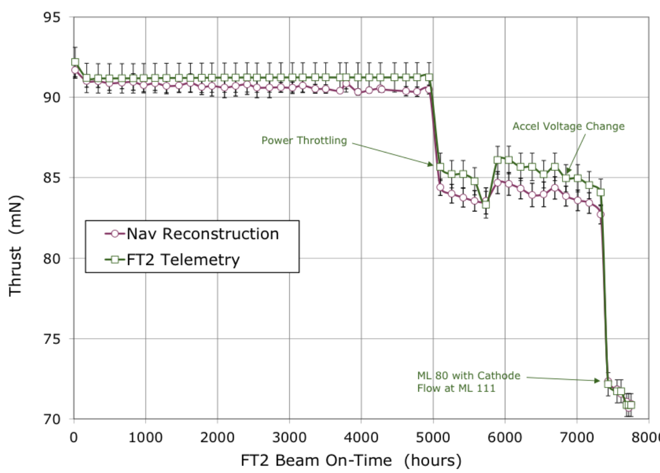

Figure 4. Measured and calculated thrust for FT2 during the cruise to Vesta.

Dawn has provided invaluable experience in operating low thrust systems for deep space missions. The optimum heliocentric trajectory includes some coast periods, although thrusting has been almost continuous since a Mars gravity assist in February 2009. The thrust periods are divided into 7 day thrust arcs, with an 8-9 hour planned shutdown every week for data downlinks and command updates. Optical navigation is used to determine the actual spacecraft position and the trajectory plan was updated every 5 weeks during cruise. The corrections to the trajectory were generally small, and the period between updates could have been longer. Despite near continuous operation, Dawn demonstrated that the IPS requires very low mission operations support. For most of the cruise phase this consisted of one part time propulsion engineer and part time support to generate commands to the IPS. The spacecraft attitude control system (ACS) employs reaction wheels, a hydrazine reaction control system, and the gimbaled IPS. During IPS thrusting, the ACS commands the active TGA to provide 2-axis spacecraft attitude control and to desaturate the reaction wheels. The Dawn spacecraft uses IPS thrust vector control (TVC) as the primary mode during IPS operation. Failure of the IPS to control the spacecraft in the two axes orthogonal to the thrust vector would result in spacecraft safing, which would require considerable effort in recovery. There have been no problems so far, but the DS1 approach appears to be more robust. On DS1, a small attitude deadband was maintained with the IPS, and if the attitude drifted out of this range due to a problem with the IPS, the hydrazine RCS was engaged with a larger deadband.

The recent orbit insertion at Vesta highlights a little-appreciated advantage of electric propulsion. On spacecraft with chemical propulsion, orbit insertion is a large, critical maneuver. In contrast, Dawn's orbit insertion was not a critical event at all. The low thrust trajectory allowed the spacecraft to approach Vesta with very low relative velocity, and capture occurred during normal thrusting. The operation of the spacecraft during insertion was no different from the previous 974 days of thrusting and did not even merit deep space network (DSN) coverage during the predicted time of the event. The next regularly scheduled DSN pass was 25 hours later and just confirmed capture.

Operation at Vesta involves several orbit transfers using the IPS.[30] Capture was followed by spiral down to the initial survey orbit with an altitude of 2735 km, which was achieved August 3, 2011. The spacecraft will remain in the survey orbit for one month, then will spiral down to the high altitude mapping orbit (HAMO) at an altitude of 685 km. After one month in HAMO, the spacecraft will start the 6 week transfer down to the low altitude mapping orbit (LAMO) at an altitude of 200 km. Operations in LAMO will last about 2 months, then Dawn will return to HAMO for 3 weeks before escaping from Vesta in July 2012. Because Vesta has not been visited by spacecraft previously, there is considerable uncertainty about its gravitational field. Electric propulsion may provide an advantage because it allows a slow descent into a progressively more uncertain environment where the little known higher order terms in the gravitational field have more influence. The operating plan for the orbit transfers is somewhat different from operations during cruise. Because of uncertainties in the gravitational field and the thrust vector and magnitude, the trajectory will be flown in shorter thrust arcs with navigational updates as often as once every 3 days. The duty cycle will be reduced to accommodate optical navigation checks and more frequent science data downlinks. The orbit transfer trajectory design requires deterministic thrusting with a thrust vector that varies in time over an orbit. TVC mode, however, requires thrust vector orientations different from those for deterministic thrusting, and therefore results in trajectory errors. The trajectory is sensitive to variations in thrust at the level of a few tenths of a percent, so two modifications to thruster startup operation have been implemented to reduce thrust variability.[30] These modifications result in much more stable and predictable thrust profiles and only minor performance loss.

Dawn represents a significant milestone as the first NASA science mission to use electric propulsion, and is a resounding success for technology infusion. In this case, the prior demonstration on DS1 was essential for addressing the perceived risks in ion propulsion technology. The development of the DS1 system and demonstration in deep space retired risks associated with guidance, navigation and control, integrating the IPS into the attitude control system, interactions with the spacecraft, power management and throttling with a high power system, and low thrust mission planning. A key achievement was demonstrating that mission operations costs are not significantly higher with low thrust systems. Cost constraints on DS1 and the mission design resulted in use of a single string system, so the primary technology development required for use on Dawn was in improved fault tolerance, including the addition of more valves in the XFS. The success of DS1 and Dawn has paved the way for use of other systems such as NEXT, HiVHAc, and commercial thrusters like the BPT-4000.

III.B. Recent Accomplishments in the Development of NASA's Evolutionary Xenon Thruster (NEXT)

The objective of the NEXT project is to develop the next generation of solar-electric ion propulsion technologies for future deep space missions. NEXT, led by NASA GRC with Aerojet, L3 Communications, and JPL as partners, was selected as part of the In-Space Propulsion Technology program in 2002. It includes the development of thruster, PPU, propellant management system, and gimbal technologies.[33] The NEXT ion thruster is shown in Fig. 5. With a beam area approximately 1.6 times that of the NSTAR ion thruster, it operates at power levels ranging from 500 W to 6.9 kW with thrust levels of 26 to 236 mN and specific impulse values of 1320 to 4190 s. The thruster is qualified for a xenon throughput of 450 kg, over twice that of the NSTAR thruster. The high performance NEXT system is targeted for Flagship, New Frontiers, and demanding Discovery missions. The hardware developed as part of the NEXT effort includes an engineering model thruster, referred to as a Prototype Model (PM) thruster by the NEXT project, an engineering model PPU, engineering model propellant management assemblies, and a DCIU simulator, consisting of brassboard level cards that control the high and low pressure feed system assemblies. Each of these units has been extensively tested separately and all have completed integration and environmental tests, with the exception of the PPU, which has experienced a number of failures in testing. The focus of the work over the last two years has been on the continuing Long-Duration Test (LDT) of a NEXT thruster and resolution of the PPU failures.

The NEXT LDT was started in 2005 at NASA GRC as part of a comprehensive effort to assess thruster life using a combination of testing and analysis. The initial goals of the NEXT LDT were to demonstrate the project qualification propellant throughput requirement of 450 kg, characterize thruster performance and erosion as a function of engine wear and throttle level, identify unknown failure mechanisms, and validate thruster life models. After demonstration of the qualification propellant throughput of 450 kg in December 2009 the test was extended with the intention of operating it until it fails in order to learn more about end-of-life failure mechanisms. The test is being conducted with a laboratory model thruster (referred to as an Engineering Model thruster in the NEXT project) which has been modified to a more flight-like configuration by incorporating PM ion optics and a graphite discharge cathode keeper electrode.

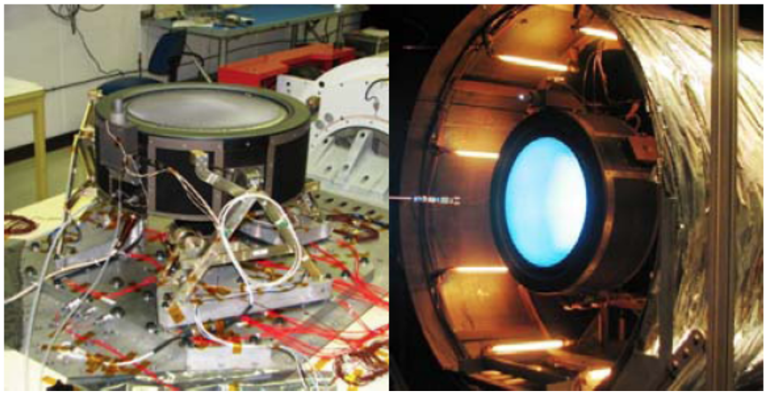

Figure 5. Photographs of the NEXT ion thruster in vibration and thermal-vacuum testing.

