Propulsion Risk Reduction Activities for Non-Toxic Cryogenic Propulsion

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The Propulsion and Cryogenics Advanced Development (PCAD) Project's primary objective is to develop propulsion system technologies for non-toxic or "green" propellants. The PCAD project focuses on the development of non-toxic propulsion technologies needed to provide necessary data and relevant experience to support informed decisions on implementation of non-toxic propellants for space missions. Implementation of non-toxic propellants in high performance propulsion systems offers NASA an opportunity to consider other options than current hypergolic propellants. The PCAD Project is emphasizing technology efforts in reaction control system (RCS) thruster designs, ascent main engines (AME), and descent main engines (DME).

PCAD has a series of tasks and contracts to conduct risk reduction and/or retirement activities to demonstrate that non-toxic cryogenic propellants can be a feasible option for space missions. Work has focused on 1) reducing the risk of liquid oxygen/liquid methane ignition, demonstrating the key enabling technologies, and validating performance levels for reaction control engines for use on descent and ascent stages; 2) demonstrating the key enabling technologies and validating performance levels for liquid oxygen/liquid methane ascent engines; and 3) demonstrating the key enabling technologies and validating performance levels for deep throttling liquid oxygen/liquid hydrogen descent engines. The progress of these risk reduction and/or retirement activities will be presented.

Nomenclature

C*	=	characteristic velocity
EPW	=	electronic pulse width
Hz	=	hertz
in.	=	inch(es)
Isp	=	specific impulse
lb_f	=	pounds force
lb_m	=	pounds mass
L*	=	characteristic length
min.	=	minutes
^oR	=	degrees Rankine
sec.	=	seconds

I. Introduction

THE PCAD Project's primary objective is to develop propulsion system technologies for exploration missions. The PCAD project is funded by the Exploration Technology Development Program in NASA's Exploration Systems Mission Directorate. PCAD has concentrated its activities on non-toxic or green propellants to meet near term Constellation Program decision gates. Implementation of green propellants in high performance propulsion systems offers NASA an opportunity to consider other options than current hypergolic propellants. The PCAD Project is emphasizing efforts in reaction control system (RCS) thruster designs, ascent main engines (AME)

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for lunar missions, and descent main engine (DME) for lunar missions. PCAD has developed the following specific objectives:

- Perform cryogenic and non-cryogenic RCS design, ignition testing, and performance testing
- Perform cryogenic ascent main engine design, ignition testing, and performance testing
- Perform cryogenic descent main engine design and performance testing

II. Liquid Oxygen (LOx) – Liquid Methane (LCH_4) Propulsion

In support of the U.S. Space Exploration Policy for returning to the Moon and beyond, NASA and its partners are developing and testing cryogenic propulsion system technologies that will meet the need for high-performance propulsion systems on long-duration missions. In particular, the lunar ascent module propulsion systems are critical performance drivers, due to the high "gear ratio" (ratio of mass launched to delivered mass to the Moon) associated with elements that are utilized through the late phases of the mission. However, due to the relatively small size of the Ascent Module, multiple propulsion system options exist. System trades for both lunar and Mars missions have indicated that LOx/LCH_4 is a promising option, due to the approximate 600- to 800-lb_m savings in overall systems mass over more conventional hypergolic systems. Because the Ascent Module is taken to the lunar surface, the indicated mass savings would be converted directly to lunar surface payload. LOx/LCH_4 propulsion for Ascent Main and Ascent/Descent Reaction Control Propulsion is currently conceded as a critical enhancing technology, due to the potential increase of lunar surface payload. The primary technology risks, as determined by the PCAD project team, associated with LOx/LCH_4 propulsion are the following:

1. Reliable/ignition pressure fed LOx/LCH_4 Reaction Control Engines (RCE)
2. Meeting minimum performance and life requirements of LOx/LCH_4 RCE and Main Engines with integrated testing
3. Reliable/ignition pressure fed LOx/LCH_4 Main Engine

The PCAD project focus' on the development of cryogenic propulsion technologies needed to provide necessary data and relevant experience to support informed decisions on potential implementation of cryogenic propellants in the Altair architecture

A. LOx/LCH_4 Reaction Control Engine Development

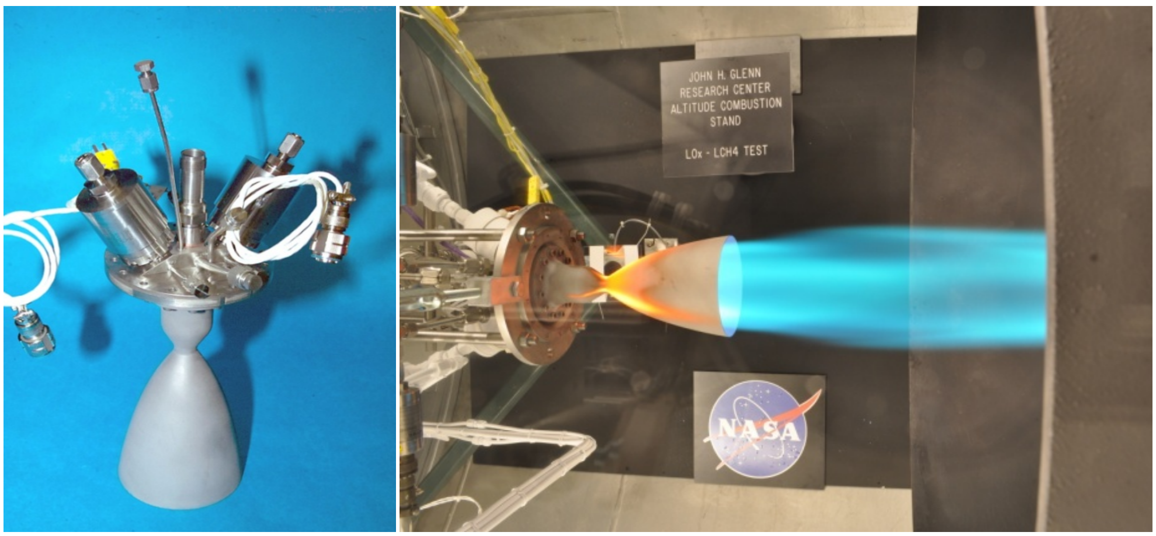
Since 2005, the PCAD project has invested in technologies leading to pre-prototype development of LOx/LCH_4 reaction control engines (RCE) with the release of contract request for proposals (RFPs). The focus of the activities were originally to support the Service Module, however in 2006 the activity was steered to support a lunar lander. The top three risks identified for RCE technology are: 1) reliable ignition; 2) Performance (vacuum specific impulse – Isp); and 3) Repeatable pulse width. To address the risks, PCAD undertook a combination of in-house and contract activities.

