

LIQUID OXYGEN/LIQUID METHANE PROPULSION AND CRYOGENIC ADVANCED DEVELOPMENT

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Exploration Systems Architecture Study conducted by NASA in 2005 identified the liquid oxygen (LOx)/liquid methane (LCH₄) propellant combination as a prime candidate for the Crew Exploration Vehicle Service Module propulsion and for later use for ascent stage propulsion of the lunar lander. Both the Crew Exploration Vehicle and Lunar Lander were part the Constellation architecture, which had the objective to provide global sustained lunar human exploration capability. From late 2005 through the end of 2010, NASA and industry matured advanced development designs for many components that could be employed in relatively high thrust, high delta velocity, pressure fed propulsion systems for these two applications. The major investments were in main engines, reaction control engines, and the devices needed for cryogenic fluid management such as screens, propellant management devices, thermodynamic vents, and mass gauges. Engine and thruster developments also included advanced high reliability low mass igniters. Extensive tests were successfully conducted for all of these elements. For the thrusters and engines, testing included sea level and altitude conditions. This advanced development provides a mature technology base for future liquid oxygen/liquid methane pressure fed space propulsion systems. This paper documents the design and test efforts along with resulting hardware and test results.

INTRODUCTION

Exploration System Architecture Study (ESAS)¹ conducted in 2005 identified Liquid Oxygen (LOx)/Liquid Methane (LCH₄) as a prime candidate propellant combination for integrated reaction control and main pressure fed propulsion of the Crew Exploration Vehicle (CEV) Service Module (SM) and the Lunar Surface Access Module (LSAM) ascent stage. This identification was based on a combination of performance and cost characteristics coupled with In-Situ Resource Utilization compatibility.

Following completion of the ESAS in July 2005, implementation activities began for the new architecture, substantially adjusting the approach of the Constellation program. At that time, significant effort was already underway for the CEV including competing Phase I and II studies with a plan for competitive downselect to the prime development contract. The changes due to the new architecture required a transition from the original plan. As part of the transition, it was decided to structure LOx/LCH₄ propulsion efforts as an advanced development project, the Propulsion and Cryogenics Advanced Development (PCAD) Project, with the objective of maturing the technologies to Technology Readiness Level six by CEV Preliminary Design Review (PDR). Hypergolic propellants were retained as a design option. The plan

allowed for the CEV to use LOx/LCH₄ if advanced development was successful and propulsion system trades showed it was the best choice.

A project plan was developed and approved in August/September 2005, allowing implementation to commence. The plan was to invest in the most immature components and integrate the components to perform propulsion system level testing. In December 2005, soon after initiation of the project, the Constellation architecture was changed from the ESAS recommendation to what was termed "lunar sooner." This change removed LOx/LCH₄ from contention for the initial Service Module design. The PCAD project was redefined to balance future service module upgrade and lunar ascent propulsion needs, and allow for a longer schedule. The scope was increased to include the LSAM descent module needs as well as the LSAM ascent module needs while funding was also reduced, but the bulk of the overall objectives of the original project were retained. In 2008, the PCAD project was split to form a companion project called the Cryogenic Fluid Management (CFM) Project. The new CFM project addressed the cryogenic fluid management needs of the CEV and LSAM and also acquired expanded scope of addressing the cryogenic fluid management needs of ground operations and in-situ resource utilization.

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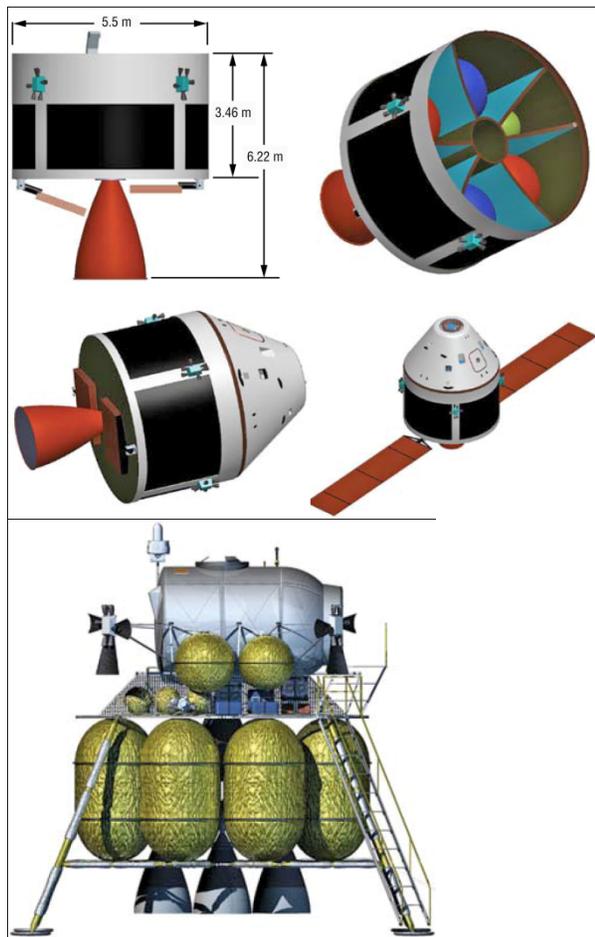


Fig. 1: Reference Luna CEV SM and LSAM.

Over a period of five years, the PCAD and CFM projects were successful in meeting their objectives, demonstrating main engine, reaction control thrusters, igniters, fluid management, and feed system components, achieving essentially technology readiness level five to six for all items funded. These will be discussed for each category of propulsion system element.

TARGET PROPULSION SYSTEM CONCEPT

The ESAS study defined a recommended reference propulsion system concept utilizing the LO_x/LCH₄ propellant combination in an integrated main/reaction control, pressure fed configuration. Sizing and performance requirements for the propulsion system were based on mission requirements for the CEV SM. The CEV SM fit within an overall architecture derived with two Design Reference Missions (DRMs) and the possibility for extension to more distant future missions. The DRMs were 1) crew or cargo transport

to the moon for lunar exploration at any surface location, for mission durations between four days and up to six months, with any time return capability; 2) crew transport to the International Space Station; and extension of at least the Crew Module to future Mars missions. The LO_x/LCH₄ propulsion combination was also recommended to be utilized in the ascent stage of the LSAM.

The ESAS recommended architecture was based on a 1.5 Launch Earth Orbit Rendezvous/Lunar Orbit Rendezvous mission mode, where the Crew Exploration Vehicle was launched on a small launch vehicle and the Earth Departure Stage (EDS) and LSAM were launched on a heavy lift vehicle. The CEV docked with the LSAM/EDS, and that package traveled to the moon. The EDS was dropped after its earth departure burn, and the LSAM descent stage provided lunar orbit injection and landing delta velocity. The ascent stage of the LSAM provided ascent delta velocity. The CEV SM provided earth return delta velocity and was discarded before direct CM entry and landing back on earth. The CEV was also designed to function for crew transport to the International Space Station and return to earth, which requires launching to the 56.6 degree inclination with transit over the north Atlantic.

