PROGRESS IN TECHNOLOGY VALIDATION OF THE NEXT ION PROPULSION SYSTEM

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ABSTRACT

The NASA's Evolutionary Xenon Thruster (NEXT) ion propulsion system has been in advanced technology development under the NASA In-Space Propulsion Technology project. The highest fidelity hardware planned has now been completed by the government/industry team, including a flight prototype model (PM) thruster, an engineering model (EM) power processing unit, EM propellant management assemblies, a breadboard gimbal, and control unit simulators. Subsystem and system level technology validation testing is in progress. To achieve the objective Technology Readiness Level 6, environmental testing is being conducted to qualification levels in ground facilities simulating the space environment. Additional tests have been conducted to characterize the performance range and life capability of the NEXT thruster. This paper presents the status and results of technology validation testing accomplished to date, the validated subsystem and system capabilities, and the plans for completion of this phase of NEXT development.

INTRODUCTION

NEXT as an integrated project is comprised of the development of an advanced xenon ion thruster, a power processor unit, xenon feed system, a gimbal, and the control algorithms for system operation.[1] The NEXT project phase 2 develops flight-like engineering model components, with sufficient performance, functional, environmental and integration testing, with life analysis and test, to validate the technology approach and hardware design. The NEXT team is composed of NASA Glenn Research Center (GRC), the Jet Propulsion Laboratory (JPL), Aerojet General Corp. and L-3 Communications Electron Technologies Inc. The NEXT project is being conducted under the NASA Science Mission Directorate In-Space Propulsion Technology (ISPT) project, which is managed by the Glenn Research Center.

NEXT is an advanced ion propulsion system oriented towards robotic exploration of the solar system using solar electric power. Potential mission destinations that could benefit from a NEXT Solar Electric Propulsion (SEP) system include inner planets, small bodies, as well as outer planets and their moons when chemical or aerocapture approaches are used to capture at the destination body. This range of robotic exploration missions generally calls for ion propulsion systems with deep throttling capability and system input power ranging from 5 to 25 kW, as referenced to solar array output at 1 Astronomical Unit (AU).

The selection process for NASA robotic science missions can be characterized as highly competitive, whether selected through a directed process or formal competition. A proposal implementing advanced technologies for a future mission can make or break the mission concept. In some concepts, a technology may enable the fundamental science breakthrough; in others, the technology may be considered too risky to implement within the mission budget and schedule constraints. It is therefore imperative that advanced technologies are well characterized prior to full consideration for a mission. That characterization consists of analyses and testing to demonstrate system level validation in relevant environments, or Technology Readiness Level 6 (TRL6). Past NASA Discovery, Mars Scout and New Frontiers Announcements of Opportunity have dictated that TRL6 be demonstrated by the Confirmation Review at the end of the project Phase B, and that the path to accomplish such be fully described in the mission proposal.[2,3,4] This paper presents the status and results of technology validation testing accomplished to date, the validated subsystem and system capabilities, and the plans for completion of this phase of NEXT development.

RESULTS AND DISCUSSION

The NEXT project products, in a representative system configuration illustrated in Figure 1, consist of a prototype model (PM) ion thruster, an engineering model (EM) Power Processing Unit (PPU), EM xenon feed system High Pressure Assembly (HPA) and Low Pressure Assemblies (LPA), a breadboard gimbal and a Digital Control Interface Unit (DCIU) Simulator that is comprised of EM-level HPA and LPA control cards and system control algorithms. These subsystems were developed under requirements specified at the ISPT project level, and the NEXT project level flow-down requirements that resulted. Project validation activities, including tests, inspections and analyses, are performed against these requirements. The following sections summarize the validation status and completion plans at the subsystem and system level. The detailed results of validation analyses and tests are described in referenced documents.