In 2006 PCAD awarded two RCE contracts to Northrop Grumman and Aerojet respectively. Each contract was focused on the development and delivery of a 100-lb_f thrust pre-prototype engine subsystem. The key performance requirements in the contracts were: 1) 317-sec vacuum Isp; 2) 4 lb_f-sec minimum impulse bit (Ibit); 3) 80-msec electronic pulse width (EPW); 4) 25,000 valve cycles and 5) operation over a range of inlet conditions from gas to liquid for start. The engine concepts put forward by each company were different in approach to meeting the contract requirements.

Aerojet put forward a concept with foundations in previous work on LOx/ethanol and internally funded activities. The first engines tested were originally LOx/ethanol 870-lb_f thrusters that were modified to accommodate LOx/LCH_4.[1] The modified units were successfully tested on the Auxiliary Propulsion System Test Bed (APSTB) in the NASA White Sands Test Facility (WSTF) Test Stand (TS) 401. The proposed 100-lb_f engine concept consisted of a compact integral exciter/spark plug system, a dual coil direct-acting solenoid valve for oxidizer and fuel, an integral igniter and injector, and a columbium chamber/nozzle with an expansion area ratio of 80:1.

Over the course of several contract option periods, multiple injector patterns were developed and manufactured using Aerojet's platelet technology, Fig. 1(a). Flow control for both the main chamber and igniter were controlled by a single set of dual coil valves. The valves were demonstrated to over 55,000 cryogenic cycles in liquid nitrogen, exceeding the 25,000 cycle life. Ignition was accomplished with the use of a spark torch igniter. Over the duration of the contract, a series of igniter and injector concepts were tested at sea level to examine engine performance. The result of the testing was an impinging injector design that successfully met all key performance criteria either by demonstration or calculations based on test data. Aerojet conducted over 1300 engine pulse tests at a variety of duty cycles for over 1900-sec total of sea level testing during the engine development.[2,3] Specifically Aerojet was able to meet 317-sec Isp calculated based on estimated nozzle losses and exceeded the 80-msec EPW requirement by demonstrating 40-msec EPW. As a result, Aerojet was able to provide five engine units to NASA for multiple engines testing on the APSTB at WSTF and two units for testing at the NASA Glenn Research Center (GRC) in the Altitude Combustion Stand (ACS).

Sea level[4] and altitude performance testing[5] has been conducted at GRC with the Aerojet engines. Figure 1(b) shows the Aerojet engine during test at GRC. A total of 60 altitude hot-fire tests were completed with the Aerojet 100-lb_f LOx/LCH_4 engine and propellant conditioning feed systems (PCFS). [6,7] The PCFS, as shown in Fig. 2, was used to obtain conditions over the range of nominal (204 °R LOx/204 °R LCH_4), cold/cold (160 °R LOx/170 °R LCH_4), to warm/warm (224 °R LOx/224 °R LCH_4). The PCFS uses a combination of cooling loops and heaters to vary the propellant conditions. Test results demonstrated that propellant conditions could be controlled to within ± 5 °R for a given set point. Altitude performance testing was conducted using a 45:1 area ratio columbium radiation cooled nozzle. The main goal of the testing was to develop specific impulse performance curves as a function of mixture ratio. Testing was also conducted over a wide range of propellant inlet conditions (pressure and temperature), to simulate operation in a variety of space environments. The engine demonstrated that meeting the required 317-sec performance is feasible for the 80:1 nozzle based on the results with a 45:1 nozzle.



(a) **(b)**

Figure 1. (a) Aerojet 100-lbf LOx/LCH_4 reaction control engine, (b) Aerojet 100-lbf LOx/LCH_4 reaction control engine in test at GRC.

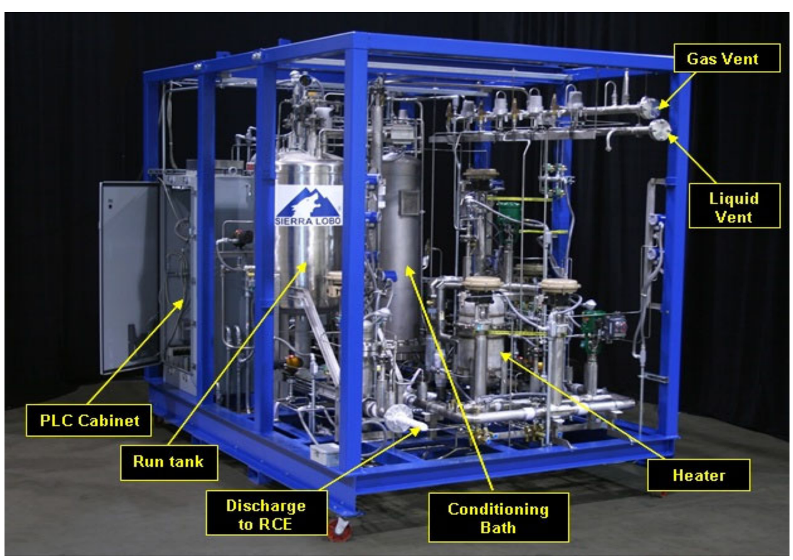


Figure 2. Propellant Conditioning Feed System skid (PCFS) at GRC.

Northrop Grumman put forward a concept with foundations in previous work on hypergol engines. The concept was regeneratively cooled with both oxygen and methane through the combustion chamber and part of the nozzle (Fig. 3).[8] The full engine area ratio (120:1) was completed with a columbium nozzle extension. Flow control for both the main chamber and igniter was controlled by a single set of single coil valves. Ignition was accomplished with the use of a spark torch igniter. A series of hardware configurations were tested, starting with workhorse hardware, to develop the engine cooling circuit. During the course of the contract Northrop Grumman ran into a number of design and manufacturing issues which slowed progress. As a result, budget limitations required changes to the scope of the contract which eliminated the planned four pre-prototype deliverables. However, Northrop Grumman was able to develop a single pre-prototype unit that was tested in vacuum conditions at their Capistrano test facility. Test results indicate that the engine concept was able to meet the performance specifications in the contract, including exceeding the specific impulse requirement. The measured Isp was approximately 331-sec, which exceeded the specification requirement of 317-sec. NASA currently has one pre-prototype unit available for further in-house testing.