The CEV SM propulsion functions were: abort mode thrust and control (sized main engine thrust), in-space maneuvers, proximity operations and docking (sized and geometrically distributed fine control capability), lunar orbit station keeping (established mission duty cycle duration), trans Earth injection, and SM disposal maneuver post CM/SM separation for CM earth entry.

LO_x/LCH₄ was selected by ESAS as one of two propellant combinations of most interest because it is high performing, non-toxic, and can be obtained from Martian and lunar in-situ resources (CH₄ from the Martian atmosphere and LO_x from the Martian and lunar soil).ⁱⁱ The integrated pressure fed propulsion system design concept to perform these functions was selected for its simplicity, reliability and lower development cost over other comparable systems.ⁱⁱⁱ

The ESAS recommended SM system consisted of one fixed position main engine, of 66.7 kN thrust, 363.6 seconds Isp and expansion ratio of 150:1. The system had 24 reaction control system (RCS) thrusters of 445 N thrust, 317.0 seconds Isp and 40:1 area ratio arrayed about the SM. The 24 thruster array was arranged into six sets of four thrusters plumbed through three redundant manifolds, each with a thermodynamic vent at the end to maintain propellant quality.

Propellant was to be provided to the main engine and thrusters by a pressure-fed propellant storage and feed system. Propellant loading was based on 1724 m/s main delta V, 50 m/s delta V with the CM attached and 15 m/s delta V post CM/SM separation for SM disposal. Tanks would be flown partially full for less demanding missions. The propellant was to be stored in two sets of two series plumbed tanks, one set for each propellant with nominal operating pressure of 2.24MPa and maximum expected operating pressure of 2.8 MPa. Both main and RCS propulsion were to be fed from the common propellant tanks. Pressurization was accomplished by regulated Helium pressurant, stored at supercritical conditions (41.4 MPa) in two tanks. Liquid acquisition was to be accomplished by a system of screen channels similar to space shuttle orbital maneuvering and reaction control propellant tanks.

ESAS recommended that this basic propulsion system capability be adapted to the lunar ascent stage. Adaptations for this application included reduced number of RCS thrusters (16) requiring fewer manifolds, slightly increased tank sizing with 1866 m/s ascent delta V with 22 m/s RCS delta V. Main engine thrust for the LSAM application was set at 44.5kN (2/3 of SM). Geometry for feedlines and supporting equipment was expected to be different than for the service module.

ESAS proposed that an investment in advanced development/risk reduction could be accomplished to achieve Technology Readiness Level six by CEV Preliminary Design Review (PDR). Development schedules for LOx/LCH₄ once the technology was matured were analyzed to be essentially the same as for the traditional bi-propellant storable alternative (Monomethyl Hydrazine/Nitrogen Tetroxide). ESAS also proposed that an early ISS flight mission use of the CEV would eliminate residual risk for LOx/LCH₄ application to lunar missions that could not be eliminated by the proposed ground based advanced development program. This was possible because the CEV had ample performance margin for the ISS mission compared to the lunar missions. The advanced development/risk reduction recommended by ESAS was planned and initiated in late 2005.

MAIN ENGINE DEVELOPMENT

The PCAD project invested in technologies leading to pre-prototype development of the LOx/LCH₄ main engine since 2005. The top three risks identified were: 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.

ASCENT MAIN ENGINE DEVELOPEMT

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-lbf thrust pre-prototype engine. The key performance targets for the activity were: 1) 7,500-lbf thrust, 355-sec vacuum Isp; 2) 90% rated thrust within 0.5 seconds; 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^{iv,v,vi}. Sea level testing was conducted with both the water and methane cooled combustion chambers at XCOR facilities in Mojave, CA.

The second contractor, KTE, chose an ablative combustion chamber in response to the contract requirements. A handful of sea level tests were conducted with the engine. As focus shifted away from a service module system to a lunar lander, NASA did not exercise the contract options due to the changing requirements.

To meet the new lunar lander engine requirements, NASA issued a new RFP for a workhorse engine. Work under this contract primarily focused on demonstrating the following requirements 1) 5,500-lbf thrust, 355-sec vacuum Isp; 2) 90% rated thrust within 0.5 seconds; 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 as workhorse, weight and component development, including engine valves, were omitted. An Aerojet ablative engine concept with liquid oxygen/liquid methane injection system was selected^{vii}. 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 NASA White Sands Test Facility (WSTF). Under the contract, three injectors were fabricated and tested at Aerojet^{viii}. A total of 48 tests were completed with both 8-inch and 10-inch length ablative combustion chambers. Most of the tests were 10-20 seconds in duration; however, one was conducted at 110-second duration. This 110-second test was used to determine erosion rates and life of the ablative chamber with the injector. Testing at NASA-

WSTF proceeded with the first injector produced under the Ascent Main Engine (AME) contract. While the sea level performance was lower than desired, altitude testing was conducted as planned to develop a correlation between the sea level and altitude results and validate nozzle performance analysis including quantifying potential loss parameters. Testing was conducted with an 8-inch long ablative combustion chamber and a radiation cooled columbium Space Shuttle OMS-E nozzle extension, which provided an area ratio of 129:1.^{ix} Design area ratio for the prototype engine design was 150:1. A total of 187 seconds of run time was accumulated on the engine including seven 20-second tests and one 40-second test. The injector, chamber and nozzle were all in good physical condition after the testing. The average vacuum specific impulse calculated for the test program was 344 lb_f-sec/lb_m and the maximum was 345 lb_f-sec/lb_m. Extrapolating to an area ratio of 150:1, a specific impulse of approximately 348 lb_f-sec/lb_m could be achieved, which is within 2% of the target. This result was higher than expected based on pretest predictions from the sea level test results. Predictions were done with the well characterized Two Dimensional Kinetics (TDK) computer code. Characteristic exhaust velocity efficiencies were estimated to be between 94 and 95%.

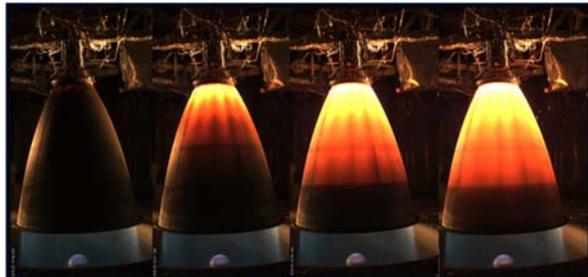


Fig. 2: Aerojet LOx/LCH₄ ascent main engine during altitude testing at NASA WSTF.

In parallel to the contract efforts, NASA conducted in-house injector development on oxygen/methane injectors. Tests were conducted on both 2-inch diameter and 6-inch diameter chambers at NASA Marshall Space Flight Center (MSFC)^{x,xi,xii,xiii,xiv}. Testing focused on the performance and stability characteristics of a swirl coaxial injector with multiple

combustion chamber lengths. The in-house tests demonstrated C* efficiencies over 98% with a 20-inch long combustion chamber. A water cooled combustion chamber was used to collect heat transfer data. Different length chambers were used to obtain performance level correlations to chamber length. The chambers were instrumented to collect combustion stability data for model comparison. In addition, microwave and spark torch ignition systems were demonstrated in sea level and altitude tests.