The thruster has set records for the longest operating duration, highest propellant throughput, and most total impulse demonstrated for an electric thruster.[34] As of early September 2011, the NEXT LDT has surpassed 37415 hours of operation, processed more than 640 kg of xenon propellant, and demonstrated greater than 24.2 MNs total impulse. Changes in thruster performance have been negligible.[34] Efficiency at full power has dropped by only 0.5% and maximum variations in efficiency and Isp over the entire throttle range have not exceeded 2.5% and 4.2%, respectively. Performance losses are thought to be the result of normal wear processes that occur early in life such as chamfering of the accelerator grid apertures and changes in cathode operation.

Discharge cathode keeper electrode erosion, one of the primary failure modes identified in the NSTAR ELT,[32] has been essentially eliminated as a concern with the use of a graphite keeper.[34] Measurements of cathode and keeper orifice diameter show no changes over the course of the test. The downstream face of the keeper is textured and may have been thinned by about 25%, based on scaling erosion rates measured earlier with molybdenum keeper electrodes. Intermittent shorting of the keeper to the cathode has been observed since 13875 hours. This has been observed in the NSTAR wear tests, and is due to flakes of tungsten sputtered from the cathode onto the upstream face of the keeper.[32] This is an indication of cathode erosion and can lead to delayed ignition and discharge extinctions at low power, but is not thought to be a significant life-limiting phenomenon.

The neutralizer cathode has set new records in cathode test duration, having accumulated over 38000 hours. The neutralizer keeper voltage has decreased steadily, which is an indication of expected erosion at the downstream end of the cathode orifice.[35] The minimum orifice diameter has not increased during the test and there is no evidence of the orifice clogging that occurred at low current levels in the NSTAR ELT.[32] The only concern noted with the neutralizer was a steady loss in plume mode flow rate margin. This was addressed by modifications to the PM neutralizer design and a new throttle table that specifies increasing neutralizer flow with xenon throughput to maintain sufficient margin above the flow rates where the plume mode transition occurs.

Reduced beam current density in the NEXT thruster compared to NSTAR has effectively mitigated accelerator grid aperture enlargement due to sputter erosion, which was identified as the first failure mode in the NSTAR ELT.[32] The accelerator grid ion impingement current has dropped slightly over the course of the test, due primarily to chamfering of the downstream ends of the accelerator grid apertures. In situ measurements show essentially no change in the minimum diameter of the apertures, however. The primary ion optics erosion mechanism is pit-and-groove erosion on the downstream face of the accelerator grid. The groove depth (representing the point of minimum erosion in the pit-and-groove pattern) continues to increase linearly with time, as shown in Fig. 6, and structural failure due to penetration of the accelerator grid is projected to occur after about 45000 hours of operation.[36,37] This is expected to be the life-limiting failure mode, but at that point the thruster will have processed over 800 kg of xenon and demonstrated over 30.7 MN-s total impulse.

The engineering model PPU designed and fabricated by L-3 Communications for the NEXT project is shown in Fig. 7. It operates with a high power input bus voltage of 80 to 160 VDC and supplies up to 7.2 kW

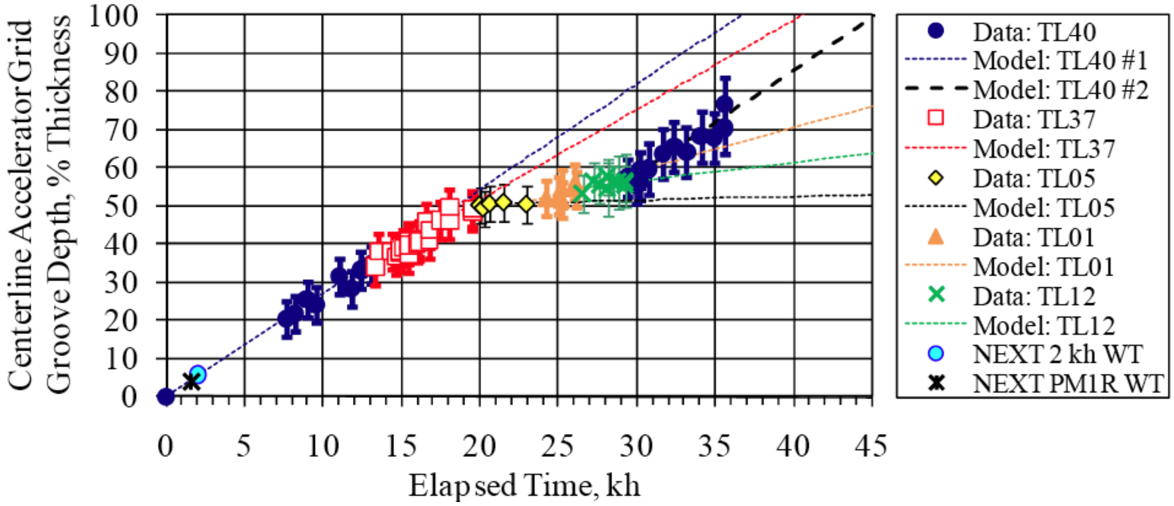

Figure 6. Measured centerline accelerator grid groove depth as a function of operating time. Extrapolation to grid penetration indicates failure after 45,000 hours.

of power to the thruster.[38] With the exception of the beam supply, the power converter topologies in the NEXT PPU are the same as the NSTAR/DAWN PPU and the mechanical design is more modular and thermally capable.

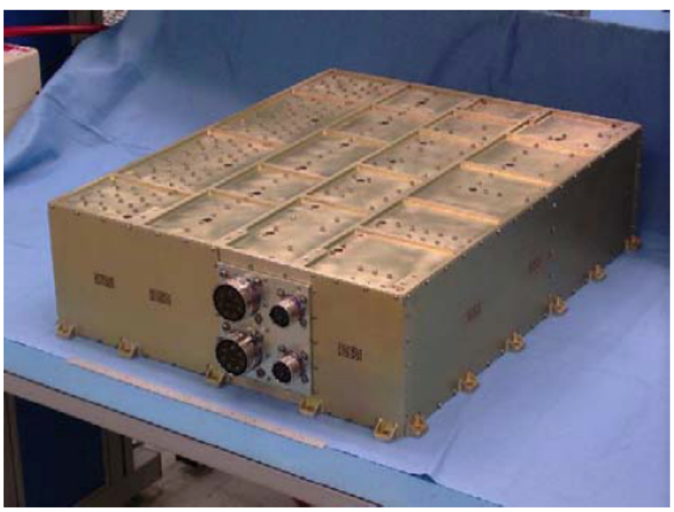

Figure 7. Photograph of the NEXT power processing unit.

The beam supply which processes most of the power consists of six parallel modules that use an innovative dual-mode, dual-bridge topology which provides high efficiency through a wide output range. An efficiency in excess of 94% was demonstrated at full power. The PPU completed functional tests in air and vacuum and satisfied many objectives of a system integration test. However, throughout these tests, the PPU experienced a series of component failures.[38] Formal failure investigations were conducted and over the last two years the root causes of these failures have been identified and corrective actions defined.