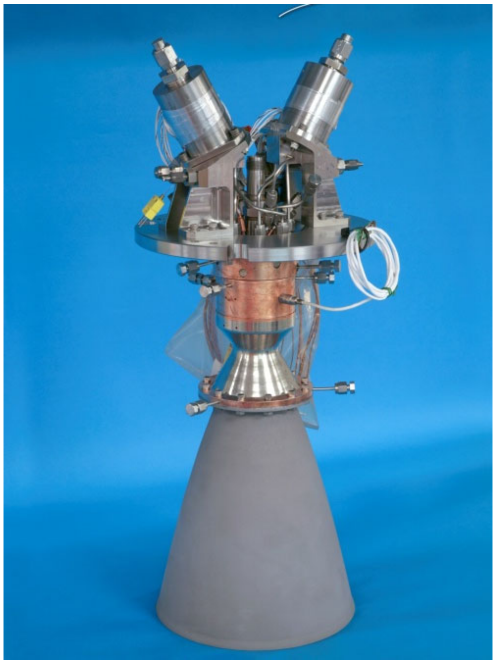


Figure 3. Northrop Grumman 100-lbf LOx/LCH_4 reaction control engine.

B. LOx/LCH_4 Reaction Control Engine Integrated Testing

Once developed, the plan was to integrate the RCE thrusters into a four engine cluster which would simulate a vehicle engine configuration. The Auxiliary Propulsion System Test Bed (APSTB), Fig. 4, at WSTF was modified with a high vacuum bell jar which serves as the engine cluster simulator. In the bell jar all propellant feed lines and valves were mounted in a way similar to a space craft system. The feed system was also fitted with a thermodynamic vent system (TVS) to condition the propellant delivery to the engines. A total of five engines were delivered from Aerojet for the APSTB testing. Engines were installed and tested at each position. Approximately 2500 pulses were conducted over a sequence of 145 tests. In one test, a total of 380 consecutive pulses were completed. Also, an additional 90 pulses were conducted with two engines firing simultaneously. The engines performed as expected, however the testing did uncover issues with the feed system design. A number of tests suffered from high flow spikes or water hammer which resulted in a number of pressure transducer failures. The data is now being used to develop improvements to feed system models. A complicating factor to the feed system was the APSTB design. Because the rig was originally designed for the Space Shuttle systems development, the rig was significantly oversized. As a result, PCAD has undertaken the development of the Integrated Propulsions System Test Bed (IPSTB). The IPSTB, like the APSTB, will be a propulsion system simulator with propellant tanks, feed lines and an engine cluster. However, the IPSTB will be designed with smaller propellant tanks and with the flexibility to change component locations or vary feed line lengths. The goal of the testing will be to examine system interactions with a number of feed system designs and to obtain the data for comparison with state of the art fluid models. Currently the IPSTB will utilize the current inventory of Aerojet and Northrop Grumman engines.

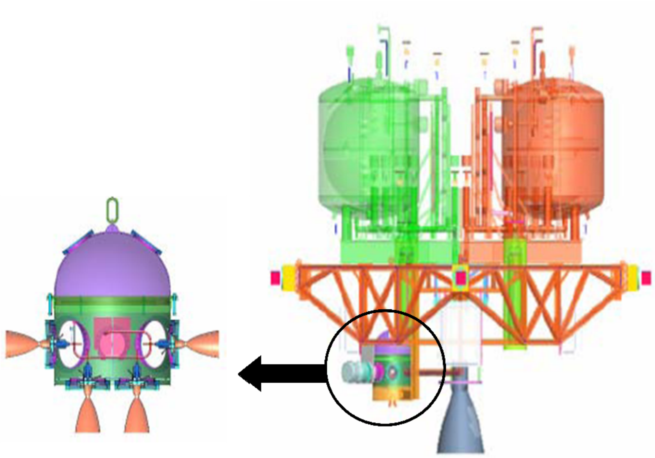
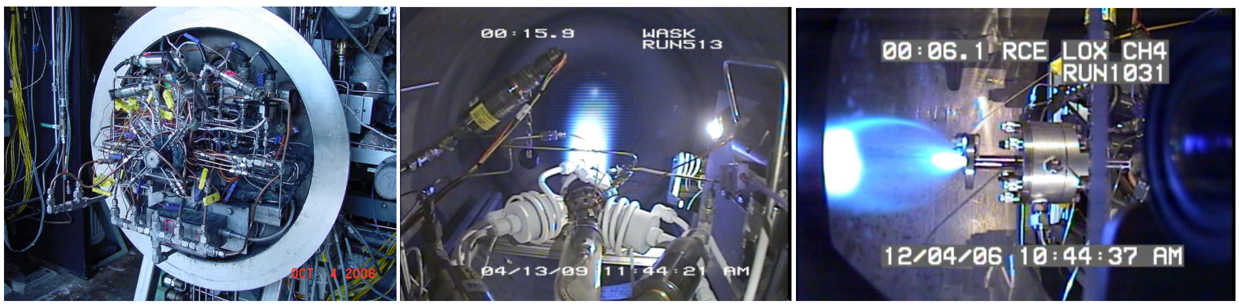


Figure 4. APSTB at WSTF showing RCE and AME test positions.

C. LOx/LCH$_4$ Ignition Risk Reduction

To address the highest risk for LOx/LCH$_4$ propulsion systems, reliable ignition, NASA has conducted numerous in-house experimental efforts to examine the issue. The work has been completed at both RCE and ascent main engine (AME) scales. Figure 5(a) shows the basic altitude test configuration in Cell 21 at GRC. The majority of the work has been conducted with spark torch igniters,[9,10,11,12,13] however there has been work done with microwave,[14,15] piezoelectric, spark torch/glow plug combination,[16] and catalytic ignitions systems. Figure 5(b) shows a spark torch configuration from WASK Inc. and Fig. 5(c) shows a NASA breadboard configuration. Overall there have been no significant issues identified that would prohibit the reliable ignition over a range of conditions with LOx/LCH$_4$. One of the last ignition specific activities completed was the demonstration of 30,000 ignition cycles on a spark torch ignition system at vacuum conditions.[17] Completion of this activity did not identify any issues with the hardware or designs for long duration applications. The work to date has identified issues with spark plug durability and the reliability of power exciter units. In both cases, PCAD has worked additional technology tasks to address the issues. There appear to be viable solutions in work to reduce the risk.