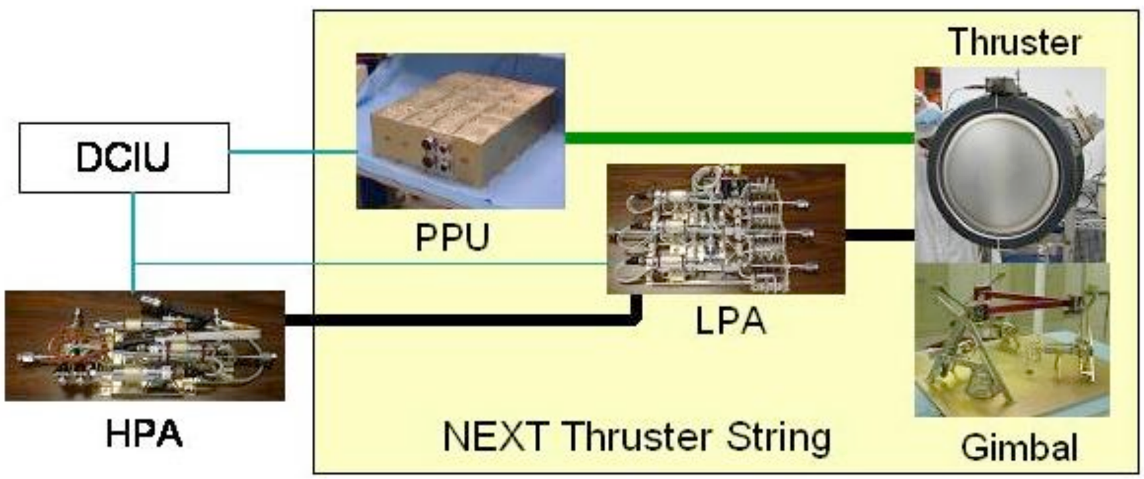

Fig. 1. NEXT ion propulsion system elements

ION THRUSTER

The NEXT ion thruster was developed through a two-phase approach. The initial design concept was developed at NASA GRC and validated through fabrication and test of five Engineering Model (EM) thrusters. The thruster concept was then transferred to Aerojet for implementation in the PM thruster design and hardware. The first Aerojet thruster article, PM1, was delivered to NASA Glenn Research Center in January 2006. Key validation activities include performance acceptance testing, environmental analysis and testing, and life analysis and testing.

PERFORMANCE ACCEPTANCE TESTING

PM1 thruster performance acceptance testing (PAT) was conducted at the NASA GRC Vacuum Facility 6 (VF6) after initial acceptance inspections. The PM1 thruster, as received, was a high-fidelity article and fully met the project objectives in transitioning the thruster design to industry production. The thruster successfully met the performance objectives at all throttle points tested. The thruster required only one mechanical rework, a significant improvement over the initial transition of the NSTAR thruster to industry, after initial testing indicated that the discharge cathode insert was operating at too low of a temperature. A minor modification was made to the cathode to reduce heat loss through the assembly. Performance data collected during the PAT as summarized in Table 1 with comparison to project requirements, demonstrated nominal performance over all throttle points tested. [5]

As a practice, the thruster PAT, or a subset thereof, is performed prior to and after each major test sequence. Performance Acceptance Tests performed throughout the overall thruster test program are routinely compared to the baseline performance data, as established by extensive performance testing of multiple EM thrusters and the PM PAT described above, to assess thruster health.

Beam Voltage, V	**Beam Current, A**	**Input Power, kW**		**Thrust Efficiency**		**Thrust, mN**		**Specific Impulse, s**	
		Req't	**PM1**	**Req't**	**PM1**	**Req't**	**PM1**	**Req't**	**PM1**
1800	3.52	6.860	6.86	0.708	0.710	236.4	237	4188	4190
1179	3.52	4.707	4.70	0.677	0.680	191.6	192	3393	3400
1800	2.00	NS	3.99	NS	0.710	NS	134	NS	4310
1179	2.00	NS	2.75	NS	0.675	NS	108	NS	3490
1800	1.20	2.439	2.46	0.645	0.642	80.2	80.4	3999	4000
1179	1.20	1.704	1.71	0.606	0.605	65	65.1	3240	3240
679	1.20	NS	1.13	NS	0.523	NS	49.2	NS	2450
275	1.00	0.538	0.55	0.325	0.322	25.5	25.5	1401	1400

NS: Requirement not specified at this throttle point

Table 1. PM1 Thruster meets performance requirements (beginning-of-life) across tested points

THERMAL MODELING AND DEVELOPMENT TESTING

The NEXT thruster thermal characteristics have been thoroughly evaluated through a combination of modeling, analysis and development testing. A high fidelity thermal model of the NEXT PM ion thruster has been developed.[6] This thermal model was validated through the thermal testing of the thruster under various throttling levels and environmental conditions during a Thermal Development Test. This model has also been used to predict thruster temperatures in various mission scenarios, and supports assessment of the thermal impact of any design changes and spacecraft integration. The results from the thermal model were used to establish the thermal environmental requirements necessary for testing. The thermal model consists of two different types of models that are necessary to predict the thruster temperatures. The first model predicts the heat flux from the thruster plasma during operation at various throttling points. The results from the plasma model heat fluxes are used as an input into the second model. This second model is a traditional finite-difference code that predicts the thruster temperatures based on the thruster thermal conductivity and surface radiation properties and also predicts the environmental heat fluxes based on orbital parameters.

The Thermal Development Test (TDT) of the PM1 thruster was performed at the JPL Patio Vacuum Facility. [7] The driving thruster thermal requirements are based on a deep space design reference mission, in which a multi-thruster system operates at full power on a Venus gravity assist trajectory through the inner solar system. The associated solar heat flux was determined analytically; the spacecraft geometry was assumed to have the thrusters enclosed in an adiabatic shell to prevent heat flux back to the spacecraft. This system and environment was simulated at an individual thruster level with a thermal shroud, in which heat lamps were mounted to simulate solar heat flux to a single PM thruster, as illustrated in Figure 2. Prior to the TDT, the PM1 thruster was disassembled and instrumented with more than 40 thermocouples to measure temperatures at critical components and locations.