The first incident involved failure due to shorting of four out of six high voltage diodes used to rectify power from the power transformers in one of the beam modules. Tests after the diode failure revealed that the part has a substantially higher reverse recovery time at elevated temperature, which causes excess power dissipation and migration of aluminum from braze material into the sillicon diode. The corrective action was to replace these diodes with a part that has superior reserve recovery time, although concerns about a single source for this part must be addressed.[38]

Three separate failures of multi-layer ceramic (MLC) capacitors on the input filters of different beam modules occurred during testing. Additional tests and modeling ultimately showed that the failures were due to dielectric cracking from mechanical vibrations in an adjacent ferroelectric material in the capacitors. The vibrations were induced by high current oscillations in a tank circuit formed by the parasitic inductance of input wires in combination with the filter capacitors in the beam module. The corrective action involves replacing the MLC capacitors with military-grade parts that are not susceptible to time-varying electric fields and modifications to the input filter to reduce the current oscillations.[38]

One of the six beam supply modules sustained significant damage during the last MLC capacitor failure, so subsequent functional testing was completed with the remaining five modules after rework to implement the corrective actions. Ambient pressure testing at room temperature and at the flight allowable high temperature of 50°C demonstrated that the unit operates with the same high total efficiency measured before and meets all functional requirements, with the exception of the telemetry accuracy on certain signals. This issue is currently under investigation.[38]

The NEXT project plan includes demonstration of the PPU to TRL 6. An engineering assessment team concluded that the design with the corrective actions is a good baseline for subsequent development. However, they expressed some concern that the existing unit may not survive vibration testing because of extensive rework as part of the failure recoveries. A strategy for replacement of damaged hardware, modification of circuit elements, engineering updates and completion of key models and analyses was developed to ready the unit for test and is part of the current budget submission. Subsequent planned environmental tests will address the primary environments for a flight application. The project recommends that a flight project incorporate a qualification design update, fabrication and test cycle into the flight development.[38]

As a new technology infusion strategy, NASA is employing cost cap credits on competed Discovery and New Frontiers missions for the use of some advanced technologies, such as the NEXT IPS and Advanced Stirling Radioisotope Generators (ASRGs). Proposers using the NEXT system were given a mission cost cap $19M higher than the usual cost cap as an incentive. This is designed to partially offset the anticipated investment required to fully qualify the system for use on a science mission. Although this has not resulted in the selection of proposals employing NEXT yet, it has apparently worked for ASRGs. Two out of the three proposals selected for Step 1 development in the last Discovery round used ASRGs.

III.C. Progress in the Development of the High Voltage Hall Accelerator (HiVHAc)

The goal of the HiVHAc activity, led by NASA GRC with Aerojet as a partner, is to develop a long-life engineering model (EM) Hall thruster system to TRL 6. The performance and lifetime objectives include a throttling range of 0.3 to 3.5 kW with a thrust efficiency > 55% and specific impulse > 2,700 s at full power, with a xenon throughput capability > 300 kg, corresponding to an operational lifetime of >15,000 hrs. The HiVHAc project and Air Force Research Laboratory (AFRL) are furthering the development of an ISPT-funded Advanced Xenon Feed Module (AXFM) by VACCO Industries. The project is also leveraging PPU developments in industry for use with the HiVHAc thruster.

To demonstrate the HiVHAc project performance, throttleability, and lifetime goals, two laboratory thrusters were built and tested. The NASA-77M operated with input powers between 0.2 and 2.8 kW, specific impulse levels between 922 and 2,911 s and thrust efficiencies between 31% and 54%. The NASA-77M was also subjected to two 300 hour wear tests to assess discharge channel wear characteristics. The subsequent design, the NASA-103M.XL, demonstrated a total efficiency between 33% and 55%, a specific impulse between 1,062 s and 2,779 s at power levels between 0.3 and 3.5 kW, respectively. The NASA-103M.XL thruster also incorporated a discharge channel replacement mechanism designed to extend thruster life. The feasibility of this approach was demonstrated in a 5000 hour wear test at a discharge voltage of 700 V.

After the successful demonstration of the channel replacement mechanism in the laboratory model thruster, NASA GRC teamed with Aerojet to develop a HiVHAc EM thruster that meets the project goals. The EM thruster incorporates the channel replacement mechanism and is designed to survive structural and thermal environments based on a deep space reference mission identified for NEXT.

However, extensive testing of the EM thruster revealed issues with the magnetic circuit, thermal design, and operation of the channel replacement mechanism. The thermal issues are a result of a thruster power density that is approximately 1.5 times higher than state-of-the-art thrusters. Rather than derate the thruster, NASA GRC and Aerojet implemented a number of design changes. The design modifications included a new anode isolator and anode mount design that greatly enhanced heat conduction from the anode assembly, a new magnetic circuit that preserved the HiVHAc EM thruster magnetic field topology while operating at lower electromagnet currents, new electromagnet designs that operate at lower temperature, coatings to thruster surfaces to enhance radiation, and the addition of a radiator. To resolve the problems with the channel replacement mechanism the redesign incorporated new boron nitride discharge channel configurations that are structurally more robust and a new discharge channel advancement mechanism that is simpler and more robust. The design changes were necessary to ensure sustained thruster operation for the entire thruster operating power range and to ensure that the channel replacement mechanism deploys and functions as designed. The new HiVHAc thruster, shown in Fig. 8 is designated HiVHAc EM-R (rework).

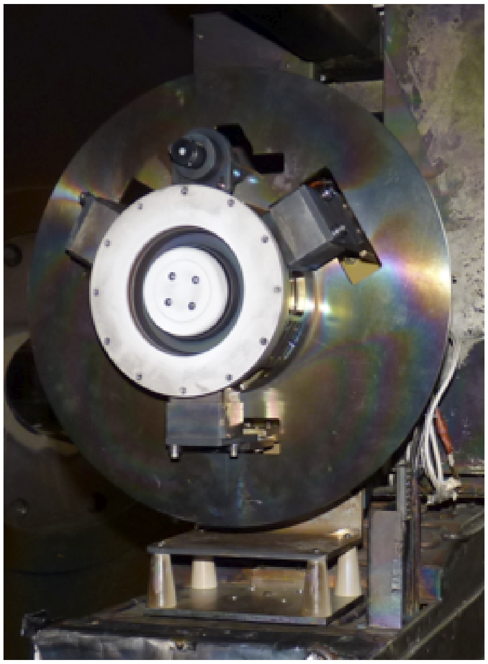

Figure 8. Photograph of the HiVHAc EM-R thruster.

The measured discharge efficiency and specific impulse (not including cathode flow) for the EM thruster are plotted in Fig. 9. A peak total thruster efficiency (including cathode flow) of 59% was demonstrated when operating the thruster at a discharge voltage of 400 volts at 3.5 kW. A peak total specific impulse of 2720 s was demonstrated when operating the thruster at a discharge voltage of 650 volts at 3.7 kW. These tests will soon be repeated with the HiVHAc EM-R with the expectation that it will demonstrate a total efficiency and specific impulse that match or exceed that of the HiVHAc EM thruster. A new throttle table for the HiVHAc EM-R thruster will also include operation at a discharge voltage of 700 volts which will result in a thruster specific impulse >2,700 s. Finally, the thermal design changes incorporated in the HiVHAc EM-R design may result in a thermal operating margin, thus allowing the HiVHAc EM-R to operate at power levels approaching 3.9 kW.

The HiVHAc EM-R thruster has completed a series of functional tests and will soon undergo performance measurements, random vibration testing followed by functional, performance, and thermal characterization tests, thermal vacuum testing, and one or two short duration tests (1,000-2,000 hours each). After successfully completing the short duration testing, the thruster will enter a long duration test (LDT). Some of these tests will include the VACCO advanced xenon feed system.[39]

The HiVHAc project is considering several PPU options, including a unit based on new discharge modules under development by Aerojet,[40] and several PPU technologies being developed under SBIR contracts (see Appendix A). The most mature technology of those being developed in the SBIR program is the Colorado Power Electronics (CPE) brassboard PPU,[41] which exploits an innovative three-phase resonant topology capable of efficiently delivering full power over the wide input and output voltage ranges. The 3.8 kW PPU contains two high voltage discharge modules, cathode heater and keeper power supplies, and two electromagnet power supplies and can operate at input voltages between 80 and 160 volts. Extensive testing of CPE hardware produced as part of the SBIR in bench tests and with the EM thruster confirms that it operates over the required throttle range.