(a) (b) (c)

Figure 5. LOx/LCH$_4$ Altitude Ignition Testing at GRC (a) Test Cell 21 configuration; (b) WASK spark torch igniter during test; (c) ascent main engine class igniter during test.

In particular, advancements have been made on the exciter where Aerojet and Unison have developed a single compact exciter[18] unit, as shown in Fig. 6, to replace the current state of the art exciter box and high voltage power lines. NASA has also successfully completed altitude testing with a compact exciter developed under a Small Business Innovative Research (SBIR) Phase II task with Alphaport, Inc. Many of the issues remaining with LOx/LCH$_4$ ignition are related to the specific requirements and duty cycles that will be imposed on the systems or with the final spaceflight qualification of the units. One general area that would still require investigation is ignition in the cold thermal environment of space where both the hardware and propellants have been exposed to those conditions for a significant period of time before being required to operate.

Figure 6. Prototype Unison compact exciters configured for use with Aerojet 870-lbf RCE.

D. LOx/LCH$_4$ Ascent Main Engine Development

As with RCE, the PCAD project has invested in technologies leading to pre-prototype development of LOx/LCH$_4$ main engine since 2005. The focus of the activities were originally to support the Service Module, however in 2006 the activity was steered to support the lunar lander. The top three risks identified for RCE technology are: 1) reliable ignition; 2) performance (vacuum specific impulse – Isp); and 3) fast start (90% thrust in 0.5-sec). To address the risks, PCAD undertook a combination of in-house and contract activities.

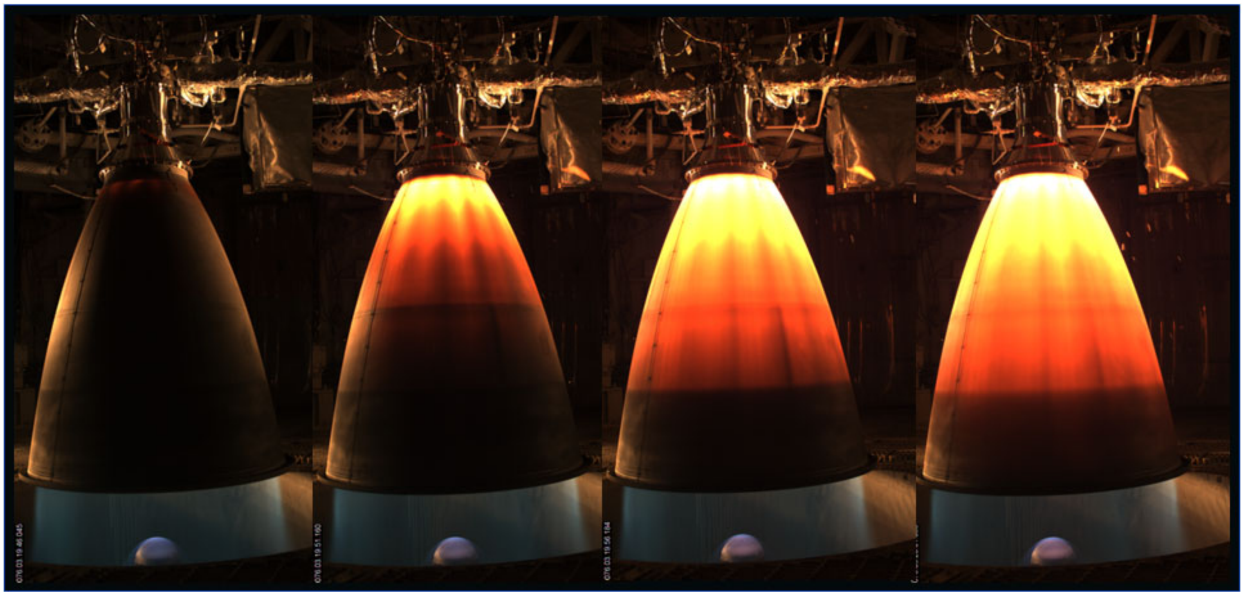
In 2006 PCAD awarded two main contracts to ATK and KT Engineering (KTE) respectively. Each contract was focused on the development and delivery of a 7,500-lb$_f$ thrust pre-prototype engine. The key performance targets for the activity were: 1) 7,500-lb$_f$ thrust, 355-sec vacuum Isp; 2) 90% rated thrust within 0.5 sec; 3) total of 24 restarts; and 5) operation over a range of inlet conditions from gas to liquid for start. The engine concepts put forward by each company were different in approach to meeting the contract requirements.

ATK teamed with XCOR to develop a pressure-fed engine concept that was actively cooled with methane.[19,20,21] To enhance the engine life, liquid methane passed through coolant channels machined into the combustion chamber. The warm methane is then injected into the engine where it mixes with liquid oxygen, creating the combustion mixture which provides the engine thrust. As part of the project execution, the ATK/XCOR team developed a "trombone" combustion chamber and injector to conduct early ground testing to examine combustion performance (C* efficiency). The trombone chamber was a water cooled thrust chamber designed to accommodate multiple length configurations to determine an optimum. The data was then used to fabricate a methane cooled workhorse combustion chamber. Sea level testing was conducted with both the trombone and workhorse combustion chambers at XCOR facilities in Mojave, California.

The second contractor, KTE, chose an ablative combustion chamber in attempts to meet the contract requirements. An ablative material is simply a thick chamber lining that slowly chars away as the engine operates. In this configuration, oxygen and methane are injected into the combustion chamber as liquids. KTE also chose to conduct smaller ignition risk reduction activities at Purdue University on both spark initiated torch igniters (SITI) and catalytic initiated torch igniters (CITI) systems. Both systems were tested successfully at sea level conditions and expected to be used in the larger engines during ground test. As part of the engine development, KTE planned to use a water-cooled combustion chamber for initial injector performance tests. A handful of hot fire sea level tests were conducted with the hardware.