Fig. 3: LOx/LCH₄ injector sea level test at NASA MSFC.

To address the key risk of 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/LCH₄ and a spark torch igniter, in altitude conditions at NASA WSTF test stand 401^{xv}. The injector was not modified from the original configuration used for the hypergolic propellant combination of NTO/Aerozine 50. 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 with Armadillo Aerospace on the testing of a 1,500-lbf thrust-class LOx/LCH₄ rocket engine^{xvi,xvii}. Sea level testing was conducted at the Armadillo facilities in Caddo Mills, TX and simulated altitude tests were conducted at NASA WSTF. 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 NASA WSTF.

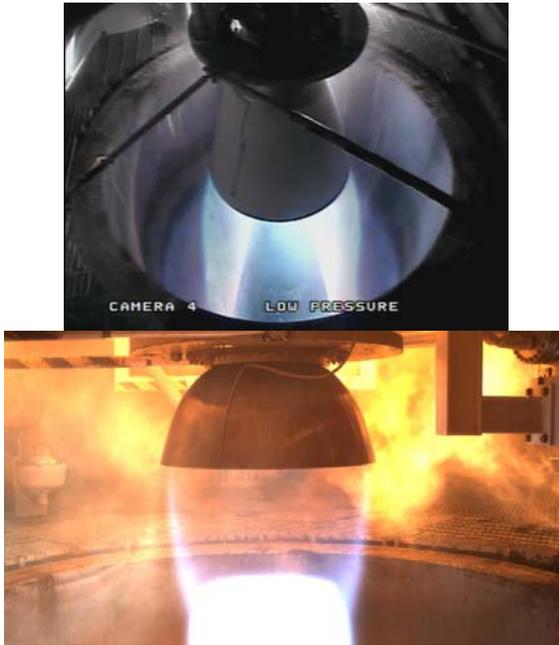


Fig. 4: LOx/LCH₄ engine testing at NASA WSTF. a) PWR RS18; b) Armadillo Aerospace dual bell nozzle engine.

REACTION CONTROL THRUSTERS

The PCAD project invested in technologies leading to pre-prototype development of LOx/LCH₄ reaction control engines (RCE) with the release of contract request for proposals (RFPs). The top three risks identified for RCE technology were: 1) Ignition reliability; 2) Performance (vacuum specific impulse); and 3) Pulse width repeatability. To address the risks, PCAD undertook a combination of in-house and contract activities.

In 2006, PCAD awarded RCE contracts to Aerojet and Northrop Grumman. Each contract focused on the development and delivery of a 100-lbf thrust pre-prototype engine subsystem. The key performance requirements were: 1) 317-second vacuum Isp; 2) 4 lbf-sec minimum impulse bit (Ibit); 3) 80-msecond electronic pulse width (EPW); 4) 25,000 valve cycles and 5) ignition and operation over a range of inlet conditions including liquid and gaseous propellants. The two suppliers pursued different engine concepts in response to these requirements.

Aerojet put forward a concept based on previous LOx/Ethanol engine development and internally funded activities. Initial testing was performed with 870-lbf engines that were originally designed to operate on LOx/Ethanol propellants and were modified to accommodate LOx/LCH₄^{xviii}. The modified units were successfully tested on the Auxiliary Propulsion System

Test Bed (APSTB) in NASA WSTF TS401. Aerojet used the test data to design their 100-lbf engine concept consisting 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. Flow control for both the main chamber and igniter were controlled by a single set of dual coil valves. Over 55,000 cycles were demonstrated at cryogenic temperatures, exceeding the specified 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. All key performance criteria were demonstrated using an impinging injector design. Aerojet conducted over 1300 engine pulse tests at a variety of duty cycles and accumulated more than 1900 seconds of operating time during sea level, engine development testing^{xix,xx}. Aerojet met the 317-sec Isp requirement, calculated based on estimated nozzle losses and exceeded the 80-msec EPW requirement by demonstrating 40-msec EPW. Aerojet provided 5 engines to NASA that were subsequently tested in a multiple engine configuration on the APSTB at NASA WSTF and 2 units for testing at NASA Glenn Research Center (GRC) in the Altitude Combustion Stand (ACS).

Sea level^{xxi} and altitude performance testing^{xxii} has been conducted at NASA GRC with the Aerojet engines. A total of 60 altitude hot-fire tests were completed with the Aerojet 100-lbf LOx/LCH₄ engine and propellant conditioning feed systems (PCFS)^{xxiii,xxiv}. The PCFS was used to obtain conditions over the range of nominal (204 °R LOx/204 °R LCH₄), cold/cold (160 °R LOx/170 °R LCH₄), to warm/warm (224 °R LOx/224 °R LCH₄). 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.

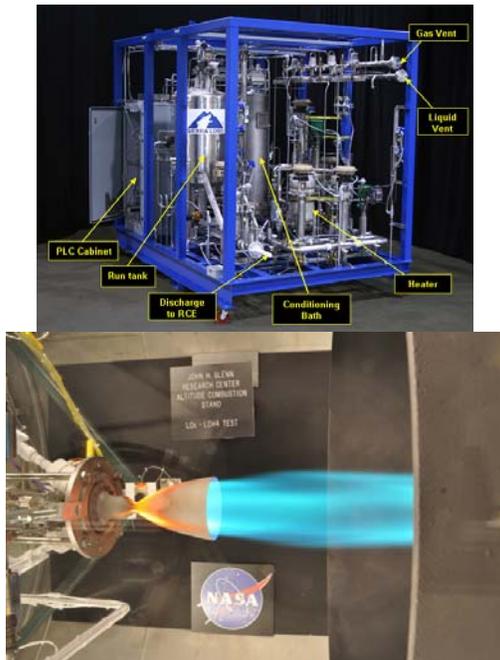


Fig. 5: a) Propellant Conditioning Feed System skid (PCFS);
 b) Aerojet 100-lbf LOx/LCH₄ reaction control engine in test at NASA GRC.

Northrop Grumman put forward a concept based on previous work on hypergolic propellant engines. The combustion chamber and a portion of the nozzle were regeneratively cooled with both oxygen and methane^{xxv}. The full engine area ratio (120:1) was completed with a columbium radiation cooled nozzle extension. Flow control for both the main chamber and igniter was controlled by a single set of independent single coil fuel and oxidizer 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 encountered a number of design and manufacturing issues, which slowed progress. As a result, budget limitations required changes to the scope of the contract, which resulted in the elimination of the three pre-prototype engine deliverables. Northrop Grumman developed 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

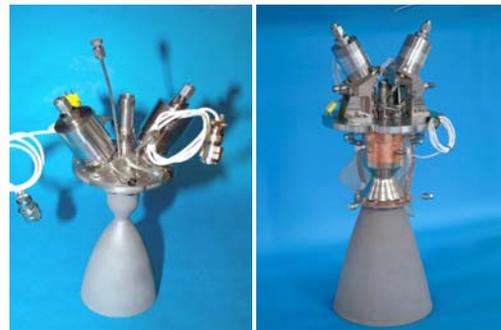


Fig. 6: Aerojet 100-lbf LOx/LCH₄ reaction control engine (Top) and Northrop Grumman 100-lbf LOx/LCH₄ reaction control engine (Bottom).