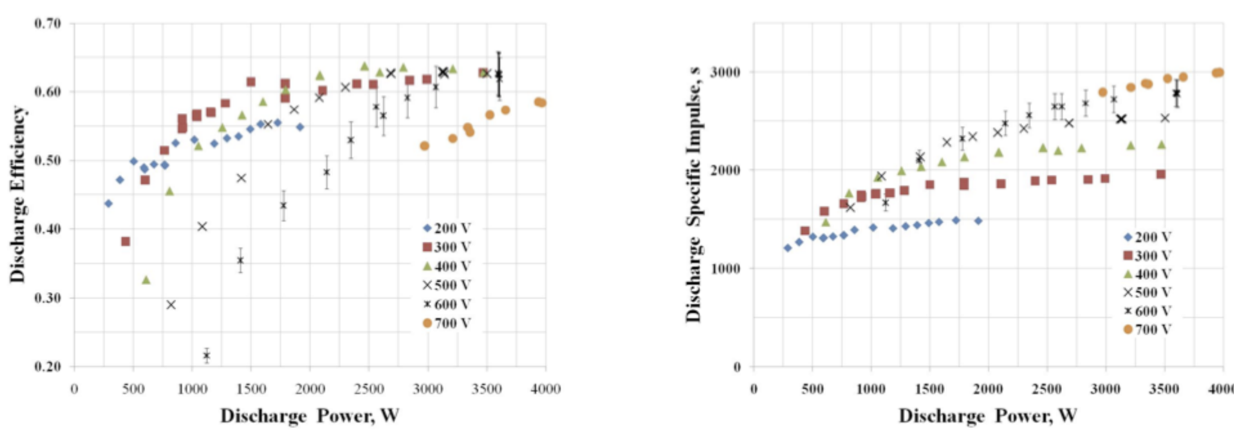

Figure 9. Measured discharge efficiency and specific impulse for the HiVHAc EM thruster.

III.D. Basic Thruster Physics Investigations and Model Development

For the last fifteen years, JPL has been developing and experimentally validating physics-based models of ion engine, Hall thruster, and hollow cathode failure modes. This was motivated by the recognition that analysis would have to play a significant role along with testing in qualifying thrusters for tens of thousands of hours of operation in deep space missions, and that a fundamental understanding of the wearout processes was critical to developing that capability. This work was supported initially by the NSTAR program and more recently by the AISP project. The focus over the last two years has been on understanding channel wall erosion in Hall thrusters and barium depletion in hollow cathodes.

The Hall thruster component of this work has employed results from wear tests, laboratory experiments and numerical simulation to develop an approach for designing thrusters that are free from channel wall erosion, which has been the primary life-limiter in Hall thrusters. In a Qualification Life Test (QLT) of the BPT-4000 Hall thruster that recently accumulated over 10000 hours it was found that the erosion of the acceleration channel practically stopped after about 5600 hours.[12] To understand the detailed physics behind this result, numerical simulations of the thruster were performed using the 2-D axisymmetric, magnetic field-aligned-mesh (MFAM) plasma solver Hall2De, which has been under development at JPL since 2009.

The simulations revealed that several processes led to this significant reduction in erosion, all related to the topology of the magnetic field that was exposed near the wall when the channel receded from its early-in-life geometry to its steady-state configuration.[42] Specifically, the particular topology and magnitude of the magnetic field near the diverging walls of this thruster in its steady-state configuration had the following consequences: 1) reduction of the accelerating electric field parallel to the wall, prohibiting ions from acquiring significant impact kinetic energy before entering the wall sheath, 2) reduction of the potential fall in the sheath at the wall, further diminishing the total impact energy that ions could gain before striking the material, and 3) reduction of the ion number density, decreasing the flux of ions to the wall. Because of these three phenomena, the magnetic field configuration resulted in an effective shielding of the walls from any significant ion bombardment.

The identification of magnetic shielding physics by Hall2De in Aerojet's BPT-4000 Hall thruster design is potentially a breakthrough. If this fundamental understanding can be applied to properly design the shape of the magnetic field near the channel walls, their erosion can be eliminated, thereby retiring the major risk associated with the use of this electric propulsion technology on NASA science missions. In a proof-of-principle effort to demonstrate that magnetic shielding can be designed into Hall thrusters, an existing laboratory thruster has been modified with the guidance of physics-based numerical simulation.[43] The magnetic circuit of the 6 kW H6 thruster, designed by the Air Force, the University of Michigan, and JPL as a testbed for Hall thruster physics, was modified to produce a magnetic field topology near the walls that should reduce the total energy and flux of ions to the channel insulators by greater than 1 and 3 orders of magnitude, respectively. Erosion rates in this modified configuration will be measured experimentally, but are predicted to be at least 2-4 orders of magnitude lower than those in the baseline configuration. At such rates, erosion over the anticipated operating time of Hall thrusters in deep-space science missions is expected to be negligible. Planned tests at JPL also include extensive measurements of plasma properties in the baseline and modified channels with wall-mounted probes and fast scanning probes to validate the code.

Dispenser hollow cathodes used in ion and Hall thrusters rely on a consumable supply of barium released by impregnant materials in the pores of a tungsten matrix to maintain a low work function on the electron-emitting surface. Failure due to evaporation and subsequent loss of barium has been observed in dispenser cathodes used in vacuum devices, and barium depletion has been identified as a potential failure mechanism in hollow cathodes as well. However, research over the last four years has revealed that barium transport in state-of-the-art hollow cathodes differs fundamentally from that in vacuum cathodes, and barium depletion is unlikely to be a significant failure mechanism in these cathodes.[44–47] This conclusion is based on extensive analysis of two cathodes operated for 8200 and 30472 hours, respectively, in NSTAR ion engine wear tests, physics-based models of the xenon plasma inside hollow cathodes that have been validated with experimental measurements, and models of barium transport through the xenon plasma.

Experiments and simulations show that 3 key processes control barium depletion in hollow cathodes.[45] First, erosion and redeposition of tungsten at the downstream end of the emitter result in the formation of a dense tungsten shell on the surface. This occurs right in the primary electron emission zone, and the reduced porosity of the matrix due to this restructuring inhibits the flow of barium to the surface. Despite this, cathodes exhibit no signs of degradation. Second, barium is very effectively recycled in the discharge, greatly extending dispenser cathode life. Barium supplied from the insert upstream of the electron emission zone is ionized in the high density xenon plasma and pushed down to the emitter by the electric field. Resupply through the gas phase by recycling compensates for the loss of vapor flow from the interior when the tungsten shell forms. Very little barium vapor is lost through the upstream or downstream cathode boundaries. Third, the dynamics of barium transport in the plasma result in a high neutral barium partial pressure near the emitter surface. Over a substantial fraction of the emitter length, the barium partial pressure exceeds the equilibrium vapor pressure of barium over the reaction front and suppresses the reactions. As a result, barium is supplied primarily from a narrow region located between the downstream part of the emitter covered with the dense tungsten shell and the upstream region where the barium partial pressure exceeds the vapor pressure. Because recycling very effectively reduces the loss of barium from the hollow cathode, the restricted supply from this narrow region is adequate.

The restructuring of the tungsten surface is not well-understood, but has been characterized experimentally. Barium recycling and suppression of reactions by vapor buildup were identified by modeling. Recent measurements of the depth of barium depletion from the cathode inserts from both of the NSTAR tests reveal that barium loss is confined to a narrow region near the downstream end, confirming this picture of hollow cathode operation.[45,47] These tests show that impregnated hollow cathodes can have extremely long lifetimes if they are configured to exploit barium recycling. Even after 30472 hours of operation there was negligible depletion of the barium supply in the ELT cathode. Current research is focused on determining if barium recycling is equally effective in large, high current hollow cathodes.

III.E. The Use of Commercial Systems for Deep Space Missions

Commercial electric propulsion technology has now matured sufficiently that ion and Hall thrusters developed for communications satellite applications are potential candidates for deep space missions. Propulsion systems procured from active commercial product lines are attractive because they are likely to be substantially cheaper than NASA-unique systems with very low flight rates, such as that built for the Dawn mission. In addition, commercial Hall thruster systems can be as much as 50% less expensive than an equivalent ion thruster system.[13] Efforts have been underway for several years to evaluate and qualify commercial systems for deep space applications.

In most cases, the qualification programs completed for commercial systems meet or exceed the requirements of deep space missions. However, some differences between the GEO satellite and science mission requirements need to be addressed in delta qualification programs. Planetary missions generally require a greater throttling range to accommodate variations in solar array power with solar distance, a greater PPU input power range for spacecraft with unregulated buses, and longer lifetime. Previous efforts to address these issues have been reported elsewhere, but there have been three important recent developments.

First, a recent study of low cost spacecraft for missions to near earth objects (NEOs) showed that commercial Hall thruster systems can be used with no modifications to the PPUs.[48] In addition to being scientifically interesting bodies, NEOs are the focus of near-term human exploration plans, so robotic precursor missions to survey a range of potential targets for subsequent human exploration would be of great value. Models for solar array and engine performance combined with low-thrust trajectory analyses for a large range of NEO targets show that the array output voltage variation is within the known operating range of both the SPT-100 and BPT-4000 PPUs for a solar range of 0.7 AU to 1.5 AU. This enables the use of small, inexpensive, heritage-based spacecraft that can be launched as secondary payloads to explore virtually any NEO of interest and in many cases multiple NEOs in a single mission.