To meet the Altair engine requirements, NASA issued a new RFP for a workhorse engine. Work under this contract would primarily be focused with demonstrating the main requirements of 1) 5,500-lbf thrust, 355-sec vacuum Isp; 2) 90% rated thrust within 0.5 sec; 3) total of 24 restarts; and 5) operation over a range of inlet conditions from gas to liquid for start. However, since the hardware was designated workhorse; weight and certain component developments such as valves, were omitted. From the competitive process, Aerojet was selected as the contractor. Aerojet put forward an ablative engine concept with liquid oxygen/liquid methane injection.[22] The overall activity was broken into two phases. The first phase involved Aerojet fabrication and sea level testing of multiple injector designs. The second phase was NASA taking delivery of the engines and conducting altitude performance testing at WSTF. Under the contract, three injectors were fabricated and tested at Aerojet.[23] A total of 48 tests were completed with both 8-. and 10-in. length ablative combustion chambers. Most of the tests were conducted at between 10 to 20 sec; however, one was conducted at 110-sec duration. Performance levels were lower than expected due to excessive film cooling along the combustion chamber wall. To improve performance, two additional injectors were fabricated. The second injector incorporated an alternate injector pattern than the first injector. A total of seven tests were completed before testing was stopped due to high heat release near the injector face resulting in excessive ablative erosion. Due to heating issues and low overall performance, this injector was not a viable candidate for altitude testing. The third injector was an iteration of the first injector, only with a lower percentage of film coolant. Testing was cut short due to excessive heating at the injector face.

Testing at WSTF proceeded with the first injector from the Aerojet AME contract. While the sea level testing performance levels were lower than desired, it was felt the altitude testing could still provide useful information. In particular, the team was interested in developing a correlation between the sea level results and altitude tests. The tests results would also provide key data to use in validating nozzle performance analysis, including quantifying potential loss parameters. Testing[24] was conducted with an 8-in. long ablative combustion chamber and a radiation cooled columbium Space Shuttle OMS-E nozzle extension, which provides an area ratio of 129:1. Design area ratio for the vision prototype engine design is 150:1. A total of 187 sec of run time was achieved on the engine including seven 20-sec tests and one 40-sec test. Figure 7 shows clearly the nozzle heating of the AME during testing, from left to right, at 5-, 10-, 15-, and 20-sec. The injector, chamber and nozzle were all in good physical condition after the testing. Calculated vacuum specific impulse numbers for the test program averaged approximately 344 lb_f-sec/lb_m and peaked at 345.3 lb_f-sec/lb_m with the 129:1 area ratio OME nozzle. Extrapolating to 150:1 conceptual flight design point a Isp~ 348-sec could be achieved. This is within 2% of the target. This result higher than expected based on pretest predictions from the sea level test results. Predictions were done with the well characterized Two Dimensional Kinetics (TDK)[25] computer code. Characteristic exhaust velocity efficiencies were estimated to be between 94 and 95%.



5-sec **10-sec** **15-sec** **20-sec**

Figure 7. Aerojet LOx/LCH$_4$ ascent main engine during altitude testing at WSTF.

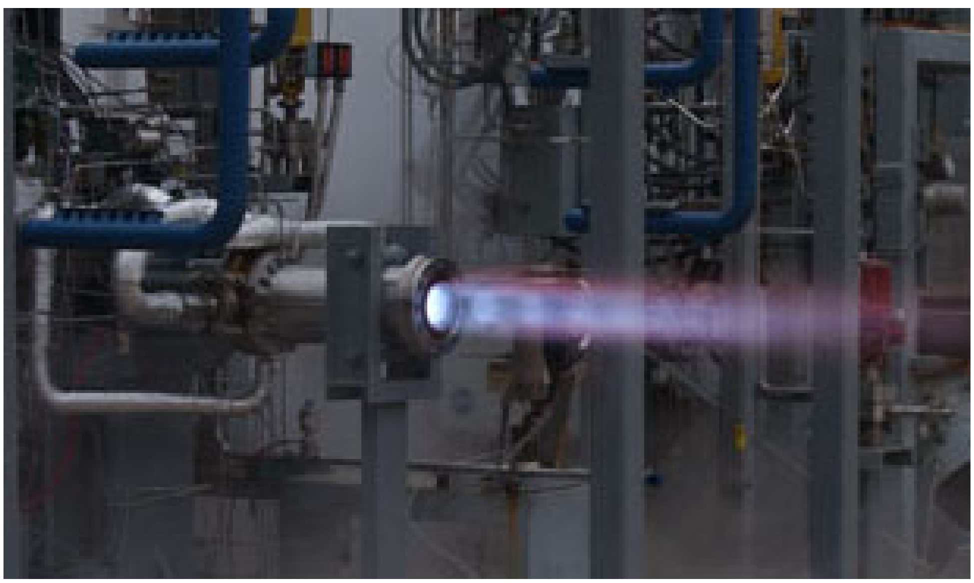


Figure 8. LOx/LCH$_4$ injector sea level test at MSFC.

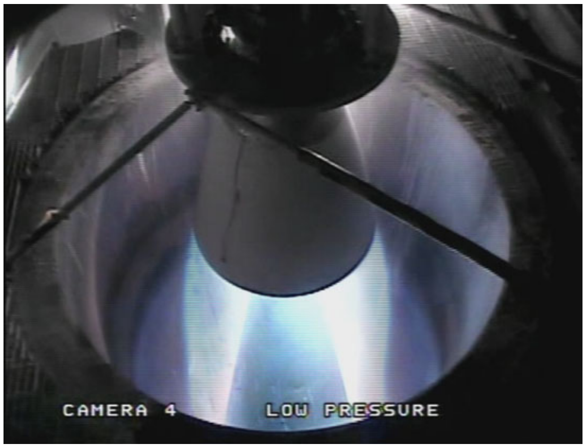
E. LOx/LCH$_4$ Ascent Main Engine Component Development

In parallel to the contract efforts, NASA conducted in-house injector development on oxygen/methane injectors. Tests were conducted on both 2-in diameter and 6-in diameter chambers at the NASA Marshall Space Flight Center (MSFC).[26,27,28,29,30] Figure 8 shows a MSFC in-house injector during sea level testing. Testing has been focused on the performance and stability characteristics of a swirl coaxial injector with multiple combustion chamber lengths. The in-house tests have been able to demonstrate 98%+ C* efficiencies with a 20-in long combustion chamber. The testing has also collected heat transfer data with use of a water cooled combustion chamber; combustion stability data for model comparison; and chamber length correlations to obtain performance levels. In addition, work has been successful in demonstrating microwave and spark torch ignition systems in sea level and altitude tests.