331 sec, which exceeded the specification requirement of 317sec. NASA currently has one pre-prototype unit available for further in-house testing.

IGNITION RISK REDUCTION

To address the highest risk for LOx/LCH₄ propulsion systems, reliable ignition, NASA conducted numerous in-house experimental efforts to examine the issue. The work was completed at both RCE and AME scales. The majority of the work was conducted with spark torch igniters^{xxvi,xxvii,xxviii,xxix,xxx}, however there was work done with microwave^{xxxi,xxxii}, piezoelectric, spark torch/glow plug combination,^{xxxiii} and catalytic ignitions systems. Overall there were no significant issues identified that would prohibit the reliable ignition over a range of conditions with LOx/LCH₄. 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^{xxxiv}. Completion of this activity did not identify any issues with the hardware or designs for long duration applications. The work identified issues with spark plug durability and the reliability of power exciter units. In both cases, PCAD worked additional technology tasks to address the issues. There appear to be viable solutions in work to reduce the risk.

Many of the issues remaining with LOx/LCH₄ ignition are related to the 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.

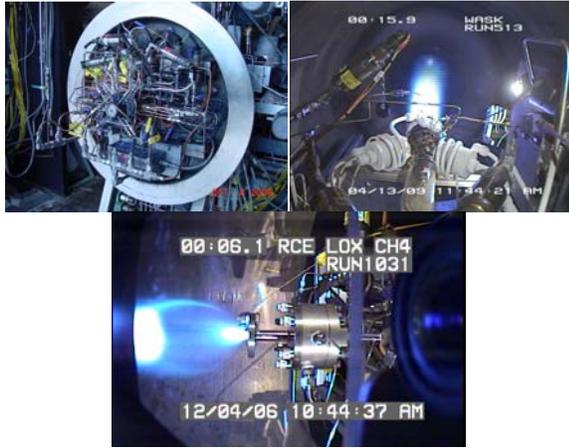


Fig. 7: LOx/LCH₄ Altitude Ignition Testing at NASA GRC a) Test Cell 21 configuration; b) WASK spark torch igniter during test; c) ascent main engine class igniter during test.

CRYOGENIC FLUID MANAGEMENT

The proposed lunar missions included two key environmental challenges for the propellant storage and delivery system: (1) thermal environments during several mission phases including ascent, low earth orbit loiter, trans lunar injection, low lunar orbit capture and loiter, lunar surface standby, and lunar orbit rendezvous and docking, and (2) acceleration environments that included launch loads, microgravity of orbital loiter and lunar transfer, lunar surface gravity, and potentially adverse accelerations due to RCS thruster firing during docking maneuvers. Developing a vehicle that could satisfy the defined mission requirements and concept of operations while exposed to these environments required technology solutions for thermal control, tank pressure control, propellant management, propellant gauging and propellant transfer (engine feed). The technology developments completed in each of these areas under the PCAD and CFM projects are briefly summarized here.

The primary requirement for the CFM system of the lunar lander ascent stage concept was the need to operate in space for up to 240 days without propellant loss. Because a pressure fed propulsion system was selected and because the PCAD project was developing thrusters with the ability to operate over a wide range of propellant inlet conditions, the thermal control solution included using the sensible heat of the propellants and allowing the saturation pressure and temperature of the oxygen and methane to rise throughout the duration of the mission. To meet the 240 day storage requirement, analysis indicated that a high performance “passive” solution was possible that would rely on thick multilayer insulation, low thermal conductivity structural support and tank mixing to keep

pressure within the desired control bands without venting. Helium, stored at cryogenic temperature, is used to pre-pressurize the tanks prior to thruster operations.

Thermal Control

The application of LOx/LCH₄ for lunar ascent propulsion created the need for extremely long duration and mass efficient storage of LOx and LCH₄ without boil-off losses. To better understand the challenges of this thermal control problem, the CFM project initiated several analytical modeling activities including full computational fluid dynamics models, lumped capacitance models^{xxxv}, and the development of system and component level thermal, structural and power models.

The Cryogen Storage Integrated Model (CryoSIM) is a CFM system/subsystem tool to support overall mission performance prediction of in-space cryogenic storage systems^{xxxvi}. Its development was driven by the need to standardize and integrate a number of existing NASA cryogenic codes that use various algorithms with varying degrees of documentation, verification, and availability. CryoSIM is an iterative insulation temperature and heating rate solver to model cryogenic tank thermal performance, while interfacing with a Thermal Desktop based vehicle thermal model to predict vehicle temperatures and heat loads. CryoSIM utilizes inputs such as tank geometry, propellant load, material properties, insulation design, internal component (e.g., Thermodynamic Vent System (TVS), Liquid Acquisition Device (LAD), mass gauge (MG)) details, radiation and conduction sink temperatures, and mission duration, to provide estimates for: insulation mass; layer density; fluid temperatures; TVS, LAD, and MG mass and input power parameters; heat loads to insulation, supports, and penetrations; and propellant boiloff mass.

CryoSIM was used to assess the feasibility of storing LOx and LCH₄ in the lunar ascent stage tanks from the launch pad through a full 180 day stay on the lunar surface near a pole, plus transportation and contingency times totaling 240 days. A feasible approach was determined that would load methane below its normal boiling point into tanks protected from radiative heating in space by thick multi-layer insulation (MLI) blankets. Low thermal conductivity structural tank supports were also required, however, the modeling indicated that active refrigeration (cryocoolers) would not be required. NASA awarded contracts to two different vendors to evaluate similar concepts and each arrived at the same conclusion.



Fig. 8: MLSTC tank insulated with 61 layers of MLI.

A thermal vacuum test, referred to as the Methane Lunar Surface Thermal Control (MLSTC) Test, was developed to validate the analytical thermal control predictions for the ascent stage propellant tanks^{xxxvii}. This test was conducted in the NASA GRC Small Multipurpose Research Facility thermal vacuum chamber. As the tank applied thermal insulation system performance historically has a significant variation due to degradation caused by the penetrations through the MLI, gaps due to fit issues (particularly for spherical tanks or dished ends), and local compression of the insulation, the test focused on this aspect of the thermal control problem. The test article was a 1.22 m diameter tank insulated with a high performance 61-layer MLI blanket at an average density of 7.1 layers per inch and procured from and applied by a specialty vendor. Unfortunately the insulation was damaged during test depressurization cycles resulting in areas of significant compression. After 77 days of testing, which included a simulated ground hold, a simulated launch ascent atmospheric pressure profile, four different background temperatures at vacuum, and densified propellants generation, the test series generated much MLI, launch ascent heating, and tank pressurization data. Due to the insulation damage, the heating was higher than pre-test predictions, but analysis based on the as-tested insulation measurements was consistent with test results.