Second, recent performance measurements at JPL demonstrated that the BPT-4000 is capable of operating at much higher specific impulses than the qualification level.[49] Commercial Hall thrusters generally operate at a maximum specific impulse of 2000 s or less. This is lower than that of ion thruster systems (3100-4200 s) or Hall thrusters developed specifically for planetary missions, such as HiVHAc (2700 s), and results in lower delivered mass capability for some potential missions. An engineering model BPT-4000 provided by Aerojet and Lockheed Martin Space Systems was operated over a discharge voltage range of 500-800 V and

2.5-5.5 kW discharge power, producing thrust levels of 134-288 mN with a specific impulse range of 1980-2720 s and total efficiency values of 50%-59%. The results showed that operating the BPT- 4000 at specific impulses of 2600 s and 4.5 kW discharge power was within the existing capabilities of the magnetic circuit, thermal margins, and voltage isolation. Earlier tests demonstrated stable operation down to 150 W, yielding a power throttling range of 30:1. Thruster lifetime would have to be demonstrated over the extended voltage range in a delta qualification, but a scaling analysis shows that at least the same total impulse as already demonstrated should be achievable at higher Isp, and the observation of magnetic shielding in this design indicates that it may have much longer life.

Finally, Aerojet has recently demonstrated a modular power converter compatible with the unregulated input bus voltage of NASA spacecraft and capable of providing up to 800 V output for higher Isp operation.[40] This is part of an effort by Aerojet and JPL to develop a scalable PPU architecture that can meet the needs of future commercial and NASA applications. Two converters were fabricated and tested in parallel to demonstrate current sharing at voltages up to 400 V and in parallel to demonstrate the full 800 V capability. Converter power efficiency was measured over the full input and output voltage range, with peak efficiencies exceeding 94%. Demonstration of these power modules retires the major risk associated with converting Aerojet's existing regulated PPU technology to a new, more flexible architecture.

IV. Astrophysics Missions

Astrophysics missions in some cases require high ΔV primary propulsion and could benefit from electric propulsion systems such as NEXT.[50] Other types of in-space astrophysics observatories require very precise spacecraft positioning and pointing, a unique niche for electric microthrusters. One example of this is the proposed Laser Interferometer Space Antenna (LISA) mission, which is designed to detect low frequency gravitational waves produced by sources such as supermassive blackholes and the merger of binary blackholes. The LISA mission concept consists of three spacecraft flying in an equilateral triangle formation in an earth-trailing heliocentric orbit. The spacecraft are separated from each other by 5 million km and each contains a free-floating test mass. The relative distances between the test masses are monitored to within a resolution of 20 picometers by laser interferometry, yielding an extremely sensitive measurement of the distortions expected from gravitational waves. The spacecraft must fly free around the test masses, maintaining the relative position of the spacecraft and the test mass to within 1 nanometer. Microthrusters are required to eliminate disturbances such as solar photon pressure and keep the spacecraft centered on the test mass. The metrology and microthruster technologies for LISA will be demonstrated on the ST7/LISA Pathfinder mission.

Busek Co., Inc. delivered two flight-qualified clusters of four Colloid Micro-Newton Thruster (CMNT) systems to JPL in February and May of 2008. The clusters, shown in Fig. (10), will provide precise spacecraft control for the ST7 drag-free technology demonstration. To achieve the nanometer-level precision spacecraft control requirements, each of eight thruster systems is required to provide thrust between 5 and 30 μN with resolution 0.1 μN and thrust noise $0.1\mu\text{N}/\sqrt{Hz}$. The CMNT was developed by Busek Co. Inc., with on-site support from JPL in design and testing, over six years into a flight-ready and flight-qualified microthruster system, the first of its kind.[51,52] Flight-unit qualification tests included vibration and thermal vacuum environmental testing, as well as performance verification and various acceptance-level tests. All tests were completed successfully at Busek prior to delivery to JPL. At JPL, the units successfully passed through mass distribution, magnetic, and EMI/EMC measurements and tests as part of the integration and test (I&T) activities with the integrated avionics unit (IAU). Flight software sequences including operation of the thrusters were tested and validated successfully to the extent possible in ground testing at ambient atmospheric conditions, including full functional and 72 hour autonomous operations tests.

In August of 2009 the cluster assemblies were delivered to ESA along with the IAU for integration of the full DRS instrument into the LISA Pathfinder (LPF) spacecraft, shown in Fig. (11). Mechanical and electrical integration onto the LPF spacecraft, including a full-functional test (thrusters on, high voltage PPUs active up to 2 kV, but propellant valves not open) through the LPF ground data system, was completed by the end of November 2009. ATLO activities at the spacecraft-level with the DRS activated have included magnetics, dynamics, and EMI/EMC testing as well as operational sequence tests using both the DRS test bed, which includes a software-based thruster simulator and EM IAU, and the flight units on the spacecraft. All tests have concluded successfully from the DRS standpoint. Future ATLO activities will include spacecraft level thermal vacuum and acoustic environmental testing as well as additional operations tests including closed-loop simulations with the LTP and DRS test beds.

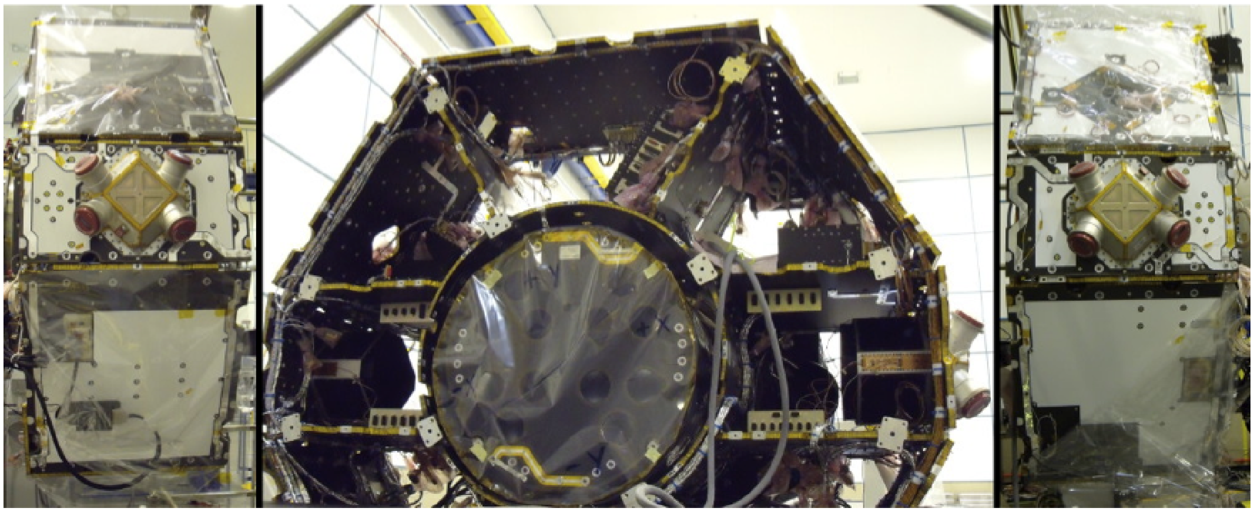

Figure 11. Photograph of the ST7 LISA Pathfinder spacecraft. The Colloid Micro-Newton Thruster clusters are visible on each side of the spacecraft bus.

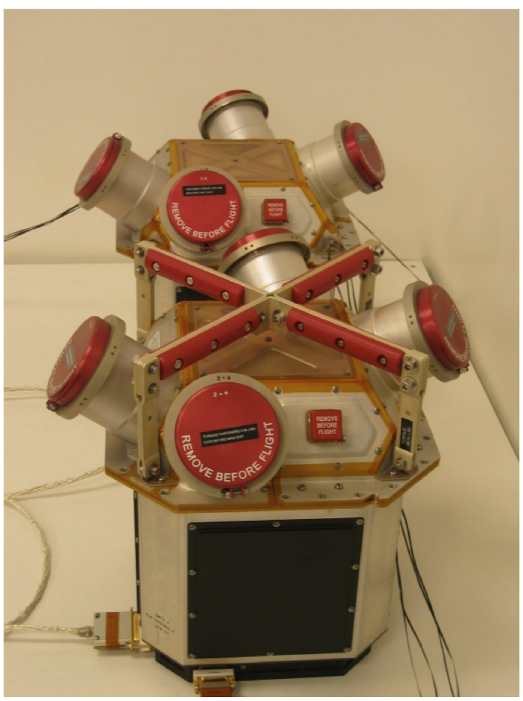

Figure 10. Photograph of the ST7 flight Colloid Micro-Newton Thruster clusters.