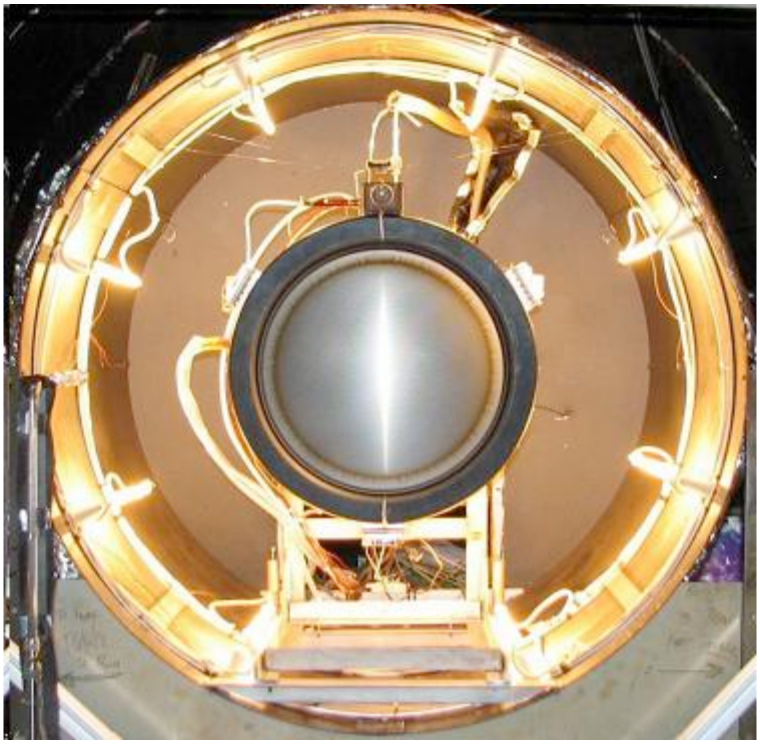

Fig. 2. PM1 thruster installed in thermal shroud for thermal development testing

The objectives of this test were to collect data necessary to validate the thruster thermal model and to establish reference temperatures for subsequent thermal/vacuum cycle testing. The thruster was first operated at various throttle conditions to collect data on self-heating to support model development. The worst-case self-heating thruster throttle point is at 1179 V beam voltage, 3.52 A beam current, at which the discharge losses are at a maximum. The thruster was then operated under an external heat flux of 650 W representing the deep space design reference mission. This condition determined the reference temperature, 183 °C measured at one of the three gimbal pads on the thruster body, for expected nominal hot bias conditions. Finally, the external heat flux was increased to 1000 W to evaluate thruster design margins. Testing was also performed at cold bias conditions, with thruster start-up and operations after cold soak, to evaluate thruster response to cold conditions. Thruster PAT was successfully performed after the TDT, demonstrating negligible changes in thruster operating characteristics.

The thruster thermal model was finalized using TDT data. The model now correlates very well with the test results, with temperatures generally matching within 5 °C. The thruster has substantial thermal design margins on all critical engine components. In Table 2, margins are shown with respect to the worst-case hot analysis results for two configurations. The thruster plasma screen open area fraction (OAF) on the thruster as tested was 50%. The final thruster configuration will have a solid plasma screen on the cylindrical section and a 20% OAF screen on the cone. The TDT identified one potential design issue, with the temperatures on the surface of the external wire harness near the thruster pass-through exceeding the rated temperature of the outer harness wrap at high temperature operations. The project is investigating alternative configurations to mitigate this issue.

Beam Voltage	Maximum Allowable Temperature	1179 V	Margin	1179 V	Margin
Beam Current		3.52 A		3.52 A	
Discharge Power		475 W		475 W	
Cylindrical Plasma Screen OAF		50%		0%	
Conical Plasma Screen OAF		50%		20%	
Front Magnets	360°C	279°C	≥81°C	286°C	≥74°C
Cylindrical Magnets	360°C	239-255°C	≥105°C	250-266°C	≥94°C
Conical Magnets	360°C	257-258°C	≥102°C	264-265°C	≥95°C
Cathode Magnet	360°C	280-283°C	≥77°C	286-288°C	≥72°C
Propellant Isolator	265°C	187°C	≥78°C	195°C	≥70°C
Optics Harness	260°C	193°C	≥67°C	210°C	≥50°C
Titanium Mounting Ring	*	211-275°C	*	223-276°C	*
Screen Grid Support	*	251-270°C	*	259-270°C	*
Accel Grid Support	*	267-304°C	*	276-305°C	*
Gimbal Pads	*	132-187°C	*	141-195°C	*
Exit Wire Harness	150°C	194°C	-44°C	199°C	-49°C

*margin well exceeds 100°C

Table 2. Thermal margins at tested (50%) and final (0/20%) plasma screen configurations

STRUCTURAL ANALYSIS AND VIBRATION TESTING

The structural dynamic environment requirements were specified on the thruster/gimbal assembly to most accurately represent the expected configuration and response. Requirements were developed by JPL to encompass the range of Delta II, Delta IV Medium, and Atlas 5 launch vehicles. Aerojet completed a finite element structural model during thruster design. The breadboard gimbal underwent a complete structural design and was capable of full environmental testing. Dynamic analyses were performed separately for the thruster and gimbal, with assumed interface conditions. The gimbal was vibration tested with a thruster mass model, as described in a following section, to evaluate gimbal design capability prior to thruster testing, and to further evaluate the loads at the thruster/gimbal interface.