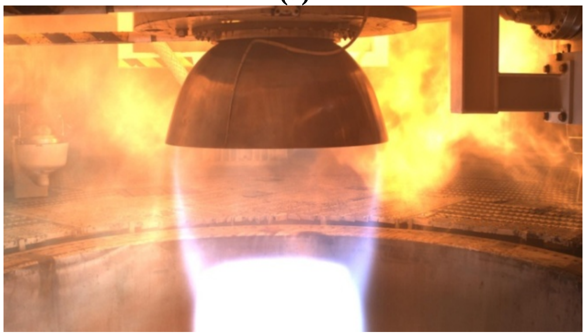
A pressure fed methane regeneratively cooled engine could be used to meet a lunar lander mission. One area identified from the ATK testing is flow instabilities in the coolant channels with methane at subcritical conditions. NASA is conducting in-house experiments[31] with a heated tube facility to simulate a methane coolant channel to examine flow stability and characterize heat transfer properties.

To address the key risk of a main engine ignition at vacuum and to provide a pathfinder engine for WSTF altitude testing, NASA and Pratt & Whitney Rocketdyne (PWR) tested an unmodified RS-18 engine with LOx/LCH$_4$ and a spark torch igniter, in altitude conditions at WSTF TS401(Fig. 9(a)).[32] Because the injector was not modified from the original configuration used for the hypergolic propellant combination of NTO/Aerozine 50, it was not expected to provide a high C* efficiency. However, three successful main engine vacuum ignitions were conducted which met the main objective of the test.

In conjunction with the Innovative Partnership Program (IPP) and PCAD, work began at the NASA Johnson Space Center (JSC) with Armadillo Aerospace on the testing of a 1,500-lb$_f$ thrust-class LOx/LCH$_4$ rocket engine.[33,34] Sea level testing was conducted at the Armadillo facilities in Caddo Mills, Texas, and simulated altitude tests were conducted at WSTF (Fig. 9(b)). Testing examined engine performance and ignition, both gas torch and pyrotechnic, at altitude conditions. The rocket engine was designed to be configured with three different nozzle configurations, including a dual-bell nozzle geometry. A total of 10 hot-fire ignition and dual-bell nozzle tests were conducted at WSTF.



(a)



(b)

Figure 9. LOx/LCH$_4$ engine testing at WSTF. (a) PWR RS18; (b) Armadillo Aerospace dual bell nozzle engine.

III. Liquid Oxygen (LOx) – Liquid Hydrogen (LH$_2$) Propulsion

One of the mission enabling technologies to support future lunar missions is the development of a LOx – LH$_2$ deep throttling descent engine. The descent main engines must be able to throttle and remain controlled by the crew to provide a soft landing or to maneuver to a different landing site. Rocket engines typically have a fixed point design that does not allow power levels to throttle over a wide range of operating conditions. If not designed properly, throttling a rocket engine can create low frequency instability in engine pressure, which can cause a reduction in performance or even damage to the engine or vehicle. As currently defined, deep throttling for the lunar missions is a 10:1 ratio, or an engine that can stably throttle from 100 to 10% power. The PCAD project is exploring three options through contracted efforts to develop deep throttling technologies. The first is with the Common Extensible Cryogenic Engine (CECE),[35] a modified RL10 from PWR. A second effort is technology development for an expander cycle engine with Northrop Grumman Aerospace Systems (NG) based on the Pintle injector. The third option is a throttling injector concept being developed by Aerojet. Along with the contracted efforts, NASA is exploring in-house technology efforts with the development of an expander cycle test bed at MSFC.

A. Descent Engine Technology Contracts

The CECE contract with PWR was initiated in July 2005 with the development of the Demo 1.0 activity. The primary focus of Demo 1.0 was to assemble a deep throttling technology demonstrator from existing expander cycle RL10 parts. A number of key components were changed to develop the demonstrator including the fabrication of a fixed-geometry, high pressure drop injector, change out of turbine bypass (TCV) and oxidizer control valves (OCV), adding a larger turbine bypass valve (TBV), and a variable area cavitating venturi (VACV).

The first test series, Demo 1.0, completed four test runs between April and May 2006 at the PWR E6 facility in West Palm Beach, Florida. Figure 10 shows the CECE during altitude testing in the E6 facility. The testing was able to obtain baseline performance and stability data from 20 to 90%. To meet the requirements, testing was completed down to 10% power. However, at 16% power, lower power chugging oscillations were detected. Despite the chugging the tests were successful because it quantified the baseline operating boundaries and provided valuable data to update performance and operations models. It was determined from the data analysis[36] that the chugging was the result of vapor formation in the injector oxygen manifold. A second series of tests, Demo 1.5, were conducted in March and April 2007 with the same engine configuration as Demo 1.0. A total of four tests accumulated a total of 1162-sec of run time. The testing explored the boundaries of the chug instability over a range of mixture ratios and chamber pressures. During the testing additional technology challenges were identified, in particular, 1 Hz instability in the fuel system due to film boiling at low power. There was also a 4000 Hz, 1T combustion oscillation observed between 30 to 40% power. Testing was also conducted at throttle rates from 100 percent/sec down to 2.5 percent/sec. Overall the Demo 1.0 and Demo 1.5[37] testing developed a wide ranging set of baseline performance data down to 10% power and identified key technology needs for future efforts.

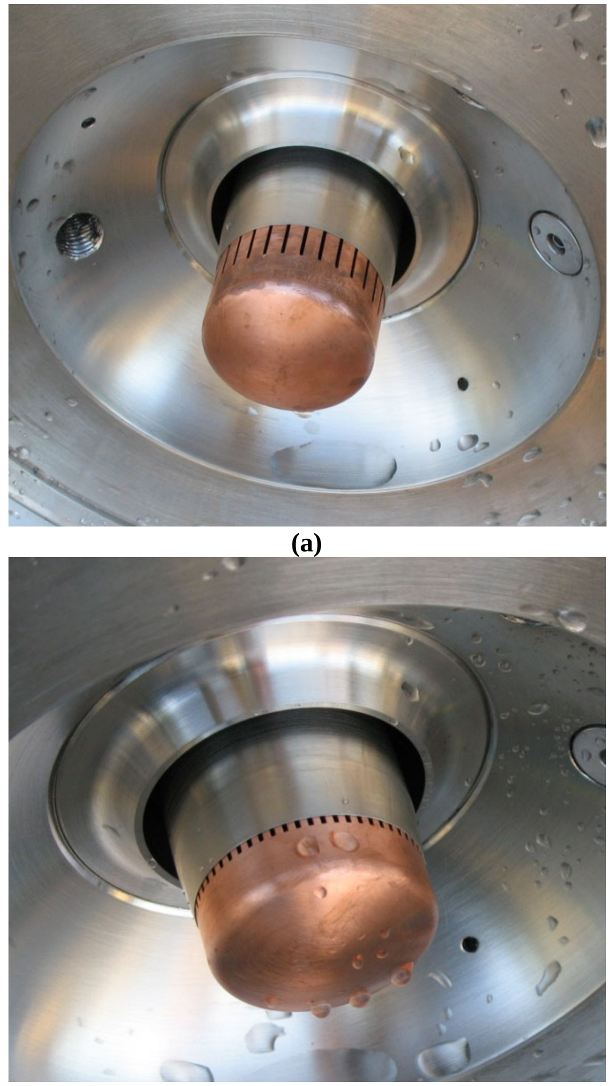
Figure 10. PWR CECE during altitude testing in PWR E6 test stand.