While ATLO activities have been ongoing since 2009, two key components of ESA's LTP have not yet been delivered or integrated: the caging mechanism for the GRS and the FEEP thrusters. Under current plans, the spacecraft will go into storage beginning late 2011 until those components are ready (at the thruster delivery, the original launch date was scheduled for April 2012). With both subsystems working to complete development and qualification testing as fast as possible, the best schedule estimate has the spacecraft coming out of storage in August of 2013 for completion of ATLO with an April 2014 launch. Within a few weeks of launch, all eight CMNTs will be fully activated to provide a back-up option to the FEEPs during propulsion module separation and de-spin/tip-off recovery. Three months later, the thrusters will be fully utilized and demonstrated as a drag-free actuator in the 3-month long DRS experiment.

Because the thrusters have been loaded with propellant since late 2007 for acceptance and proto-flight qualification testing at Busek, the predicted shelf-life of the propellant and various wetted feed-system components has come under some scrutiny (the new requirement is up to 7 years with the predicted launch date of April 2014). To provide a better understanding of propellant and material stability, JPL's Chemistry Laboratory has performed various accelerated high-temperature material and propellant exposure tests. The materials include flight-batch samples of all wetted parts (stainless steel tank and feed lines, silicone elastomers, and wetted Microvalve parts) and propellant from the same batch as what was originally loaded in each thruster. Fortunately the materials show no sign of any relevant property change (elasticity, yield strength, etc.) over an equivalent 7-year exposure. The propellant used in all the exposure tests was also examined for any material leaching or change in physical properties (conductivity, surface tension, IR spectrum, etc.) and was found to be stable within the errors of each measurement over the same period. In addition, Busek is conducting functional and leak tests of the Microvalve and thruster manifold (the two main components of the CMNT feed system) under similar accelerated conditions. These tests will be completed by November of 2011 (no issues have been observed to date) providing an equivalent 8 years of exposure and 1 year of operation. Busek and JPL also have additional samples of the propellant stored under ambient conditions in flight-light tanks that have been filled since early 2006 and show no degradation in physical properties to date. At this time, there is no reason to believe the thrusters will behave differently on-orbit in 2014 than they performed in qualification tests in 2007, but they will not be fully operated (valves will remain closed) until on-orbit operations.

V. Human Exploration Missions

Recent studies show the benefits of electric propulsion for pre-placing cargo and transporting crew in human exploration missions. For example, use of SEP systems with power levels on the order of 300 kW can reduce the initial mass required in low earth orbit (IMLEO) for piloted missions to NEOs by 40 to 60% with no difference in flight time compared to chemical propulsion.[53,54] In this"Electric Path" concept, a SEP freighter is used to pre-position a deep space habitat and exploration vehicle in a high Earth orbit. A chemical propulsion stage and a crew vehicle are launched to rendezvous with the SEP freighter, and a chemical propulsion maneuver places these elements in a heliocentric trajectory. The SEP system is then used to rendezvous with the NEO and performs the maneuvers required to return the crew to Earth. A similar architecture with vehicle power levels of 300-700 kW can perform conjunction-class missions to Phobos and Deimos with reductions of 44-60% in IMLEO compared to chemical propulsion options.[55]

A conceptual design for a 300 kW SEP vehicle for human NEO missions developed by NASA's Human Exploration Formulation Team (HEFT) is shown in Figure 12.[23] Mass and cost models for the the elements

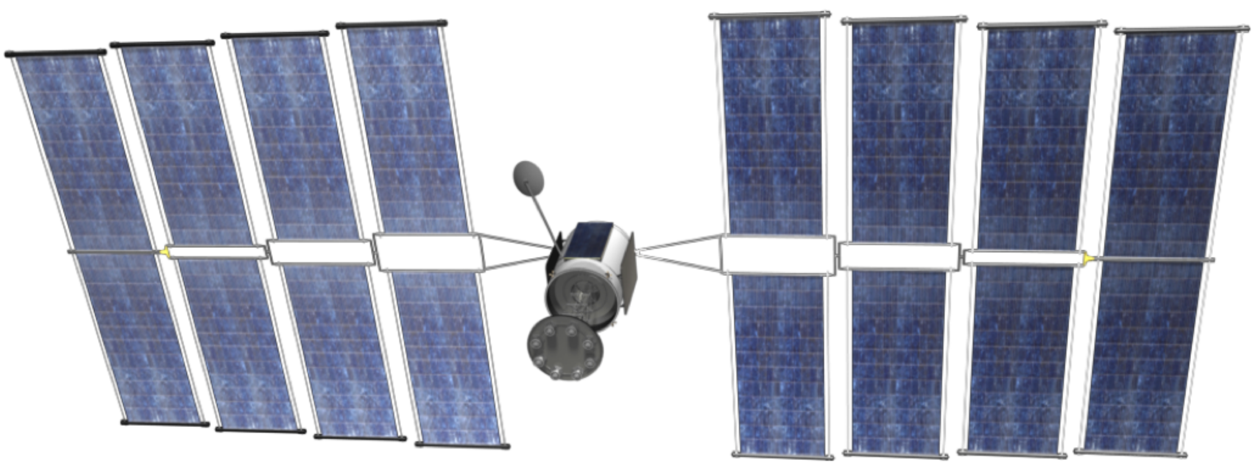

Figure 12. Conceptual design of a 300 kW SEP vehicle for human NEO missions. The wingspan of the solar arrays is 66 m.

of the propulsion system show a broad optimum in the number of thrusters for a given system power, with individual thrusters operating at 20-50 kW as a reasonable choice for vehicles up to 500 kW in power.[56] The HEFT study assumed a total of 7 operating thrusters and one spare for the 300 kW vehicle, each operating at 43 kW. The AISP Project is focused on developing the technologies required for these applications. In parallel, a flight demonstration of the key system elements at a power level on the order of 30 kW is under consideration by the Exploration Technology Development Program and the Crosscutting Capability Demonstrations Division. Five industry study contracts to help define a demonstration mission concept have just been awarded. The technology development efforts associated with human exploration missions are described in this section.

V.A. High Power Hall Thruster Development

High power Hall thruster development in 2010 and 2011 under the new AISP project leveraged 20 and 50 kW-class thruster hardware developed at NASA GRC under previous programs.[22]

NASA GRC demonstrated excellent performance on both krypton and xenon with the NASA-300M, nominally a 20 kW-class thruster designed and fabricated under the ESMD ESR&T program in 2005.[57] The NASA-300M was operated at power levels between 5 and 20 kW and discharge voltages up to 600 V. Peak thrust and total efficiency of 1.13 N and 67%, respectively, were obtained at 20 kW and 400 V. The peak total specific impulse of 2,916 s was demonstrated at a discharge voltage of 600 V and 20 kW. With krypton propellant a peak thrust of 0.90 N was obtained at 20 kW. As expected the thrust to power ratio was lower with krypton than with xenon due to its lower atomic mass. A peak total thrust efficiency and specific impulse of 63% and 3,220 s, respectively, were achieved at a discharge voltage of 600 V and 20 kW.

The Hall2De code developed by JPL is being used by the JPL and GRC team to model the discharge plasma of the NASA-300M and to develop a magnetically-shielded, long-life version, which involves changes to the magnetic circuit and insulator geometry.[22] NASA GRC is developing an extensive suite of plasma diagnostics to help validate the code. These include near- and far-field plume diagnostics such as Langmuir-, emissive-, and Faraday probes, ExB charge state sensors and retarding potential energy analyzers, as well as high speed Langmuir and emissive probes to measure plasma density, potential, and electron temperature inside the discharge channel.

Figure 13. Photograph of the 457Mv2 thruster.