Tank Pressure Control

In the nominal operation scenario, tank pressure is allowed to rise within the tank and control is accomplished by mixing the tank to cool the ullage gas. A TVS was included as a backup system in the event that thermal control was worse than expected.

LCH₄ testing was conducted at the NASA MSFC using the multipurpose hydrogen test bed (MHTB) to evaluate the performance of a spray-bar TVS with subcooled LCH₄ and gaseous helium (GHe) pressurant^{xxxviii}. Thirteen days of testing were

performed, with total tank heat leak conditions of about 715 W and 420 W at a fill level of approximately 90%. A total of 23 TVS cycles were completed. The TVS successfully controlled the ullage pressure within a prescribed control band. A liquid subcooling operation demonstrated the capability of the TVS to remain on for long durations (over 14 hours of continuous operation) and to reduce the liquid saturation pressure to a desired target. The TVS was also successful at maintaining liquid saturation pressure within a control band. These accomplishments were significant since they demonstrated the capability of the TVS to deliver a required temperature and pressure to the engine inlet. During a brief special test, the TVS was used to reduce ullage pressure without the recirculation pump, demonstrating a potential contingency mode.

An extensive liquid oxygen TVS test series was conducted at the NASA GRC SMiRF facility in 2008. These tests successfully demonstrated several key operational characteristics:

- Validating the effects of liquid oxygen properties on TVS performance
- Demonstrated fully autonomous pressure control
- Demonstrated tank fluid pressure control
- Demonstrated the ability to control tank pressure and temperature at the same time (requires helium addition)
- Demonstrated successful operation with both spray-bar and axial jet mixing devices.

PROPELLANT MANAGEMENT DEVICES

For the liquid methane and liquid oxygen propellant combination, an innovative screen sump PMD concept was proposed by the NASA CFM community. The PMD traps liquid and utilizes the capillary force of a dutch twill fine mesh screen LAD to provide liquid only to the RCS during the omni-g LLO maneuvers. The CFM Project completed the following cryogenic fluid acquisition technology tasks:

- 1) Measured the “Bubble Point” pressure (defined as the differential pressure across the screen that overcomes the surface tension of the liquid on the screen) of both subcooled liquid methane and liquid oxygen^{xxxix, xl}
- 2) Investigated the effect of heat entrapped in a screen channel LAD due to engine soak-back or parasitic tank heating

- 3) Investigated the effects of performance degradation of a screen channel LAD due to long term helium solubility in liquid oxygen.
- 4) Measured the flow “rangeability” of a screen channel LAD to permit both the high continuous flow rate required by the Ascent Stage main engine and the short intermittent flow rate required by the RCS thrusters through a single LAD channel.
- 5) In addition, a CFD thermal model of the screen sump PMD concept was completed.

The CFM Project funded Innovative Engineering Solutions, Inc (IES) through a competitive procurement to develop an independent PMD concept based on NASA’s LDAC-1 Altair Ascent Stage configuration.^{xlii} IES selected two PMD types, including a traditional partial four-screen channel device and a novel, expanding volume device that makes use of a stretched, flexing screen. The two selected concepts satisfied all the Altair Ascent Stage design requirements provided by NASA. A significant finding by IES was that advantage could be taken of unique descent and ascent stage design features to simplify the PMD designs. These features are 1) high propellant tank operating pressures, 2) high thermal conductivity aluminum tanks for propellant storage, and 3) stringent insulation requirements. Consequently, it was possible to treat LO2 and LCH4 as if they were equivalent to earth-storable propellants because they would remain substantially sub cooled during the lunar mission: boiling and vapor formation would become non-issues.”

PROPELLANT GAUGING

Although there are several methods for determining liquid level in a cryogenic propellant tank, there are no proven methods to quickly gauge the amount of propellant in a tank while it is in low-gravity.^{xliii} Timely propellant quantity knowledge in low-gravity is considered to be an enabling propulsion system technology for the DRMs. The CFM project successfully matured two technologies for gauging cryogenics in a low gravity environment.

The pressure-volume-temperature (PVT) method of liquid quantity gauging in low-gravity is based on calculations assuming conservation of pressurant gas within the propellant tank and the pressurant supply bottle. This method is currently used to gauge the remaining amounts of storable propellants onboard the Space Shuttle’s orbital maneuvering system and on

Earth-orbiting communications satellites. There is interest in applying this method to cryogenic propellant tanks since it requires minimal additional hardware or instrumentation. Consequently, a PVT gauging experiment with liquid oxygen was completed at NASA GRC using a large-scale cryogenic test tank with an attached cold, high-pressure helium supply bottle. The results indicated that by mixing the tank to achieve isothermal conditions before making the measurement, uncertainties < 2% could be achieved^{xliiii}.

The Radio Frequency Mass Gauge (RFMG) is a novel technology being developed at NASA to enable low gravity propellant quantity gauging^{xliiv}. The RFMG measures the electromagnetic eigenmodes, or natural resonant frequencies, of a tank containing a dielectric fluid using an RF network analyzer that measures the reflected power from an antenna probe mounted internal to the tank. At a tank resonant frequency there is a drop in the reflected power, and these inverted peaks in the reflected power spectrum are identified as the tank eigenmode frequencies using a peak-detection software algorithm. This information is passed to a pattern matching algorithm, which compares the measured eigenmode frequencies with a database of simulated eigenmode frequencies at various fill levels. A best match between the simulated and measured frequency values occurs at some fill level, which is then reported as the gauged fill level. The database of simulated eigenmode frequencies is created by using RF simulation software to calculate the tank eigenmodes at various fill levels. The approach has been validated for oxygen and methane in settled tank testing and for a simulant fluid on an aircraft performing parabolic flights. The results indicate that gauging uncertainties of 1% or better should be possible.

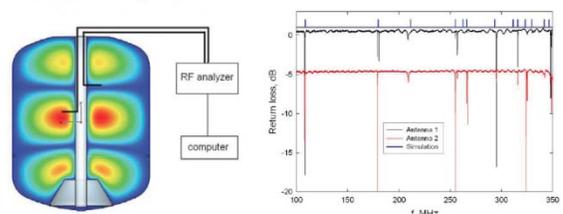


Fig. 9: RF measurement system (left) and sample spectra acquired from the liquid oxygen test tank (right). The Antenna 2 spectrum is offset vertically for clarity. The simulation spectrum shows vertical lines at the calculated eigen frequencies.

PROPELLANT TRANSFER

A final technical challenge for an integrated main and RCS propulsion system is to enable thermally efficient, single-phase, distribution of liquid from the cryogenic storage tank. The CFM Project invested in the design, analysis, development, and test of a propellant distribution system for the cryogenic RCS^{xiv}. The primary issue is delivering properly conditioned cryogenic propellants to the RCS engine inlet interface through long, small diameter feedlines. The approach used a thermodynamic vent to chill the propellant manifold when there was insufficient propellant flow rate and demonstrated that the thermal performance met or exceeded inlet condition requirements for RCS engines.