Thruster/gimbal assembly vibration testing was performed at the JPL vibration test facility.[8] A conservative test sequence was defined, consisting of low level sine sweep, random vibration at PF -18 dB and PF -6dB levels prior to proceeding to the 10 Grms qualification level for 2 minutes for each axis. Sine sweep tests were also conducted after random vibration testing in each axis to assess response changes. The test configuration is shown in Figure 3.

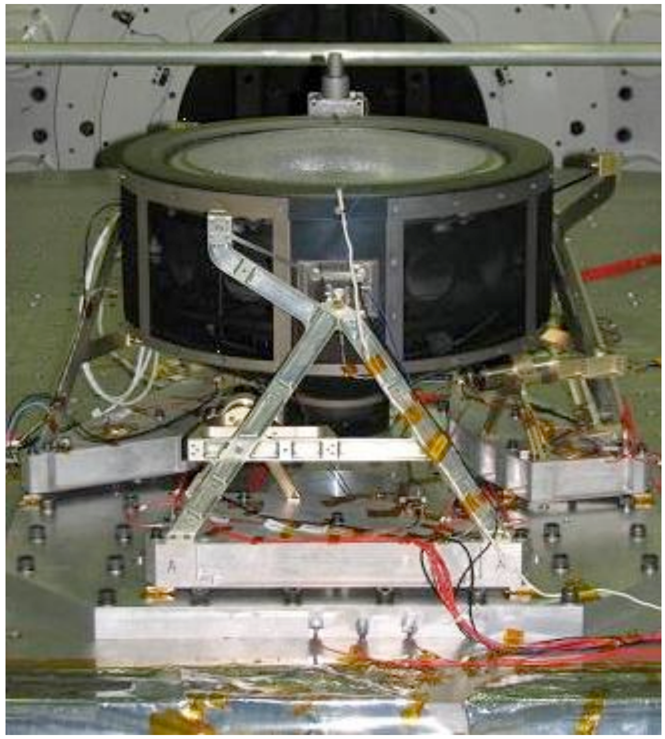

Fig. 3. PM1/Gimbal assembly in vibration test fixture

Vibration testing was completed to qualification levels in all three axes. Overall, the results were very successful. The thruster/gimbal assembly first mode was in the 90 – 100 Hz range, depending on vibration axis. Sine sweeps did not indicate any substantive thruster structural/mechanical state changes, which was later confirmed during thruster inspection and disassembly after environmental testing. The test did identify a number of thruster design and fabrication issues, including two component failures, which are in the process of resolution. The issues uncovered are relatively minor for this stage of the technology development process, and the project has taken the opportunity to resolve them. The issues, and the subsequent response, are summarized below.

Both the neutralizer cathode assembly and discharge cathode assembly suffered mechanical failures at the heater termination joint, resulting in recoverable breaks in the electrical connection and generation of debris from the ceramic potting compound. Subsequent hardware and design evaluation, and analyses, identified specific weaknesses in the mechanical design, including excessive mass and moment arms for components that implement the cathode electrical connection to the thruster top assembly. Design changes have been implemented, and replacement cathode assemblies are currently in fabrication. As these mechanical failures were recoverable; electrical continuity was readily re-established without intervention during functional checks between testing at each axis; the condition was documented and the environmental test sequence proceeded.

A few fasteners on the thruster backed out or lost torque during vibration tests. Upon the initial occurrence, all fasteners were re-checked and tightened to torque specifications as necessary. Analysis of the design and thruster assembly process identified improper fastener selection, through design or part selection, in each case.

Finally, a number of debris/contamination issues were identified. The PM1 thruster had been disassembled and re-assembled multiple times at this point in the test sequence, including thermocouple installation and removal, so the thruster was not expected to meet flight unit cleanliness levels. Post-test inspection revealed excessive assembly debris in the discharge chamber, metallic debris in regions of the discharge chamber assembly, and fibrous debris on the exterior of the discharge chamber (in addition to ceramic potting compound debris previously noted). The metallic debris was traced to the use of non-enclosed self-locking nut plates on the gimbal pads and plasma screen assembly, both of which are locations that encounter frequent fastener removal/installation cycles. The nut plates are being replaced with non-debris-generating parts. The fibrous debris was identified as fiberglass particulates from the thruster harness assembly. An impregnated fiberglass sheath was included on the thruster wires as a carry-over from the NSTAR thruster design baseline. The Aerojet NEXT thruster design, through tight control of wire routing and thermal stand-offs, eliminated the inferred need for the fiberglass sheath. The sheath is being removed from the design baseline. Assembly debris is being addressed through careful evaluation of the assembly and cleaning processes throughout the discharge chamber and top assembly processes.

The project is implementing the above changes, excepting the full assembly process changes, in a modification to the PM1 thruster, identified as PM1R (rework). The PM1R thruster will undergo re-validation performance acceptance and environmental testing after delivery in the summer of 2007.