Preliminary performance testing of the NASA-457Mv2 thruster was also conducted at GRC in 2011.[22] The NASA-457Mv2 is an improved version of a 50 kW-class thruster whose development was initiated in 2000 for the NASA Space Solar Power Concept and Technology Maturation Program to enable space solar power systems and other high power spacecraft. The first laboratory version of the 50 kW thruster, designated NASA-457M, had a 457 mm outer diameter ceramic discharge chamber and a centrally-mounted high current hollow cathode. It was fabricated in 2002 and tested on xenon and krypton at power levels up to 72 kW and demonstrated anode efficiencies as high as 65% and specific impulse values as high as 3050 s.

The In-Space Propulsion Transportation Program funded the development of the NASA-457Mv2 thruster, which retained the same discharge chamber critical dimensions, but incorporated an improved magnetic circuit and a new mechanical design which eliminated deficiencies with respect to anode mounting and electrical isolation, concentricity, and thermally induced mechanical interferences. The thruster, shown in Fig. 13, was fabricated in 2004, but funding was terminated before performance testing could be completed.

In tests recently completed under the AISP project, thruster power was varied from 5 to 25 kW over a 200-500 V discharge voltage range. Preliminary results indicate that at a total thruster operating power of 25.2 kW and a discharge voltage of 500 V, the NASA-457Mv2 generated 1.28 N of thrust and had a total thrust efficiency and specific impulse of 63% and 2,520 s, respectively. This is a significant performance gain over the NASA-457M thruster, indicating improved design and fabrication approaches.

Planned activities with the NASA-300M and NASA-457Mv2 include additional performance and thermal measurements, characterization of the plasma properties in the near- and far-field plume of the thrusters to help validate physics-based models of thruster operation, application of those models to estimate the propellant throughput capability, and structural and thermal modeling.[22]

V.B. High Current Cathode Development

High power Hall thrusters in the 20 to 100 kW range will require cathodes capable of operating at 50 to 400 A with lifetimes in excess of 10000 hours. High current lanthanum hexaboride (LaB_6) and dispenser hollow cathodes are under development to satisfy these requirements at JPL and GRC, respectively. LaB_6 cathodes have been used extensively in space on Russian Hall thrusters and in many high current industrial applications. LaB_6 cathodes operate at a higher temperature for a given emission current density than BaO dispenser cathodes, but offer very high current capability, long life due to low evaporation rates, and orders of magnitude less sensitivity to oxidizing impurities and air exposure. BaO dispenser cathodes have been used in most Hall and ion engines in the West and may also offer long life at higher current levels with barium recycling and emitter configurations that provide increased emitter area.

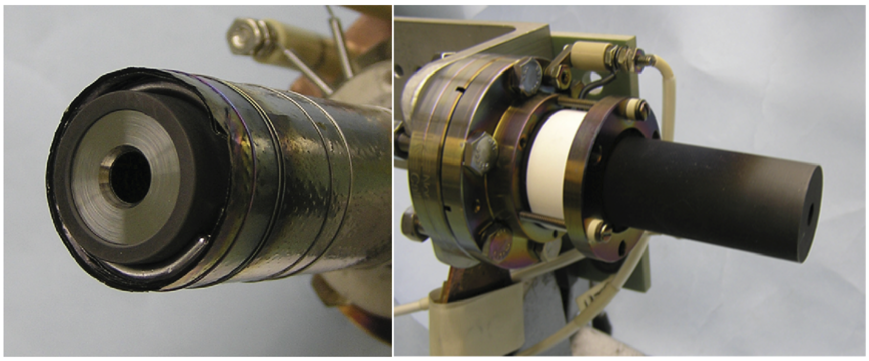

Figure 14. High current LaB_6 cathode with a 1.9 cm diameter emitter has been operated at up to 250 A.

A 60 A hollow cathode with a LaB_6 insert outer diameter of 0.64 cm and an inner diameter of 0.38 cm was previously developed at JPL for the H6 Hall thruster.[58] This cathode demonstrated long life heaters and the use of graphite sleeves or tubes to act as a diffusion barrier and prevent reactions between the LaB_6 insert and refractory metals. LaB_6 cathodes with insert outer diameters of 1.27 cm and 1.9 cm based on the H6 design (Fig. 14) were fabricated and operated at up to 250 A to demonstrate high current capability, high power heater reliability, ease of handling, and resistance to propellant impurities. Internal measurements of plasma density, potential, and electron temperature were obtained at 100 A to help validate the OrCa2D cathode code developed at JPL.[59,60] Wear tests up to 300 hours in duration revealed negligible erosion of the inserts, and indicate that the primary technical issues are thermal management of the cathode orifice plate and sputtering of downstream components due to production of high energy ions. Current efforts are focused on resolving the orifice plate heating with relatively straightforward changes to the thermal design, further characterization of the high energy ion production and methods for mitigation, and extended wear testing.

Research and development at NASA GRC is focused on high current dispenser cathodes and will include design and testing of laboratory model cathodes at emission current levels up to 150 A.[22] These cathodes will employ impregnated porous tungsten emitters with novel geometries and various cathode orifice plate and keeper orifice plate configurations. A flight-like unit based on an optimized configuration will be characterized and subjected to a long duration wear test. Preliminary experiments of a 1.27 cm diameter cathode with a novel emitter configuration that has enhanced emission area diameter cathode demonstrated orifice plate temperatures 50-100 °C cooler than the state-of-the-art designs, yielding longer emitter life.[22]

V.C. Power Processing Systems

The PPUs required for high power SEP vehicles face a number of significant development challenges. The 300 kW vehicle study assumed the use of PPUs with 43-kW input power, 250 to 350 V input voltage, 95% efficiency, 60 °C baseplate temperature, and a mass of 80 kg. This represents about an order of magnitude more power handling capability and twice the input voltage of existing PPUs. To meet these requirements new solid state components such as diodes and transistors will have to be developed and qualified, because state-of-the-art radiation-hardened flight parts do not have adequate voltage capability. NASA GRC is developing models to predict converter mass and losses to support trade studies of module power level, transistor choice, switching frequency, circuit topology and input/output voltage. In addition, commercial-grade, non-flight silicon components and new solid state component technologies that may enable high power, high voltage operation, such as silicon MOSFETs, IGBTs, SiC MOSFETs and GaN MOSFETs, are being tested to characterize performance and switching and conduction losses.[22]

Even without the additional challenges of developing and qualifying new parts, PPUs are expensive and time-consuming to develop. High voltage arrays, required to minimize the mass and losses in conductors, offer the potential for simpler, direct drive architectures. In this approach the operating voltage of the Hall thrusters is matched to the array output voltage, eliminating the need for high power DC to DC discharge converters. Solar arrays operate most efficiently with a DC current output at the peak power point. Hall thrusters, however, are subject to large amplitude discharge current oscillations, so a direct drive system would consist primarily of filtering, switches to isolate thrusters from the array, and low power auxiliary supplies for cathodes and magnets. If direct drive can be successfully developed, it would result in a significantly higher efficiency system, yielding a slightly smaller solar array and significantly less waste heat. The simpler system could reduce PPU mass by up to a factor of 4, with additional savings in structural, thermal management, and array mass.[23] Direct drive systems are potentially much easier to develop than conventional PPUs, reducing development cost, time, and risk. Direct drive has been studied at power levels up to one kW,[61–64] but successful startup and stable operation at high power with multiple thrusters must be demonstrated. To address these issues, JPL is developing a direct drive testbed consisting of a 12 kW commercial solar array system coupled to a large vacuum test facility. This testbed will be used to operate a single H6 Hall thruster and two 5 kW Hall thrusters provided by NASA GRC.

V.D. Advanced Concepts

The AISP project also includes development of lower TRL propulsion concepts with a particular emphasis on those technologies that are scalable to very high power levels. Potential applications for these technologies include more ambitious human missions that will require higher system power levels, such as fast piloted Mars missions. The portfolio includes research on high power magnetoplasmadynamic (MPD) thrusters and pulsed inductive thrusters as well as collaboration with the Air Force on high power nested Hall thrusters. Progress on pulsed inductive thruster research at the NASA MSFC is summarized in this section as an example of the advanced concepts work.