The Demo 1.6[38] test campaign was designed to evaluate mitigations for the low frequency combustion instability ("chug") observed at low power conditions during the Demo 1.0 and Demo 1.5 test programs. To eliminate the oxygen manifold film boiling, a new injector was designed which incorporated a thermal barrier coating on oxygen side of the inner propellant plate. The goal was to reduce the heat transfer from the warm hydrogen into liquid oxygen and prevent the film boiling. To mitigate the chug, the Demo 1.6 injector was modified from the previous configurations to include a spray-on insulation to reduce heat transfer to the LOx manifold, which was believed to be a significant contributor to the low power instability. In addition, gaseous helium injection into the LOx manifold was used as a means to stabilize the system. Also explored in this test series was mitigation for a low power 1 Hz fuel system oscillation caused by sub-critical hydrogen boiling in the chamber cooling jacket. Reduced area gas venturis were utilized to avoid the 1 Hz fuel-size oscillation by keeping the cooling jacket supercritical down to lower engine power levels.

The final test of the CECE engine, Demo 1.7[39] was designed to test the ability of starting the engine at low power and to demonstrate closed loop control of a throttling engine. Demo 1.7 testing[40] successfully demonstrated a number of engine modes of operation including chamber pressure and mixture ratio closed-loop control over a wide range of throttled power levels, fast throttle ramp rates, minimum power down to a smooth start to 10% power, eleven rapid relights demonstrated (many achieved as 2 relights within the same test matrix run), and high power, high mixture ratio operation. Finally the testing demonstrated low power stability, including chug-free operation down to 5.9% power. This represents a 17.6:1 overall cryogenic deep throttling ratio in a complete expander cycle engine system with all system-level interactions which greatly enhanced the value of the technology database acquired. Total Demo 1.7 engine testing has concluded with a total run time of 2,403.0-sec (40.0-min). Total CECE demonstrator engine run time has concluded with 7,435.8-sec (123.9-min).

The second contracted effort developing deep throttling LOx/LH_2 engine technologies is with Northrop Grumman Aerospace Systems (NGAS) on the TR202 contract.[41,42] The work with NGAS was started in June 2005 and is also focused on an expander cycle engine. The focal point of the NGAS engine concept is the variable area pintle injector, which is similar to the injector used on the Apollo Lunar Module Descent Engine. The first phase of the contract was focused on the design and development of a test-bed pintle injector. The injector design has a oxidizer centered pintle where the oxygen flows through a central passage and is injected radially through individual orifices into the combustion chamber. The fuel is injected through an annular sleeve around the center pintle post. The fuel creates a sheet that impinges with the radial oxygen flow. The injector throttling is controlled by articulating the fuel sleeve along the length of the pintle to either increase or decrease the oxygen flow area. Figure 11 shows the throttling pintle injector: high thrust setting (a) and low thrust setting (b) during water flow testing. For a flight engine the sleeve would be controlled by an actuator based on throttle inputs from the flight profile. For ground testing the fuel sleeve/throttle position did not have a position actuator.

Testing was conducted with both ablative and water cooled combustion chambers.[43,44] The ablative chamber test series encompassed 22 tests at a nominal mixture ratio of 6, and thoroughly explored injector momentum rate ratio design space; confirmed expectations for excellent high performance potential over the high-end of the throttle power range; demonstrated stable deep-throttle combustion performance at 25 and 10% power conditions; and, validated the thermal integrity of the hardware design. A total of six Pintle configurations were tested using two fuel injection ring sizes. The ablative chamber test series yielded sufficient understanding and confidence in the injector design to justify change over to calorimeter chamber hardware, which enables accurate determination of performance and heat transfer characteristics in a follow-on test series. Testing with the calorimeter was successful in meeting all primary and secondary technical objectives including high performance (>98% C* (combustion) efficiency); stable 10:1 deep throttling; measurement of heat transfer characteristics; evaluation of off-nominal oxidizer to fuel mixture ratio (MR) sensitivities; and evaluation of L* sensitivity. The majority of the test program was devoted to an extensive Design of Experiments (DOE) for optimized injector performance in which the major influencing parameters were characterized. After extensive testing, the team arrived at an optimized high-performance injector design. Testing of the optimized injector demonstrated stable combustion over the full 10:1 throttle range, and heat transfer characteristics were within anticipated ranges.



(a)

(b)

Figure 11. Northrop Grumman throttling pintle injector: high thrust setting (a) and low thrust setting (b) during water flow testing.

In 2009, Aerojet was also awarded a contract to develop deep throttling injector[45]technologies. The contract builds upon an internal research project the company conducted to demonstrate 10:1 throttling with a 1,500-lb$_f$ injector.[46] The current effort will focus on 10:1 throttling with a 9,000-lb$_f$ thrust injector. The engine system envisioned is an expander cycle LOx/LH$_2$ engine. The injector is anticipated to be sea level tested in 2011 with a hydrogen regenerative cooled combustion chamber supplied under a Space Act.

B. NASA In-House Component Development

NASA is conducting several complementary component development activities in-house. Development of the in-house technologies will be conducted on the Lunar Lander Descent Engine Testbed (LLDETB) on Test Stand 500 at MSFC. This sea-level rig is a flexible system to accommodate change out of injectors, combustion chambers, and turbomachinery. As part of the test rig build-up a number of individual components have been fabricated and tested independently. One of the first components tested was a dual oxygen–inlet swirl coaxial element deep throttling injector.[47] The dual-inlet injector has two fixed area oxygen manifolds to maintain sufficient pressure drop across a wide range of throttle conditions. Each manifold has fixed inlet areas to the oxygen posts of the injector and flow can be independently controlled with shutoff valves. For high power cases oxygen would flow through both manifolds, however at low power, flow to the secondary manifold would be cut off. The sea level tests like that shown in Fig. 12, provided all data needed to calculate C* efficiency, heat flux, and other information such as high speed pressure data.[48] The injector achieved very high C* efficiency numbers and stable operation at the high power levels. There were some low frequency (chug) instabilities at the lower power levels. These chug modes are currently being attributed to the LOx supply temperatures which were warmer than ideal. Results from the testing will contribute to future development of a two-stage injector concept or any deep throttling technology.