THERMAL/VACUUM TESTING

A thermal vacuum test of the PM1 thruster was conducted to demonstrate compliance with anticipated mission thermal environments.[8] The test was conducted in the same facility/equipment configuration as the thermal development test described above. The test plan defined the following test sequence and features:

a) Pre-test performance acceptance test at ambient temperature,

b) Three thermal cycles, with thruster starts at both cold and hot bias conditions,

c) Temperature targets of -130 °C and +203 °C, representing qualification level -15/+20 °C margins over the anticipated allowable flight temperatures. The actual cold bias test temperature used was -120 °C to maintain reliable xenon flow control to the thruster,

d) 2 hour cold soaks at each cold bias cycle,

e) Thruster operations at two throttle points for a total of 4 hours at each hot bias cycle,

f) Post- test performance acceptance test at ambient temperature.

Thruster temperatures were monitored at the three gimbal pads and three locations on the front mask, with two of the gimbal pad thermocouples used for temperature control and failsafe. The first two thermal cycles were completed successfully, with all thruster operations and performance nominal. During the third cold bias cycle, the neutralizer heater failed open-circuit when attempting the thruster cold start. After application of environmental heating the neutralizer heater was able to operate. The project decided to terminate the third thermal cycle and proceed directly to post-test PAT. The heater issue was subsequently attributed to the neutralizer cathode heater transition failure encountered during the thruster/gimbal vibration test previously described.

Thruster performance was nominal and consistent throughout the thermal/vacuum test sequence. The thermal/vacuum test demonstrated that the NEXT thruster is compatible with anticipated mission thermal environments.

LIFE VALIDATION

The thruster life requirement of 300 kg xenon throughput was developed through analyses of a variety of NASA robotic space science missions. To meet this mission requirement to a qualification level, throughput capability of 450 kg must be demonstrated. The validation of thruster life capability is comprised of the following elements: thruster long duration testing, component-level long duration testing, and analyses of thruster life limiting and wear mechanisms.

A long duration test (LDT) of the NEXT EM3 thruster is in progress in the GRC Vacuum Facility 16; the test configuration is illustrated in Figure 4.[9,10] The facility is equipped with numerous diagnostics to provide in situ characterization of thruster wear and performance. The EM3 thruster is comprised of an EM thruster with the first set of Aerojet-fabricated PM ion optics, a graphite discharge cathode keeper, and equivalent relevant neutralizer and high voltage propellant isolator configurations to accurately represent the wear mechanisms of a PM thruster. The initial phase of the LDT is being conducted at the full power thruster throttle point of 1800 V beam voltage, 3.52 A beam current, which has been shown through analysis to provide the worst-case wear rates over the primary operating points on the throttle table.

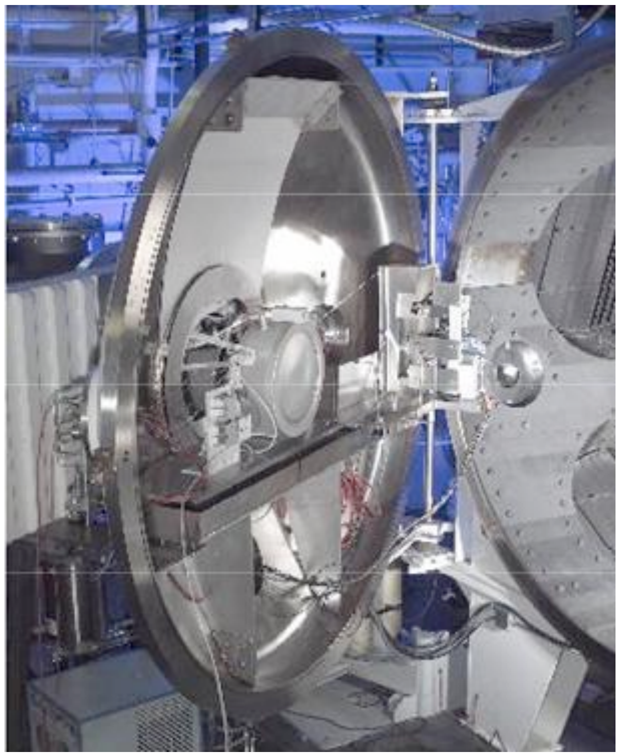

Fig. 4. EM3 Long Duration Test facility and diagnostic configuration

The thruster long duration test is proceeding very well from both the thruster performance and wear perspectives. At the time of publication, the test has exceeded 9000 hours of operation and 185 kg of xenon throughput, with a resulting thruster total impulse exceeding 7.5 x10^6 N-s. This total impulse is the highest ever demonstrated on an ion engine, exceeding that demonstrated in the NSTAR Extended Life Test (ELT). Thruster wear is within expected values and performance is consistent with predicted changes over the life of the thruster.[11] Cathode keeper orifice plate erosion has effectively been eliminated as a relevant wear mechanism. Aperture erosion at the center of the grids (peak beam current) at the full power throttle point is also effectively eliminated. Figure 5 compares center aperture erosion on the NEXT thruster to the NSTAR ELT data. At this time, accelerator grid pit and groove erosion is projected to be the first failure mode. Based on data collected to date, grooves are projected to begin eroding through the accelerator grid after 720 kg of xenon throughput. Accelerator grid through-holes do not define thruster failure, but are indicative of on-coming structural failure of the grid.