SUMMARY AND CONCLUSIONS

In 2005, ESAS identified LOx/LCH₄ as a prime candidate propellant combination for integrated reaction control and main pressure fed propulsion of the CEV SM and the LSAM ascent stage. In response to the ESAS, the PCAD project and, subsequently, the CFM project were formed with a focus on technology maturation that would address risks with a LOx/LCH₄ propellant combination.

The PCAD project invested in technologies leading to pre-prototype development of LOx/LCH₄ main engine and RCE systems. Risks of performance and fast start were investigated for main engine technology. RCE technology risks were investigated for performance and pulse width repeatability. Additionally, specific focus was given to reliable ignition for both the main engine and RCE, since it was the highest risk for LOx/LCH₄ propulsion systems, with NASA conducting numerous in-house experimental efforts to examine the issue.

The CFM Project invested in technologies that would reduce risk and satisfy key environmental challenges for LOx/LCH₄ propellant storage and delivery systems proposed in the lunar missions. Specific focus was given to required technology solutions for thermal control, tank pressure control, propellant management, propellant gauging and propellant transfer (engine feed).

Overall, during the course of both the PCAD and CFM projects, there were no significant issues identified that would prohibit the use of LOx/LCH₄ as a propellant combination for a main engine or RCS. Due to the technology risk reduction work conducted in both projects, future missions can consider with more confidence an expanded trade space that now includes LOx/LCH₄ as a propellant combination.

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Liquid Oxygen/Liquid Methane Propulsion and Cryogenic Advanced Development

Exploration Systems Architecture Study conducted by NASA in 2005 identified the liquid oxygen (LOx)/liquid methane (LCH₄) propellant combination as a prime candidate for the Crew Exploration Vehicle Service Module propulsion and for later use for ascent stage propulsion of the lunar lander. Both the Crew Exploration Vehicle and Lunar Lander were part the Constellation architecture, which had the objective to provide global sustained lunar human exploration capability. From late 2005 through the end of 2010, NASA and industry matured advanced development designs for many components that could be employed in relatively high thrust, high delta velocity, pressure fed propulsion systems for these two applications. The major investments were in main engines, reaction control engines, and the devices needed for cryogenic fluid management such as screens, propellant management devices, thermodynamic vents, and mass gauges. Engine and thruster developments also included advanced high reliability low mass igniters. Extensive tests were successfully conducted for all of these elements. For the thrusters and engines, testing included sea level and altitude conditions. This advanced development provides a mature technology base for future liquid oxygen/liquid methane pressure fed space propulsion systems. This presentation highlights the design and test efforts along with resulting hardware and test results.



Liquid Oxygen/Liquid Methane Propulsion and Cryogenic Advanced Development

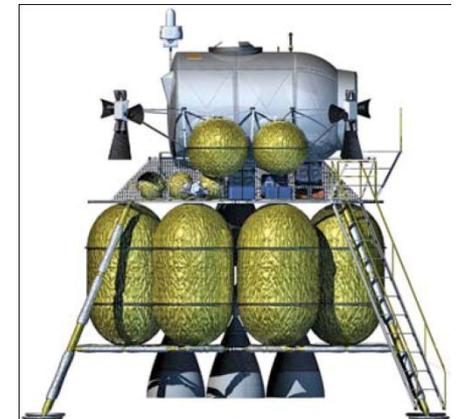
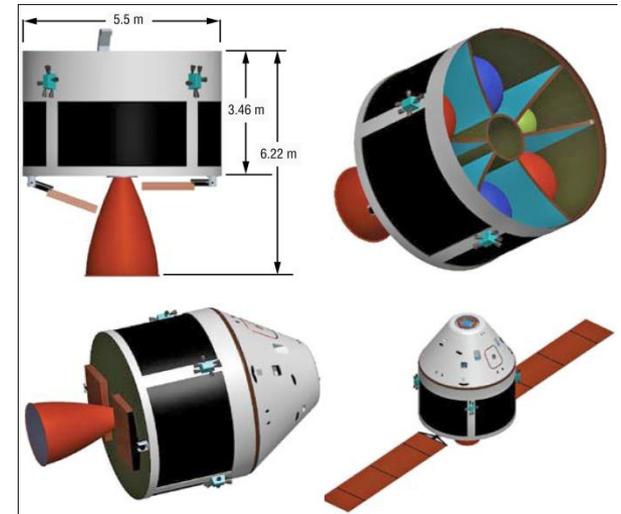
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IAC-11-C4.1.5

LOx/LCH₄ Propellant Combination

- Exploration System Architecture Study (ESAS) in 2005
 - Identified LOx/LCH₄ as prime candidate propellant combination
 - Target propulsion system concept: Integrated reaction control and main pressure fed propulsion system
 - Used for the Crew Exploration Vehicle (CEV) Service Module (SM) and Lunar Surface Access Module (LSAM)
- Mature LOx/LCH₄ propulsion technologies to Technology Readiness Level 6 by CEV Preliminary Design Review
- Advanced Development Projects Created
 - Propulsion and Cryogenics Advanced Development (PCAD)
 - Cryogenic Fluid Management (CFM)



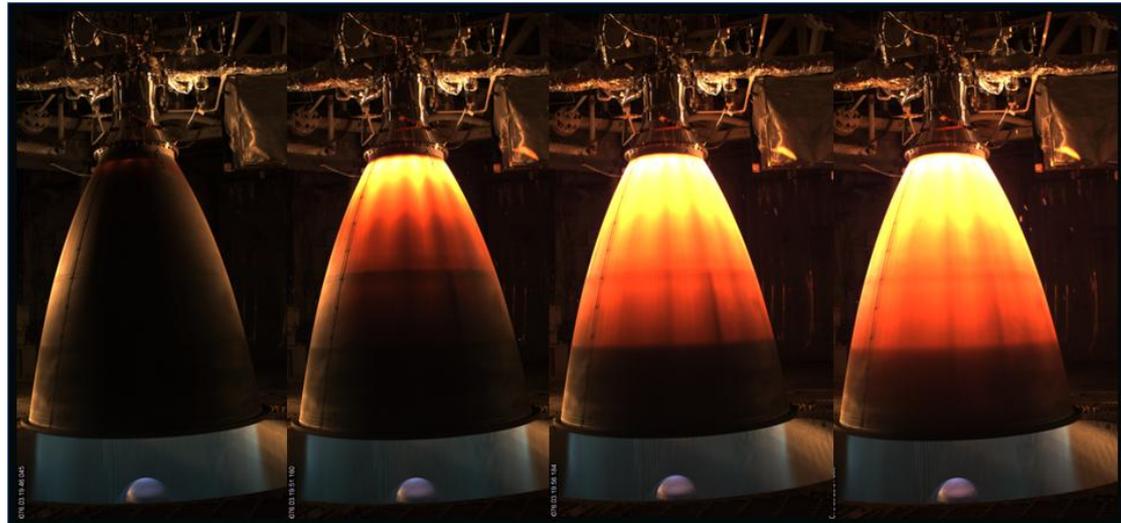
Engine Development Technology Risks

(Addressed in PCAD Project)