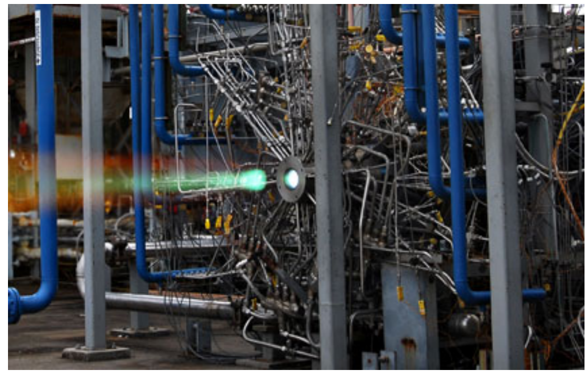


Figure 12. NASA two-stage throttling LOx/LH_2 injector during sea level testing with water cooled calorimeter at MSFC.

An important technology in the control of deep throttling engines is the ability to control the cooling flow from the combustion chamber to the fuel turbo pump. To examine improved control, work under an Innovative Partnership Program (IPP) with Vacco Industries developed an advanced turbine bypass valve (ATBV). The goal of testing was to determine the effective flow area versus valve position at nine equally spaced points in the valve travel and exercise the valve under engine conditions to examine seal performance. Figure 13 shows the ATBV in a test position at MSFC. The test program consisted of two tests series to determine the flow coefficient versus position and evaluate the ATBV design while operating in simulated engine temperature, flow rate, and pressure conditions. The team was also able to operate the valve in various simulated engine environments to fully characterize the performance of the ATBV design.

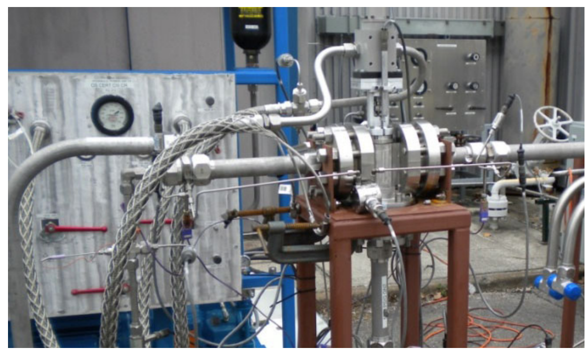


Figure 13. ATBV during performance testing at MSFC.

IV. Conclusion

The Propulsion and Cryogenic Advanced Development (PCAD) Project Team led by the NASA Glenn Research Center (GRC) in partnership with the NASA Marshall Space Flight Center (MSFC), the NASA Johnson Space Center (JSC), the NASA White Sands Test Facility (WSTF), and industrial partners, is conducting a focused technology development effort to advance high performance cryogenic propulsion systems. Over the last 5 years this team has been a model for cross center collaboration. To date the team has made great strides in reducing the primary risk of LOx/LCH_4 ignition. At the beginning of PCAD, concerns were expressed that the ignition of LOx/LCH_4 was not feasible. However, with a combination of in-house and contractor activities, the PCAD team has shown that LOx/LCH_4 can be reliably ignited over a wide range of conditions. Also, under contract, PCAD has demonstrated that reaction control engines can be developed to the pre-prototype level which meets mission requirements. Due to the nature of pulsed operation, it can be argued that the LOx/LCH_4 reaction control engine was the most challenging problem facing the team. However, despite the team's successes, new challenges have arisen during the course of the project. For the reaction control engines, system interactions and operations in a cluster proved to be difficult. The engine and flow system operation were sensitive to system design and operation, hence the requirement to move forward with the Integrated Propulsion System Test Bed (IPSTB). There is also still individual work to be done with the reaction control engines with additional vacuum testing. Much of the performance work was done at sea level at single set point flow inlet conditions. PCAD is planning to do extensive testing to evaluate engine performance across a wide range of propellant inlet pressure and temperatures. Testing will also be conducted to simulate the hot and cold variations the engine will see during space operations.

The ascent main engine has not had as much success as the reaction control. While the RCE work has done much to reduce the risks associated with the propellant combination, ultimately it is the performance of the ascent main engine which will determine if LOx/LCH_4 is a viable candidate for the lunar ascent vehicle. Based on the system studies, the success is tied to the ability to demonstrate the highest level of vacuum specific impulse, with 355-sec being the current target. The amount of weight savings to the vehicle is directly tied to the Isp level achieved by the main engine. A lower specific impulse will result in a lower mass savings for the LOx/LCH_4 option versus the current hypergolic baseline. The current effort with the Aerojet design is to see just how close the team can get a main engine to that goal of 355-sec. Once successful, the next step will be to develop the main engine technologies with a pre-prototype engine. This engine could be either ablative or regeneratively cooled.

The descent main engine activities have successfully demonstrated stable throttling to 10% thrust or less with multiple injector concepts using liquid oxygen and liquid hydrogen propellants. The Pratt & Whitney Rocketdyne CECE demonstrator engine test series concluded with 7,435.8-sec (123.9-min) of total run time. The testing demonstrated chamber pressure and mixture ratio closed-loop control over a wide range of throttled power levels, fast throttle ramp rates, minimum power down to a smooth start to 10% power, eleven rapid relights demonstrated, and high power, high mixture ratio operation. The testing also demonstrated low power stability, including chug-free operation down to 5.9% power. This represents a 17.6:1 overall cryogenic deep throttling ratio in a complete expander cycle engine system with all system-level interactions which greatly enhanced the value of the technology database acquired. Testing with a pintle injector from Northrop Grumman was successful in meeting all primary and secondary technical objectives including high performance (>98% C* (combustion) efficiency); stable 10:1 deep throttling; measurement of heat transfer characteristics; evaluation of off-nominal oxidizer to fuel mixture ratio (MR) sensitivities; and evaluation of L* sensitivity. Finally, a NASA in-house developed dual oxygen manifold injector was also able to demonstrate stable throttling to a 10% power level.

The PCAD team continues to build upon the success to date and strives to provide timely and relevant data to NASA mission study teams so an informed decision can be made on the direction of the next propulsion system for exploration missions.

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