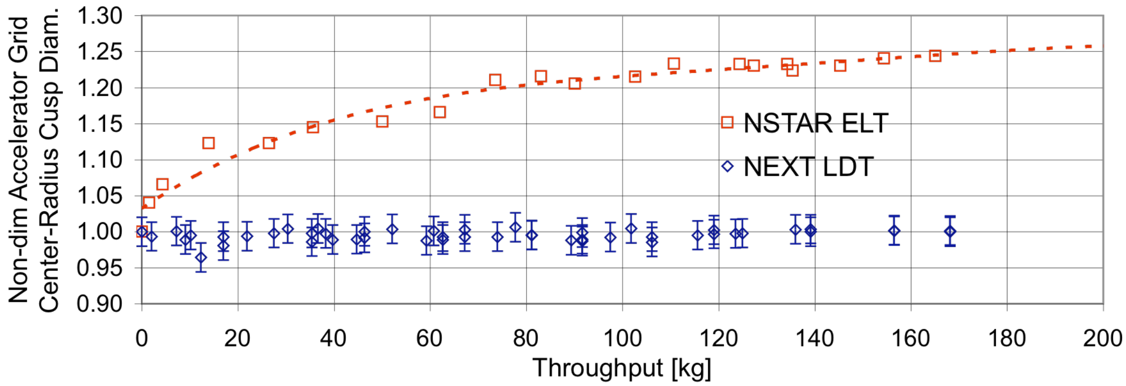

Fig. 5. NEXT center-radius aperture diameter (at full power throttle point), compared to NSTAR ELT (at multiple throttle points)

Per the LDT test planning, thruster operation at full power will continue until 300 kg of xenon throughput is accomplished. At that time, operations at other relevant throttle points are planned to assess wear rates. After accomplishment of 450 kg xenon throughput, operations will resume at full power to maximize wear data collection. The project envisions continuing the long duration test to failure to support the thruster life validation needs of future mission users. Other options are under consideration and are discussed in a following section.

Two component life tests will be performed by the NEXT project. A high voltage propellant isolator (HVPI) life test was initiated in 2006 and has surpassed 9500 hours under test conditions of 265 °C, 2100 V and 32 torr inlet pressure. The test is being performed in the GRC Vacuum Facility 61, using two PM HVPIs provided by Aerojet. To date, HVPI leakage current is within specification; data trends do not indicate any concerns. Cathode heater life testing will also be initiated in 2007. The testing will be conducted on spare Aerojet PM thruster neutralizer and discharge cathode heaters in GRC Vacuum Facility 62.

Extensive analyses have been performed to characterize thruster wear mechanisms and life.[12] Analyses completed as of this publication predict that first thruster failure will occur after 750 kg xenon throughput. When a factor of 1.5 is divided from the test and analyses predictions, thruster life qualification to a throughput of greater than 500 kg is likely. This provides a 66% margin over the project requirement and most demanding mission need identified to date. Such extended capability may be useful in high ΔV, long duration missions, such as sample returns.

POWER PROCESSING UNIT

The NEXT EM power processing unit (PPU) was designed and fabricated by L3 Comm ETI, Inc.[13] At the time of publication, the EM PPU is completing functional acceptance testing. After completion of functional testing on the PPU resistive load, the EM PPU will be delivered to GRC for functional testing with an EM thruster in GRC Vacuum Facility 6. The EM PPU will then be subjected to qualification-level vibration and thermal/vacuum testing. EMI/EMC testing will also be performed to characterize the capability and emissions of the unit.

PROPELLANT MANAGEMENT SYSTEM

The NEXT EM high pressure and low pressure assemblies (HPA, LPA), the primary flow control components of an overall xenon feed system, were designed and fabricated by Aerojet.[14]

Initial validation of the PMS technology was accomplished during breadboard system integration testing in the first NEXT project phase in 2003.[15] EM assemblies were fabricated in two variations, flight-like and non-flight-like. Non-flight-like assemblies use lower cost equivalents of some flight-level parts, such as pressure transducers. The non-flight-like assemblies will be used primarily for integration testing. Key PMS validation activities include performance acceptance and environmental testing, all of which have been completed.

PERFORMANCE ACCEPTANCE TESTING

The NEXT PMS LPA and HPA are flow-calibrated at the assembly level. Performance acceptance testing and calibration of each assembly were accomplished in initial functional testing.[14] Required performance capabilities, including flow rate range and accuracy, were successfully achieved.

VIBRATION TESTING

The PMS structural dynamic environment requirements were specified on the flight-like assemblies to most accurately represent the expected configuration and response. Requirements were developed by JPL to encompass the range of Delta II, Delta IV Medium, and Atlas 5 launch vehicles. Vibration testing of the flight-like HPA and LPA was performed at the Aerojet vibration test facility.[14] A low level sine sweep was performed before and after the random vibration test of each axis. Random vibration tests were performed to the 14.1 Grms qualification level, with a duration of 2 minutes at each axis.