- Reliable Ignition
- Performance (Vacuum specific impulse – Isp)
- Fast Start (90% thrust in 0.5-sec) – Ascent Main Engine
- Pulse width repeatability – Reaction Control Engines

LOx/Methane Main Engine Contract

- Aerojet selected to complete a 5,500 pound constant-thrust, pressure-fed workhorse engine
- Phase I: fabrication and sea level testing of multiple injector designs
 - Aerojet fabricated and tested three injectors
 - 48 tests completed, 10-20 sec tests
 - One 110 sec test to determine erosion and life of ablative chamber
- Phase II: Delivery of engine and conducting Altitude Testing at NASA White Sands Test Facility (WSTF)
 - Testing in NASA WSTF at altitude
 - Total of 187 seconds of run time on engine
 - Seven 20-sec tests and one 40-sec test



Aerojet LOx/LCH4 ascent main engine in test at NASA (WSTF)

LOx/Methane Main Engines In-House

NASA LOx/LCH₄ and LOx/GCH₄ Injectors

- Performance and stability of swirl coaxial injector with multiple combustion chamber lengths tested at NASA Marshall Space Flight Center (MSFC)
- Heat transfer data collected
- Microwave and spark torch ignition demonstrated

NASA and Pratt & Whitney Rocketdyne (PWR) Ignition

- ◆ Tested unmodified RS-18 engine in altitude at NASA WSTF
- ◆ Spark torch ignition

Innovative Partnership Program: PCAD, NASA Johnson Space Center, and Armadillo Aerospace

- ◆ Testing of 1,500-lbf thrust-class engine
- ◆ Sea Level testing at Armadillo Facilities
- ◆ Simulated altitude testing at NASA WSTF

Injector Sea Level Test at NASA MSFC

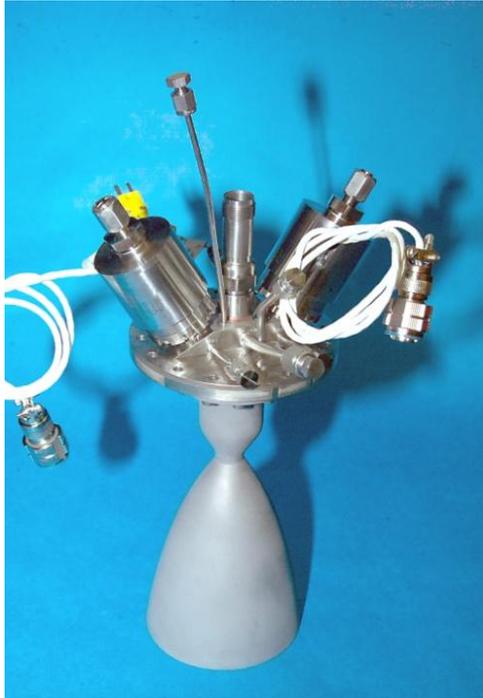


Engine Testing at NASA WSTF. a) PWR RS-18; b) Armadillo Aerospace dual bell nozzle engine

LOx/LCH₄ Reaction Control Engines

Aerojet 100-lbf LOx/LCH₄

- Radiative cooled with Columbium chamber/nozzle
- All key performance criteria demonstrated
- Seven engines delivered



**Aerojet 100-lbf LOx/LCH₄
Reaction Control Engine**

Northrop Grumman 100-lbf LOx/LCH₄

- Dual propellant cooled with Columbium nozzle extension
- Met contract performance requirements and exceeded 317 sec Isp requirement

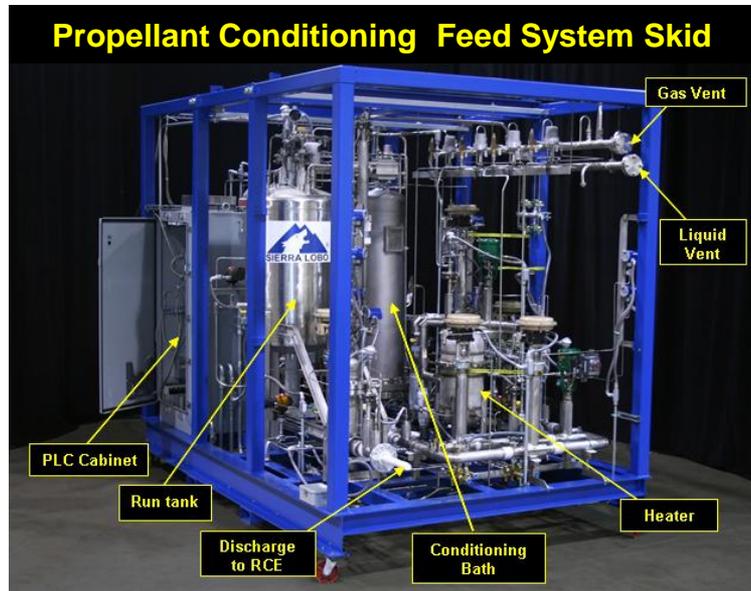
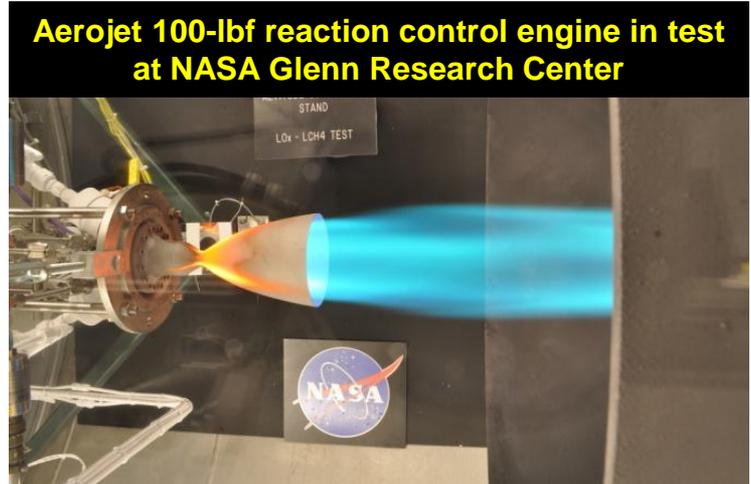