A recent review of pulsed inductive thruster research and technology[65] has served as the baseline for the work at NASA-MSFC, with present efforts centered on advancing the TRL of these thrusters. The thruster design presently under study is the Microwave Assisted Discharge Inductive Plasma Accelerator (MAD-IPA).[66,67] The thruster is based around a conical theta pinch design, with sets of coils connected to a capacitor bank that is quickly switched to provide a high discharge current in the coil. Microwaves are used to preionize propellant, providing seed plasma that reduces the energy required for the primary pulsed discharge to form and accelerate an inductive current sheet. This reduction in required energy permits the use of off-the-shelf solid-state technology (multiple silicon thyristors in a series/parallel configuration) to switch the high current in the thruster.

The present design has been manufactured using rapid-prototyping techniques, with the thruster fabricated in three separate conical layers. The thruster has been fabricated from a high temperature thermoplastic, which won't survive long with power applied to the thruster but should be useful in permitting single pulse and short repetition rate operation. Current trace pathways for the conical coil and gas distribution channels were designed into the layers, and the inner cone has been spray-coated with boron nitride to increase the survivability. The thruster is shown mounted on a hanging pendulum arm in the left frame of Fig. 15, with an end-on photograph of the microwave-produced plasma shown in the right frame. Performance testing of the rapid-prototype thruster on the thrust stand in single pulse mode and for short bursts at a repetition rate commensurate with several kW of input power is presently planned. A thrust stand pulsed calibration apparatus has been fabricated and incorporated into the thrust stand design to support this testing, permitting calibration for both single pulse and repetitive burst-pulse mode operation.[68] In parallel with experimental efforts, modeling efforts using a well developed one-dimensional plasma acceleration model coupled to a set of circuit equations have been undertaken. One objective is to understand the plasma pinching process in a conical theta pinch pulsed inductive accelerator. The initial results of this effort indicate it is extremely unlikely that much of the energy invested in compressing the plasma can be recovered as directed axial motion of propellant, implying that a flat plate design with the electromagnetic force primarily directed axially may be superior to any pinch-based accelerator.[69] A second effort has focused on the inclusion of an energy equation into the one-dimensional acceleration model, which permits the tailoring of the model to different gases when coupled with a real-gas equation of state, ionization model, resistivity model, and model for the specific heat ratio. This modeling enhancement obviates the need to use more detailed 2-D computational fluid solvers, reducing the time and effort needed to model the accelerator.

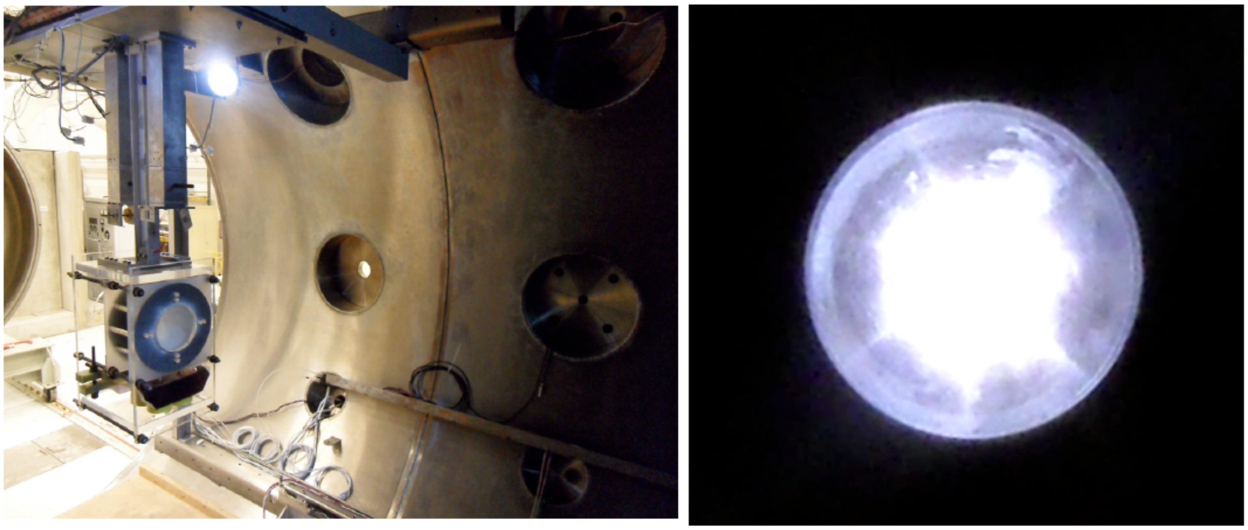

Figure 15. MAD-IPA thruster mounted in a vacuum chamber on a thrust stand (left). Microwave preionization plasma (on nitrogen) in the MAD-IPA (right).

Recent results show good agreement with quantitative performance data for a thruster operating on argon propellant.[70]

VI. Conclusions

NASA has initiated a healthy electric propulsion technology development effort through a number of different programs. The work spans the range of fundamental thruster physics investigations to flight implementation on the Dawn spacecraft. New products under development include the NEXT ion thruster and the HiVHAc Hall thruster for deep space missions, commercial systems adapted for use in interplanetary space, colloid microthrusters for precision pointing and disturbance reduction on astrophysics missions, and high power thrusters for human exploration.

A. Recent NASA Small Business Innovation Research (SBIR) and Technology Transfer (STTR) Contracts

2009 Phase 1 and 2 SBIR Contracts:

1. Arkansas Power Electronics International, Inc., "Silicon Carbide (SiC) Power Processing Unit (PPU) for Hall Effect Thrusters," (09-2-S3.04-8940)
2. Busek Co., Inc., "Miniaturized Low-Power Piezo Microvalve for NanoSat and CubeSat Propulsion," (09-2-S2.01-9277)
3. Busek Co., Inc., "Low Mass Low Power Hall Thruster System," (09-2-S3.04-8131)
4. Busek Co., Inc., "High Efficiency Hall Thruster Discharge Power Converter," (09-2-S3.04-8809)
5. Colorado Power Electronics, Inc., "Low-Cost High-Performance Hall Thruster Support System," (09-2-S3.04-8077)
6. Microcosm, Inc., "Unibody Composite Pressurized Structure (UCPS) for In-Space Propulsion," (09-2-S3.04-9659)
7. MSNW, LLC, "Low Mass Electromagnetic Plasmoid Thruster with Integrated PPU," (09-2-S3.04-9095)

2010 Phase 1 SBIR Contracts:

1. Advanced Cooling Technologies, Inc., "High Temperature Radiators for Electric Propulsion Systems,"

(10-1-X2.04-9362)

2. Aerophysics, Inc., "Self-Healing Field-Emission Neutralizers for Electric Propulsion," (10-1-S3.04-8303)
3. Busek Co., Inc., "Dual Mode Low Power Hall Thruster," (10-1-S3.04-8134)
4. Busek Co., Inc., "20mN, Variable Specific Impulse Colloid Thruster," (10-1-S3.04-8878)
5. e-beam, inc., "Reservoir Cathode for Electric Space Propulsion," (10-1-X2.04-9863)
6. Eltron Research and Development, Inc., "Inorganic Nanostructured High-Temperature Magnet Wires," (10-1-X2.04-9626)
7. Espace Inc., "Precision Electrospray Thruster Assembly (PETA)," (10-1-S2.01-8167)
8. MSNW, LLC, "A Modular Electric Propulsion System with On-Demand Power Scaling," (10-1-S3.04-8645)
9. Starfire Industries LLC, "Hall-Effect Thruster Modifications for Dual-Mode Electric Propulsion," (10-1-S3.04-8822)
10. Tech-X Corporation, "Predicting Hall Thruster Operational Lifetime Using a Kinetic Plasma Model and a Molecular Dynamics Simulation Method," (10-1-S3.04-9270)
11. Tethers Unlimited, "High Thrust Efficiency MPD Thruster," (10-1-X2.04-9542)

2009 Phase I STTR Contracts:

1. ElectroDynamic Applications, Inc. and the University of Michigan, "High Power High Thrust Ion Thruster (HPHTion): 50 CM Ion Thruster for Near-Earth Applications," (09-1-T3.01-9881)
2. Tech-X Corporation and Wright State University, "Next-Generation Ion Thruster Design Tool to Support Future Space Missions," (09-1-T3.01-9893)

2010 Phase I STTR Contracts:

1. Busek Co., Inc. and The Pennsylvania State University, "Magnesium Hall Thruster for Solar System Exploration," (10-1-T3.01-9872)

B. Acknowledgments

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