The HPA and LPA vibration tests were completed successfully. First modes were greater than the 100 Hz required. Minor variations in first mode response, between random excitation and sine sweep testing results, were directly attributable to tie downs of the assembly harnesses to the test fixture.

THERMAL/VACUUM TESTING

Allowable PMS assembly flight temperatures, ranging from +27 to +50 °C, are driven by the qualification temperatures of critical flow control components. The HPA and LPA must be controlled within this range, regardless of environmental conditions. Thermal/vacuum testing of flight-like assemblies was conducted to qualification temperatures of +12 to +70 °C to validate successful design.

The PMS assembly thermal/vacuum testing was conducted at Vacuum Facility 12 at Aerojet. The test profile is illustrated in Figure 6. Functional tests (FT) were performed throughout the test sequence and consisted of: outlet pressure control for the HPA, and flow rates, accuracies, and rate change responsiveness for the LPA. In addition, the LPA was tested in a contingency flow control mode during functional testing during two of the temperature cycles. Control thermocouples were located at the assembly interface plate, representative of a spacecraft sense point.

The PMS thermal/vacuum tests were fully successful. The HPA and LPA retained flow control characteristics through cold and hot cycling. The test also demonstrated successful operation in a contingency mode over the full thermal range. The validation testing and analysis of the NEXT PMS has thus been completed.

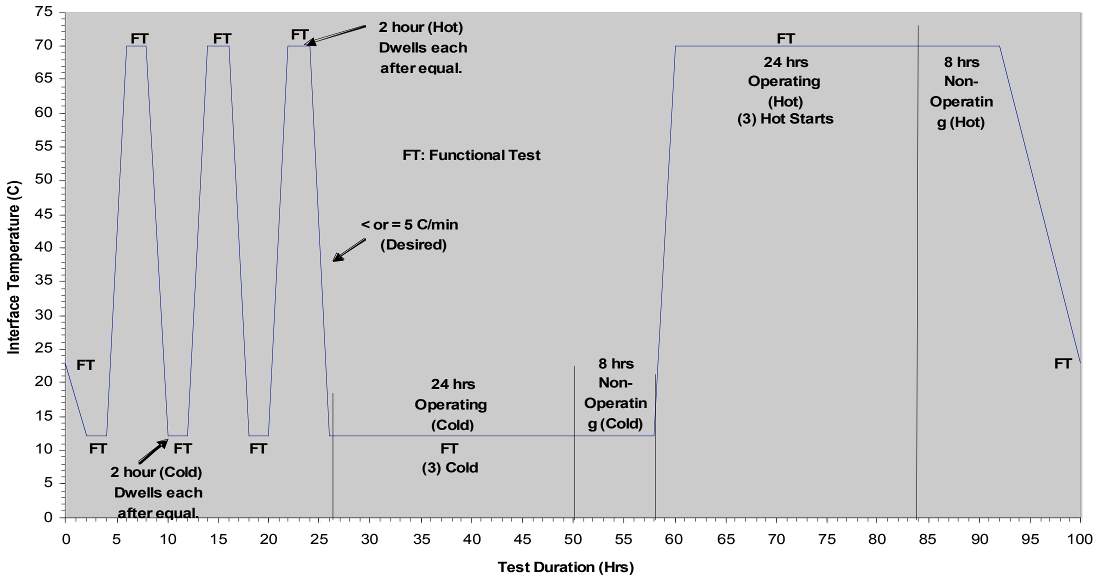

Fig. 6. EM PMS thermal/vacuum test profile

GIMBAL

The NEXT gimbal was designed and fabricated by Swales Aerospace, under contract to the Jet Propulsion Laboratory. [16] One complete gimbal assembly was delivered, with sufficient parts to assemble a second assembly. The gimbal was designed as a flight-packaged engineering unit, but without substantive thermal analysis. Therefore, only functional testing and structural dynamic analyses and testing were performed on the gimbal.

FUNCTIONAL TESTING

Initial testing with a geometric simulation of the thruster was performed to demonstrate functionality and gimbal capabilities. The gimbal provides maximum authority of ±19 ° and ±17° about the primary gimbal axes and a rough cone about the thruster centerline within those boundaries.

Functional testing of the gimbal with the PM thruster integrated was also performed before and after vibration testing. As the gimbal was designed as a micro-gravity unit, a pulley/mass system was used during functional testing to offset the thruster mass. The gimbal maintained full functionality through the various performance and vibration test sequences.

VIBRATION TESTING

The gimbal has undergone two vibration test sequences. The first test was performed with a thruster mass model to evaluate gimbal response prior to integrated thruster/gimbal testing, and to characterize the loads at the gimbal/thruster interface to allow further analysis of thruster structural margins. [17] Limited shock testing was also performed in this configuration. The test was performed at the JPL vibration facility, to qualification levels, and was fully successful. Description of the integrated thruster/gimbal vibration test is provided in the thruster section above. Gimbal functional tests, as illustrated in Figure 7, were executed before and after all vibration tests demonstrated full functionality was maintained.