**Northrop Grumman 100-lbf
LOx/LCH₄ Reaction Control
Engine**

LOx/LCH₄ Reaction Control Engine Testing

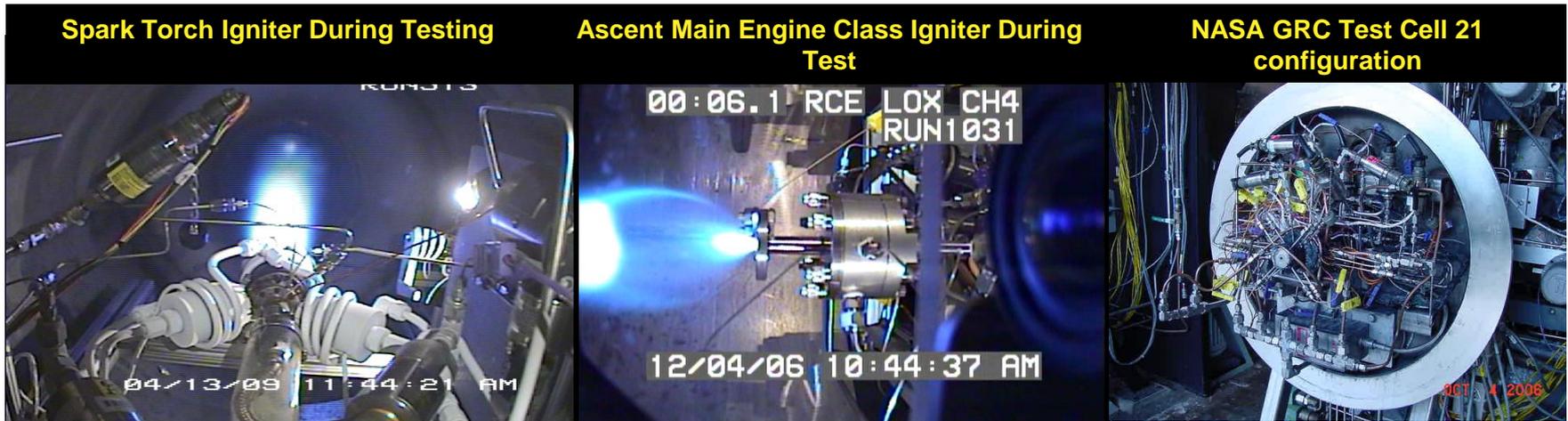
Altitude Performance Testing

- Completed 60 altitude hot-fire tests with Aerojet 100-lb_f engine and propellant conditioning feed systems (PCFS)
- PCFS was designed to control the propellant temperature conditions
 - Oxygen temperature 160-224 R
 - Methane temperature 170-224 R
 - Controlled temperature +/- 5R for a given set point
- Developed Specific Impulse Curves as a function of mixture ratio
- Tested wide range of propellant inlet conditions to simulate operation in a variety of space environments



LOx/LCH₄ Ignition Risk Reduction

- Reliable ignition highest risk for LOx/LCH₄
- Numerous NASA in-house experimental efforts conducted
- Over 30,000 altitude pulse cycles on RCE class spark torch igniter
- NO significant issues identified to prohibit reliable ignition with LOx/LCH₄



Cryogenic Fluid Management Technology Challenges

(Addressed in CFM Project)

- Thermal Environments
 - Thermal Control
 - Tank Pressure Control
- Acceleration Environments
 - Propellant Management Devices
 - Propellant Gauging
 - Propellant Transfer

Thermal Environments

Thermal Control

- ◆ Initiated several analytical modeling activities: full CFD models, lumped capacitance models, and system and component level thermal, structural, and power models
- ◆ Cryogen Storage Integrated Model (CryoSIM) used to assess feasibility of storing LOx and LCH4 for 180 days plus 240 transportation and contingency days
- ◆ Conducted Methane Lunar Surface Thermal Control (MLSTC) test at NASA GRC to validate analytical thermal control predictions

Tank Pressure Control

- ◆ Evaluated performance of Thermal Vent Systems (TVS)
- ◆ LCH4 testing conducted at NASA MSFC
 - ◆ Evaluated spray-bar TVS
 - ◆ Successful control of ullage pressure, long duration operation, reach target liquid saturation temperature
- ◆ LOx testing conducted at NASA GRC
 - ◆ Validated effects of LOx properties on TVS performance
 - ◆ Demonstrated fully autonomous pressure control, tank fluid pressure control, control pressure and temperature at the same time, and successful operation with spray-bar and axial jet mixing

MLSTC Tank Insulated with 61 Layers of MLI



Acceleration Environments

Propellant Management Devices (PMD)

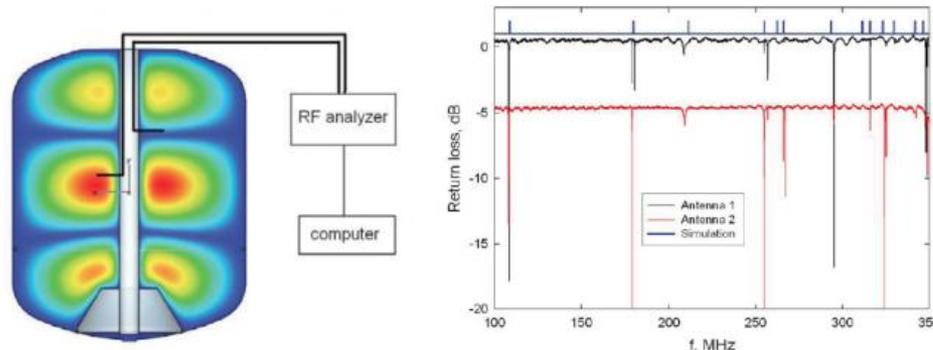
- ◆ Innovative screen sump PMD concept proposed by NASA CFM Community
- ◆ Measured “Bubble Point” of both subcooled LOx and LCH₄
- ◆ Investigated effect of heat entrapment, performance degradation, and flow “rangeability” of liquid acquisition device (LAD)
- ◆ Completed a CFD thermal model of screen sump PMD concept

Propellant Gauging

- ◆ Matured two technologies for gauging cryogenics in a low gravity environment
- ◆ Pressure-volume-temperature (PVT), uncertainties < 2% achieved
- ◆ Radio Frequency Mass Gauge (RFMG), uncertainties of 1% or better possible

Propellant Transfer

- ◆ Design, development, and test of a propellant distribution system for cryogenic RCS
- ◆ Approach used a thermodynamic vent
- ◆ Demonstrated that thermal performance met or exceeded requirements for RCS



RF Measurement system (left) and sample spectra acquired from the LOx test tank (right)

Summary and Conclusions

- In 2005, ESAS indentified LOx/LCH₄ as prime candidate propellant combination
- The PCAD and CFM Projects were formed
 - Focus on technology maturation
 - Address risks with LOx/LCH₄ propellant combination
- PCAD Project
 - Pre-prototype development of LOx/LCH₄ main engine and RCE
 - Risks of reliable ignition, performance, fast start, and pulse width repeatability investigated
- CFM Project
 - Invested in technologies that would reduce risk and satisfy key environmental challenges for LOx/LCH₄ propellant storage and delivery systems
 - Focuses on solutions for thermal control, tank pressure control, propellant management, propellant gauging, and propellant transfer
- No significant issues identified in either project that would prohibit use of LOx/LCH₄ as a propellant combination for main engine or RCS
- *Future Missions have expanded trade space that includes LOx/LCH₄*