Fig. 7. Gimbal functional testing with the PM1 thruster

INTEGRATED NEXT SYSTEM

The project has planned to perform three key integrated system tests; a multi-thruster interactions test, a single string system integration test with the most mature project products, and a multi-string system integration test operating up to three thrusters simultaneously. Additionally, plume/plasma modeling has been performed to allow analysis of future thruster configurations and characterization of environmental interactions with the spacecraft.

MULTI-THRUSTER INTERACTIONS TEST

A multi-thruster interactions test was executed to evaluate two areas of system integration concerns; demonstrating that multiple thrusters perform together without degrading individual thruster performance or life, and understanding the plasma environment the spacecraft may be subjected to in single or multi-thruster operations. The test was performed at the GRC Vacuum Facility 6 in December 2005. The primary components of the test included: four EM thrusters mounted on a reconfigurable array, one of which was an instrumented non-operational unit, a gimbal simulator articulating one thruster, laboratory power supplies, laboratory xenon feed systems, extensive plume and plasma diagnostics, and a data acquisition and control system.

Test objectives and sequences concentrated on engineering evaluation of thruster operations,[18] plume characterization and interactions, with variation in thruster spacing and gimbaling effects considered,[19,20] neutralizer performance in multi-thruster arrays,[21] and plasma environments around the thruster array and at the spacecraft interfaces.[22] Additionally, thermocouples were placed in various locations on the array to measure thermal interactions between operating and non-operating thrusters in support of thermal modeling.

The multi-thruster interactions test was fully successful, as reported in the references above, demonstrating:

- multi-thruster performance is adequately predicted by superposition of multiple single thrusters,

- multi-thruster performance is not detectably affected by thruster spacing or gimbaling,
- plasma environments at the spacecraft interface are mild,
- neutralizer performance is not impacted by multi-thruster operations,
- thruster operations were nominal with a single neutralizer operating with multiple thrusters, or thrusters operating in combination with neighboring thruster's neutralizer.

The data resulting from this test has been incorporated into thruster plume/plasma models developed by the SAIC Corporation under contract to JPL.[23] Analyses using this model confirm many of the demonstrated findings above. This model will allow NEXT system users to perform configuration specific analyses to assess system and spacecraft environments and interactions.

SINGLE STRING SYSTEM INTEGRATION TEST

The NEXT project will perform a single string system integration test in 2007 to validate system level functional and performance requirements. System-level functionality was previously demonstrated with breadboard-level units in NEXT Phase 1.[15] The primary components of this test include: the PM1R thruster, the EM PPU, flight-like EM HPA and LPA, the breadboard gimbal, the DCIU Simulator, diagnostics, and support equipment. The thruster/gimbal assembly, PPU, HPA, and LPA will be in a vacuum environment. The test will operate the system in it's full range of operating modes, nominal and contingency, across the system throttle range. This test provides the primary opportunity to validate that the DCIU Simulator control algorithms and PMS control circuitry meet project requirements.

MULTI-STRING SYSTEM INTEGRATION TEST

The project is also planning to perform a multi-string system integration test in fiscal year 2008, pending budget decisions. The objective of this test is to validate that the entire multi-string system functions nominally without interactions. The NEXT system test configuration would consist of the PM1R and two operational EM thrusters, a non-operating instrumented EM thruster, the EM PPU, the breadboard version of the PPU that was fabricated and tested in an earlier stage of the project,[13] the flight-like HPA and LPA and two additional LPAs, the breadboard gimbal and the DCIU Simulator. The third thruster would be powered by a laboratory power supply.

FUTURE NEXT VALIDATION ACTIVITIES

A range of tests and analyses are under consideration for project implementation, pending budget authority. Continuing validation of thruster life offers a number of future options. The PM1R thruster will be available for further testing after completion of the single string system integration test. Conduct of a short wear test on PM1R, with duration sufficient to establish trend data comparable to the EM3 LDT, will improve the correlation of flight thruster hardware projections with the LDT data. The EM3 LDT should be continued to failure; however a second option to replace EM3 with PM1R continues to receive consideration.

In conjunction with other system level testing, the project desires to perform radiated and conducted emissions tests of the operating thruster/PPU string in vacuum. This test would provide important environmental interface data for future users.

A number of analyses and updates have been identified, in which the results of tests described above are fed back into the design analyses for each subsystem. This design/analysis update cycle would provide final definition of the subsequent flight hardware build. Included in these analysis options are life analyses of other elements of the system.

Finally, all parts and subassemblies of the second PM thruster, PM2, are in storage at Aerojet. The opportunity exists to complete assembly of the PM2 thruster for use for unique validation tests associated with a mission.

All project validation events will be documented in a validation/verification report and reviewed at the end of the Phase 2 project.

SUMMARY AND CONCLUSIONS

The NEXT project has performed a thorough range of tests and analyses to validate Technology Readiness Level 6, and to verify that the products meet the project requirements. The tests have predominantly been very successful, with minor reworks required for some subsystems. Several key tests remain, with completion planned in 2007. TRL6 is a key milestone for introducing new technologies into NASA robotic science mission concepts. The NEXT project will have surpassed this milestone prior to initiation of the next rounds of competed mission Announcements of Opportunity, supporting full consideration in mission concept development and proposal.

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